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REVISIONS

Revisions to the original text are indicated by vertical lines in the left margin of the page, adjacent to the revised material.

The manual is provided with a "Log of Revisions" page for recording revisions by number and the dates on which they were inserted in the manual.

INTRODUCTION

GENERAL

The function of this maintenance manual is to acquaint maintenance personnel with the systems and components of the Gulfstream I and to direct them in the proper procedures for maintaining the aircraft in an airworthy condition. Only the installations made in the aircraft during manufacture have been reflected in this manual. Changes made after delivery from the manufacturer are not included. The ability of maintenance personnel is recognized and those procedures which are considered common to all aircraft have been either briefly referenced or omitted.

FORMAT

The manual has been prepared in accordance with Air Transport Association (ATA) Specification No. 100. A functional breakdown is employed whereby all data pertaining to a given system or component may be found in one chapter with a minimum of cross-referencing to other chapters.

The Electrical Power and Hydraulic Power chapters cover only the power sources and distribution equipment for these systems. Details of individual branch hydraulic or electrical systems will be found in the applicable chapters.

IDENTIFICATION OF SUBJECT MATTER

A three-dash number system is employed to identify subject matter. The first dash number identifies the chapter, the second dash number the section of the chapter, and the third dash number the subject of the section. The following example illustrates how the numbering system is used in the LANDING GEAR chapter:

32 - 6 - 2

- 32 — Identifies LANDING GEAR Chapter
- 6 — Identifies that section (group of related subjects) which provides coverage for the "Landing Gear Doors" portion of the landing gear chapter.
- 2 — Identifies a specific subject (component) of the "Landing Gear Doors". In this manual, it is assigned to the landing gear door actuating cylinders.

The "dash zero" (-0) is provided as a means for covering a complete system or sub-system. The chapter number followed by a "-0" will segregate that material covering the complete system; the chapter-section number followed by a "-0" is used for further details covering the subsystem.

PAGE NUMBER IDENTIFICATION

Page number blocks are used to separate the subject matter into the following categories:

General Coverage and Unit Description .... Pages 1 through 100

Troubleshooting .... Pages 101 through 200

Maintenance Practices .... (See Below)

Maintenance Practices include the following sub-topics: Servicing, Removal/Installation, Adjustment/Test, Inspection/Check, Cleaning/Painting, and Approved Repairs.

If all Sub-topics, under Maintenance Practices are brief, they are combined into one topic. All such combined topics are numbered within pages number block 201-200. Whenever individual sub-topics are so lengthy that a combination requires several pages, each sub-topic is treated as an individual topic. Page number blocks used for this sub-topic arrangement are as follows:

Servicing	301-400
Removal/Installation	401-500
Adjustment/Test	501-601
Inspection/Check	601-700
Cleaning/Painting	701-800
Approved Repairs	801-900

Each new subject starts with page 1, 101, 201, etc., and continues through the page block assignment to the extent necessary. The first page of each block is placed on a right-hand page.

FIGURE IDENTIFICATION

Figures (illustrations) are numbered consecutively within each topic (subject) as follows:

Figures in Description	1, 2, 3, 4, 5, etc.
Figures in Troubleshooting	101, 102, 103, etc.
Figures in Maintenance Practices	
When not sub-divided	201, 202, 203, etc.
When sub-divided	
Servicing	301, 302, 303, etc.
Removal/Installation	401, 402, 403, etc.
Adjustment/Test	501, 502, 503, etc.
Inspection/Check	601, 602, 603, etc.
Cleaning/Painting	701, 702, 703, etc.
Approved Repairs	801, 802, 803, etc.

INDEXING

Each chapter is prefaced with a table of contents identifying the subject matter within the chapter in the order of presentation. The table of contents is arranged with the following headings: DESCRIPTION; TROUBLESHOOTING, and MAINTENANCE PRACTICES.

PART NUMBERS

This manual must not be used for identifying spare parts by number. Consult the Gulfstream Illustrated Parts Catalog for this information. Part numbers are used in this manual only as a means of identification when nomenclature alone is inadequate.

LIST OF EFFECTIVE PAGES

CHAPTER 5

TIME LIMITS / MAINTENANCE CHECKS

* Pages revised by Revision 48
** Pages added by Revision 48
*** Pages deleted by Revision 48

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CHAPTER 5

TIME LIMITS / MAINTENANCE CHECKS

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TIME LIMITS/MAINTENANCE CHECKS — INTRODUCTION

1. **General**

This chapter, in general, has been prepared in accordance with the Air Transport Association of America (ATA 100 Specifications).

The maintenance program has been prepared by Gulfstream Aerospace Corporation to provide the operator with information necessary to comply with the requirements of FAR 91, Sub Part C of the General Operating and Flight Rules of the Federal Aviation Regulations.

This maintenance program is applicable to aircraft on a fleet wide basis regardless of the regulations of the country in which the aircraft is registered. Aircraft of Non-United States registry, consult the regulatory authority for the country of registry for requirements.

Life Limited Component requirements (Section 5-4) and Special Inspection requirements (Section 5-5) must be met as stated.

All components not specifically listed are on condition and are monitored during scheduled system operational tests.

Aircraft operating in a tropical zone, high humidity area, marine environment or an area of high industrial pollution should be inspected more frequently.

It is the operator's responsibility to obtain / establish the inspection requirements and time intervals for equipment installed by the operator or the outfitting agency.

Inspection requirements for the engine, navigation, communication, certain avionics equipment and APU will be determined by the appropriate manufacturer unless explicitly noted otherwise.

On any aircraft with more than 2000 landings, whenever any portion of the cockpit or cabin structure, normally concealed behind equipment, furnishings or upholstery, is exposed by modification or repair of the said equipment, furnishings or upholstery then the exposed structure should be given a thorough visual inspection for cracks, corrosion and loose fasteners. Effort should be made, where possible by such measures as lifting the edges of trim panels, to see the structure as far around the opened area as practicable. The extent of the inspection, defined by fuselage stations and stringer numbers, plus aircraft flight hours and number of landings, and the results of the inspection shall be recorded in the aircraft log book.

NOTE: This requirement may be waived if the area has been opened up and inspected within the last 1000 landings.

Revisions to the original text are indicated by vertical lines in the left margin of the page, adjacent to the revised material.

Customer Bulletins are part of this inspection program and are listed in the Customer Bulletin section of the Publication Index.

2. **Terminology**

Overhaul — Remove component from aircraft and disassemble to allow proper checking of clearances, dimensions and for evidence of corrosion, distortion and contamination.

Operational Test — That procedure required to ascertain only that a system or unit is operable.

Functional Test — That procedure required to ascertain that a system or unit is functioning in all aspects in accordance with minimum acceptable system or unit design specifications. These tests may require supplemental ground support equipment and should be more specific and detailed than an operational test.

Hard Time — This concept qualifies the type of component maintenance which imposes fixed limits for component removal.

Scrap — End of useful life.

Inspection / Check — Visual inspection required to ascertain the serviceability of a part, assembly system, specific interrelationship of parts that perform a functional operation, etc.

Service — Servicing operation including: cleaning, lubrication and condition of a component or equipment.

Hydrostatic Testing — That procedure which tests vessels containing fluids under pressure.

Calendar Month — Indicates inspection may be accomplished any day of the month that it is due in.

3. **Section Breakdown**

Section 5-1 — Scheduled Maintenance Checks This section contains all the programmed maintenance operations and frequencies to be carried out on the aircraft.

Newly issued requirements and changes to existing requirements become effective as defined by the new/changed information **following its receipt**.

Aircraft in outfitting centers may use the following:

- Flight hours and landings will remain accumulative from initial log book entries.
- Upon outfitting completion, refer to Maintenance Manual, Chapter 10, Activation of aircraft from storage.

The recommended maintenance operations to be performed on the aircraft are listed in tabular form as follows:

COLUMN 1: This column indicates where the maintenance procedures are covered in the Maintenance Manual, by Chapter and Section.

COLUMN 2: Description of maintenance operations to be performed.

COLUMN 3: CMP (Computerized Maintenance Program) Code to be consulted for procedures for maintenance operation to be performed.

COLUMN 4: Schedule of operations indicated by a "X" in the different inspection columns:

A column: This inspection should be accomplished every 150 flight hours. (Any item in A column may be extended up to 15 hours, extensions may not be cumulative).

B column: This inspection should be accomplished every 9 calendar months. (Any Item in B column may be extended up to 15 days, extensions may not be cumulative).

C column: This inspection should be accomplished every 1000 landings. (Any item in C column may be extended up to 15 landings, extensions may not be cumulative).

NOTE: Certain operations, to be performed every 2nd, 3rd, 4th, etc., are respectively annotated X2, X3, X4 etc. in the corresponding inspection column.

Abbreviations used in this section:

B/S — Bootstrap	REMVL — Removal
EA — Each	O/H — Overhaul
ENG — Engine	L/G — Landing Gear
CHG — Change	

Section 5-2 — Maintenance of Components and Time Between Overhaul

This section list all maintenance operations for items of equipment and provides their frequencies. For equipment requiring a major overhaul, it also states the time between each overhaul.

This section is presented in tabular form as follows:

COLUMN 1: Component part numbers

COLUMN 2: Nomenclature of part

COLUMN 3: Concept of component maintenance

COLUMN 4: Type of operation to be performed

Inspection / Check -- Visual inspection required to ascertain the serviceability of a part, assembly system, specific interrelationship of parts that perform a functional operation, etc.

Service -- Servicing operation including cleaning, lubrication and condition of a component or equipment.

Hydrostatic Testing -- That procedure which tests vessels containing fluids under pressure.

Calendar Month -- Indicates inspection may be accomplished any day of the month that it is due in.

3. **Section Breakdown**

Section 5-1 -- Scheduled Maintenance Checks This section contains all the programmed maintenance operations and frequencies to be carried out on the aircraft.

Newly issued requirements concerning maintenance frequency changes are effective on the date of publication. Action is to be accomplished at the earliest convenient time when maintenance is being accomplished in the same area, or no later than the next interval following the publication date.

Aircraft in outfitting centers may use the following:

- Flight hours and landings will remain accumulative from initial log book entries.
- Calendar time compliance may be adjusted to begin on the date out of outfitting. Exceptions are nine month wing internal inspection and 72 month structural inspection, which start at date of manufacture.
- Upon outfitting completion, refer to Maintenance Manual, Chapter 10, Activation of aircraft from storage.

The recommended maintenance operations to be performed on the aircraft are listed in tabular form as follows:

COLUMN 1: This column indicates where the maintenance procedures are covered in the Maintenance Manual, by Chapter and Section.

COLUMN 2: Description of maintenance operations to be performed.

COLUMN 3: CMP (Computerized Maintenance Program) Code to be consulted for procedures for maintenance operation to be performed.

COLUMN 4: Schedule of operations indicated by a "X" in the different inspection columns:

A column: This inspection should be accomplished every 150 flight hours. (Any item in A column may be extended up to 15 hours, extensions may not be cumulative).

B column: This inspection should be accomplished every 9 calendar months. (Any Item in B column may be extended up to 15 days, extensions may not be cumulative).

C column: This inspection should be accomplished every 1000 landings. (Any item in C column may be extended up to 15 landings, extensions may not be cumulative).

NOTE: Certain operations, to be performed every 2nd, 3rd, 4th, etc., are respectively annotated X2, X3, X4 etc. in the corresponding inspection column

Abbreviations used in this section:

B/S -- Bootstrap	REMVL -- Removal
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Section 5-2 -- Maintenance of Components and Time Between Overhaul

This section list all maintenance operations for items of equipment and provides their frequencies For equipment requiring a major overhaul, it also states the time between each overhaul.

This section is presented in tabular form as follows

COLUMN 1: Component part numbers

COLUMN 2: Nomenclature of part

COLUMN 3: Concept of component maintenance

COLUMN 4: Type of operation to be performed

COLUMN 5: Frequency of the operations given in flying hours, calendar time, landings or as noted. Example: every 2nd B inspection, annotated B(X2)

COLUMN 6: Chapter / Section of the Maintenance Manual (MM) to be consulted for procedures of maintenance operation to be performed.

COLUMN 7: Computerized Maintenance Program (CMP) code to be consulted for procedures of maintenance operation to be performed.

COLUMN 8: Refer to the corresponding Note on page 1 of Section 5-2.

Abbreviations used in this section.

HT — Hard Time	O/T — Operational Test
O/H — Overhaul	F/T — Functional Test
LDGS — Landings	REMVL — Removal
HRS — Hours	ENG — Engine
MOS — Months	NDT — Non-Destructive Test

Section 5-3 — Not Used

Section 5-4 — Life Limited Components

This section lists all components that have retirement (scrap) times controlled by FAA Engineering. The retirement times (frequency) are not subject to change by Gulfstream operators.

Section 5-5 — Special Inspections

This section list those maintenance checks and inspections on the aircraft which are dictated by special or unusual conditions which are not related to the time limits specified in Section 5-1.

4. Maintenance Manuals To Consult

The following publications (as revised) should be consulted for current recommendations relevant to the inspection.

A. Rolls-Royce
   (1) Rolls-Royce PLC Dart MK529 Maintenance Manual (publication M-Da7-G)
   (2) Rolls-Royce Dart Selected Service Bulletins (TSD-1789)

B. Dowty-Rotol
   (1) Dowty-Rotol Ltd. Propeller and Ancillary Equipment Maintenance Manual (865/1)
   (2) Dowty-Rotol Ltd. Accessory Gearbox and Drive Maintenance Manual (865A/1)
   (3) Dowty-Rotol Ltd. Non-modification Service Bulletins 61 and 83 Series

C. Allied-Signal — Garrett General Aviation Service Division
   (1) Garrett Maintenance Manual GTC-85-37-2 APU Maintenance Manual (49-21-51)
   (2) Garrett GTC 85-37-2 Service Bulletins

D. Aircraft Braking Systems
   (1) Nose Landing Gear Wheels — Maintenance Manual (AP-489)
   (2) Main Landing Gear Wheels — Maintenance Manual (AP-489)
   (3) Main Landing Gear Brakes — Maintenance Manual (AP-489)

5. Hydrostatic Testing of Cylinders/Bottles (Charts and Notes)

A. Fire Extinguisher Bottles

PART NUMBER	VENDOR NAME	SPECIFICATION	HYDROSTATIC TEST	SERVICE LIFE
891085	Walter Kidde	4DA900 49CFR178.58	60 Months	On Condition
891114	Walter Kidde	4D500 49CFR178.53	60 Months	On Condition

B. Emergency Blow Down Bottles

PART NUMBER	VENDOR NAME	SPECIFICATION	HYDROSTATIC TEST	SERVICE LIFE
891104	Valcor Engineering (Walter Kidde) P/N 891104	Vendor (See Note Below)	36 Months (See Note Below)	15 Years (See Note Below)
159SCH131-1	Pacific Scientific HLT/KIN-Tech Division P/N 36200158	DOT-E8299-2000	60 Months (See Note Below	15 Years (See Note Below

NOTE: Defined by FAA order 8000.40C Dated 29 May 1992 as:

- All other cylinders/spheres must be inspected and tested as DOT 3HT cylinders/spheres unless more stringent retesting and inspections are specified by a manufacturer for use of its cylinders aboard aircraft.

C. Oxygen Bottles

PART NUMBER	VENDOR NAME	SPECIFICATION	HYDROSTATIC TEST	SERVICE LIFE
ZEP-C250-48	Puritan Bennett Aero Systems Co (Zep Aero)	3A1800 49CFR178 36	60 Months	On Condition
ZC286-48-7	Puritan Bennett Aero Systems Co (Zep Aero)	3HT1850 49CFR178 44	36 Months	24 Years
ZC381-64	Puritan Bennett Aero Systems Co (Zep Aero) (Outfitter Installed)	3HT1850 49CFR178 44	36 Months	24 Years
176203-50 176203-115	Puritan Bennett Aero Systems Co (Outfitter Installed)	DOT-E8162-1850	36 Months (See Note Below)	15 Years (See Note Below)
89518015	Scott Aviation (Outfitter Installed)	DOT-E8162-1850	36 Months (See Note Below)	15 Years (See Note Below)

NOTE: Defined by FAA order 8000 40C Dated 29 May 1992 as:

- Per terms of the exemption for those cylinders/spheres manufactured under an exemption issued by the DOT research and special programs administration.

SCHEDULED MAINTENANCE CHECKS — SECTION 5-1

1. **General**

This Section contains all the programmed maintenance operations and frequencies to be carried out on the aircraft.

Newly issued requirements concerning maintenance frequency changes are effective on the date of publication. Action is to be accomplished at the earliest convenient time when maintenance is being accomplished in the same area, or no later than the next interval following the publication date.

NOTE: All information concerning "Hydrostatic Testing" of oxygen cylinders/bottles, fire extinguishers/bottles and/or air bottles through out this Section (5-1) are referred to Hydrostatic Testing of Bottles (Charts and Notes), in Section 5-0.

2. **Inspection Tables**

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS		
			A	B	C
	CHAPTER 21 — AIR CONDITIONING				
21-1-0	BLOWER OIL FILTER — INSPECTION	210510		X2	
21-1-0	BLOWER ISOLATION VALVE — OPERATIONAL TEST	210531		X2	
21-1-3	LOW FLOW SENSOR CONTROL -- SERVICE	210540		X2	
21-1-2	LOW FLOW BYPASS PRESSURE REGULATOR — FUNCTIONAL TEST	210551		X2	
21-1-4	PRIMARY HEAT EXCHANGER SCREEN — INSPECTION	211015		X2	
21-1-3	BOOTSTRAP UNIT — OIL CHECK / SERVICE	211520		50 Hours	
21-1-3	BOOTSTRAP UNIT — OIL CHANGE	211525	X2		
21-1-1	RAM AIR VALVE — FUNCTIONAL TEST (A/C 1 - 148, 322 and 323)	212510		X2	
21-1-1	RAM AIR CHECK VALVE — FUNCTIONAL TEST (A/C 149 - 200)	212520		X2	
21-1-2	TEMPERATURE CONTROL SYSTEM — OPERATIONAL TEST Cockpit Cabin	 213055 213060		X2	
21-1-0	DIFFERENTIAL PRESSURE SENSOR — FUNCTIONAL TEST	213506		X2	
21-1-0	SPILL VALVE PRESSURE REGULATOR — FUNCTIONAL TEST	213511		X2	
21-2-7	CABIN PRESSURE WARNING — FUNCTIONAL TEST	213526		X2	

"Reference to manufacturers in this Tech. Pub. are for identification purposes only and are neither specified nor furnished as a source for obtaining such parts."

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS		
			A	B	C
21-2-9	OUTFLOW VALVE – INSPECTION			X2	
	Fwd	213571			
	Aft	213591			
	CHAPTER 22 – AUTO FLIGHT				
22-0	AUTOPILOT / FLIGHT DI[illegible]OR -- OPERATIONAL TEST	222005		X2	
	CHAPTER 24 – ELECTRICAL POWER				
24-2-6	NI-CAD BATTE[illegible] – CAPACITY / DEEP CYCLE			200 Hours	
	Left	240506			
	Right	241006			
24-2-6	BATTERY PLUG CABLE – INSPECTION			X2	
	Left	240520			
	Right	241020			
24-2-6	GROUND CABLE – INSPECTION			X4	
	Left Battery	240525			
	Right Battery	241025			
	Left Starter	242005			
	Right Starter	242010			
	Left DC Generator	242551			
	Right DC Generator	243048			
	External Power	242015			
24-2-0	ESSENTIAL DC BUS -- OPERATIONAL TEST	241520		X2	
24-2-1	DC GENERATOR BRUSH – INSPECTION		X2		
	Left	242515			
	Right	243015			
	APU (200 AMP)	497530			
24-2-1	DC GENERATOR SPLINE – LUBRICATION (Non Vespal)		X		
	Left	242520			
	Right	243020			
24-2-2	DC GENERATOR VOLTAGE REGULATOR STABILITY – FUNCTIONAL TEST (Not applicable with Solid State Regulators)			X2	
	Engine-Driven	244010			
	APU	493500			
24-4-1	ALTERNATOR BRUSH – INSPECTION		X2		
	Left	244510			
	Right	245010			
24-2-6	BATTERY COOLING FAN / INVERTER – FUNCTIONAL TEST (A/C Having ASC 204)	247640		X2	

"Reference to manufacturers in this Tech Pub are for identification purposes only and are neither specified nor furnished as a source for obtaining such parts."

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
	CHAPTER 25 — EQUIPMENT FURNISHING				
25-2-1	INERTIA REEL — OPERATIONAL TEST			X2	
	Pilot	250520			
	Copilot	251020			
12-0	SEAT — LUBRICATION			X2	
	Pilot	250525			
	Copilot	251025			
	CHAPTER 26 — FIRE PROTECTION				
26-2-0	FIRE BOTTLE — WEIGHT CHECK		24 Months		
	Left Engine	260510			
	Right Engine	260535			
	APU	261010			
26-2-0	FIRE BOTTLE — HYDROSTATIC TEST (See Hydrostatic Testing of Cylinders/Bottles (Charts and Notes), Section 5-0.)		60 Months		
	Left Engine	260520			
	Right Engine	260545			
	APU	261020			
26-1-0	OVERHEAT SWITCH — TRIP POINT CHECK			X2	
	Left Alternator	265510			
	Right Alternator	265530			
	Left Generator	265520			
	Right Generator	265540			
	Right Blower	266010			
	Fwd Radio Rack	266020			
	Aft Radio Rack	266030			
26-3-0	PORTABLE HALON FIRE EXTINGUISHER — INSPECTION		1 Month		
	Cockpit	263065			
	Cabin No. 1	263090			
	Cabin No. 2	263215			
26-2-0	FIRE EXTINGUISHER SYSTEM — FUNCTIONAL TEST		24 Months		
	Left Engine	260525			
	Right Engine	260550			
	APU	261025			
26-1-0	FIRE T-HANDLE — OPERATIONAL TEST			X2	
	Left	261510			
	Right	261520			

B 1.4

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	INSPECTIONS B	INSPECTIONS C
26-3-0	PORTABLE (CO_2) FIRE EXTINGUISHER – WEIGHT CHECK			X	
	Cockpit	262010			
	Cabin No. 1	262025			
	Cabin No. 2	262032			
26-3-0	PORTABLE (CO_2) FIRE EXTINGUISHER – HYDROSTATIC TEST (See Hydrostatic Testing of Cylinders/Bottles (Charts and Notes), Section 5-0.)		60 Months		
	Cockpit	262015			
	Cabin No. 1	262030			
	Cabin No. 2	263033			
26-3-0	DRY CHEMICAL EXTINGUISHER – HYDROSTATIC TEST (See Hydrostatic Testing of Cylinders/Bottles (Charts and Notes), Section 5-0.)		60 Months		
	Cockpit	263115			
	Fwd	263020			
	Aft	263040			
26-3-0	PORTABLE (H_2O) FIRE EXTINGUISHER – HYDROSTATIC TEST (See Hydrostatic Testing of Cylinders/Bottles (Charts and Notes), Section 5-0.)		60 Months		
	Fwd Cabin	262515			
	Aft Cabin No. 1	262535			
	Aft Cabin No. 2	262550			
26-3-0	PORTABLE (H_2O) FIRE EXTINGUISHER – FUNCTIONAL TEST / SERVICE		24 Months		
	Fwd Cabin	262510			
	Aft Cabin No. 1	262530			
	Aft Cabin No. 2	262545			
26-3-0	PORTABLE HALON FIRE EXTINGUISHER – HYDROSTATIC TEST (See Hydrostatic Testing of Cylinders/Bottles (Charts and Notes), Section 5-0.)		144 Months		
	Cockpit	263055			
	Cabin No. 1	263080			
	Cabin No. 2	263205			
26-3-0	PORTABLE HALON FIRE EXTINGUISHER – WEIGHT CHECK		12 Months		
	Cockpit	263060			
	Cabin No. 1	263085			
	Cabin No. 2	263215			
26-3-0	DRY CHEMICAL EXTINGUISHER – WEIGHT CHECK		12 Months		
	Cockpit	263110			
	Fwd	263015			
	Aft	263035			
26-1-0	FIRE DETECTION SYSTEM – OPERATIONAL TEST	264505	X2		

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	INSPECTIONS B	INSPECTIONS C
26-1-0	ENGINE / ACCESSORY SECTION FIRE EXTINGUISHER SPRAY PIPE — INSPECTION		X2		
	Left	712020			
	Right	714020			
	CHAPTER 27 — FLIGHT CONTROLS				
27-7-0	STALL WARNING POTENTIOMETER — FUNCTIONAL TEST	276025		X2	
27-7-0	STALL WARNING SYSTEM — FUNCTIONAL TEST	277015		X2	
27-2-0	AILERON CONTROL SYSTEM — OPERATIONAL TEST	270505		X2	
27-3-0	RUDDER / RUDDER TABS — INSPECTION		20,000 Hours		
	Rudder	273011			
	Trim Tab	273021			
27-3-0	RUDDER BEARING / FITTING — INSPECTION	273015	3000 Hours		
27-4-0	RUDDER SPRING / TRIM TAB BEARING / FITTING — INSPECTION	273025	3000 Hours		
27-2-0	AILERON BEARING / FITTING — INSPECTION		3000 Hours		
	Left	271010			
	Right	271510			
12-0	AILERON CONTROLS / PUSH ROD — LUBRICATION		X2		
	Left	271015			
	Right	271515			
27-2-0	AILERON — NDT INSPECTION	271008		X2	
27-2-0	AILERON / AILERON TABS — INSPECTION		20,000 Hours		
	Left Aileron	271006			
	Left Trim Tab	271021			
	Left Spring Tab	271041			
	Right Aileron	271506			
	Right Spring Tab	271521			
27-4-0	ELEVATOR TRIM TAB BEARING / FITTING — INSPECTION		3000 Hours		
	Left	272020			
	Right	272520			
27-4-0	ELEVATOR TAB ACTUATOR LUG — INSPECTION (A/C Not having ASC 191)		200 Hours		
	Left	272031			
	Right	272531			
12-0	ELEVATOR CONTROLS / TORQUE TUBE — LUBRICATION	272535	X2		

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
27-3-0	RUDDER CONTROL SYSTEM -- OPERATIONAL TEST	273005		X2	
12-0	RUDDER CONTROL PEDAL -- LUBRICATION	273040		X2	
27-4-0	RUDDER SPRING TAB LOCKOUT CABLE -- INSPECTION (A/C Not having ASC 211)	273050	X2		
27-3-0	RUDDER AND RUDDER TRIM TAB -- NDT INSPECTION	273024		X2	
27-5-0	GUST LOCK CABLE -- INSPECTION	273500		X2	
12-0	CABLE PRESSURE SEAL -- LUBRICATION	273505		X2	
27-6-0	FLAP SYSTEM -- OPERATIONAL TEST	274010		X2	
27-6-0	FLAP ACTUATOR YOKE -- INSPECTION (N/A to solid yoke)			X2	
	Left Outboard	274516			
	Right Outboard	275016			
	Left Inboard	274511			
	Right Inboard	275011			
27-6-0	WING FLAP DEFLECTION CRANK -- INSPECTION			X2	
	Left	571045			
	Right	572045			
12-0	FLAP -- LUBRICATION		X		
	Left	274560			
	Right	275060			
27-6-0	FLAP -- NDT INSPECTION	274020		X2	
27-4-0	AILERON TRIM TAB BEARING / FITTING -- INSPECTION	271025	3000 Hours		
27-4-0	AILERON SPRING TAB BEARING / FITTING -- INSPECTION		3000 Hours		
	Left	271045			
	Right	271525			
27-1-0	ELEVATOR CONTROL SYSTEM -- OPERATIONAL TEST	271540		X2	
27-1-0	ELEVATOR / ELEVATOR TABS -- INSPECTION		20,000 Hours		
	Left Elevator	272006			
	Left Trim Tab	272014			
	Left Bellcrank	272004			
	Right Elevator	272506			
	Right Trim Tab	272514			
	Right Bellcrank	272504			

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
27-1-0	ELEVATOR BEARING / FITTING — INSPECTION		3000 Hours		
	Left	272010			
	Right	272510			
27-1-0	ELEVATOR — NDT INSPECTION	272508		X2	
27-4-0	ELEVATOR TRIM TAB HORN LINKAGE — INSPECTION		X2		
	Left	272016			
	Right	272516			
	CHAPTER 28 — FUEL				
28-2-3	FUEL CROSSFEED VALVE — OPERATIONAL TEST	282025		X2	
28-2-2	FUEL SHUTOFF VALVE — OPERATIONAL TEST			X2	
	Left	282515			
	Right	282525			
28-2-1	FUEL DIFFERENTIAL PRESSURE SWITCH — FUNCTIONAL TEST			X4	
	Left	284015			
	Right	284025			
28-2-1	FUEL LOW PRESSURE SWITCH — FUNCTIONAL TEST			X4	
	Left	284515			
	Right	284525			
28-1-0	FUEL TANK FLAPPER VALVE — INSPECTION			X2	
	Left	287535			
	Right	288035			
	CHAPTER 29 — HYDRAULIC POWER				
29-0	HYDRAULIC POWER SYSTEM — OPERATIONAL TEST	290510		X2	
29-4-0	HYDRAULIC RESERVOIR / FLUID / FILTERS — INSPECTION	291030		X2	
12-0	HYDRAULIC PUMP COUPLING — LUBRICATION (N/A to wet splines)		X		
	Left	291515			
	Right	293515			
29-1-0	HYDRAULIC PUMP COUPLING — INSPECTION		X4		
	Left	291520			
	Right	293520			
29-2-0	AUXILIARY TO MAIN SELECTOR VALVE — OPERATIONAL TEST (Aircraft Having ASC 109)	294050		X2	

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
	CHAPTER 30 — ICE AND RAIN PROTECTION				
30-3-0	WINDSHIELD HEAT SYSTEM — OPERATIONAL TEST	303510		X2	
30-3-0	WINDSHIELD / DV WINDOW RESISTANCE — CHECK	303520		X2	
30-6-0	PROP DE-ICE SYSTEM — OPERATIONAL TEST			X2	
	Left	304015			
	Right	305010			
30-4-0	PITOT / STALL WARNING HEAT — OPERATIONAL TEST			X2	
	Left	305510			
	Right	305515			
	Stall Warn	306010			
30-2-0	WINDSHIELD WIPER MOTOR — SERVICE / INSPECTION			X2	
	Left	306530			
	Right	306540			
30-1-0	DE-ICE VAPOR FILTER — INSPECTION			X2	
	Left	307010			
	Right	307510			
30-1-0	NACELLE / FUSELAGE PNEUMATIC DE-ICE PRESSURE CHECK VALVE — INSPECTION			X2	
	Left Nacelle	307026			
	Right Nacelle	307526			
	Left Fuselage	307031			
	Right Fuselage	307531			
30-1-0	PNEUMATIC DE-ICE PRESSURE REGULATOR VALVE — FUNCTIONAL TEST (P/N 38E20-2 ONLY)			X2	
	Left	307080			
	Right	307585			
30-1-0	Pneumatic DE-ICE DISTRIBUTOR VALVE HOSES — INSPECTION			X2	
	Left Wing Inboard	307085			
	Left Wing Mid	307090			
	Right Wing Inboard	307590			
	Right Wing Mid	307595			
30-1-0	TAIL PNEUMATIC DE-ICE PRESSURE CHECK VALVE — INSPECTION	308012		X2	
30-1-0	PNEUMATIC DE-ICE SYSTEM — OPERATIONAL TEST	309010		X2	
30-5-0	CABIN WINDOW DESICCATOR CRYSTALS — INSPECTION			X2	
	Cabin	309515			
	Left Emergency	309525			
	Right Emergency	309530			

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	INSPECTIONS B	INSPECTIONS C
	CHAPTER 32 — LANDING GEAR				
32-1-0	MAIN GEAR TORQUE ARM — NDT INSPECTION			X2	
	Left Main	320111			
	Right Main	320511			
32-2-0	NOSE GEAR TORQUE ARM — NDT INSPECTION	321016		X2	
32-1-0	MAIN GEAR SHOCK STRUT — SERVICE			X2	
	Left	320115			
	Right	320515			
32-1-0	MAIN GEAR STRUT ACTUATOR ADAPTER — NDT INSPECTION			X2	
	Left	320130			
	Right	320530			
32-1-0	MAIN GEAR UPLOCK ROLL BOLT — INSPECTION				X
	Left Main	320135			
	Right Main	320535			
32-2-0	NOSE GEAR UPLOCK ROLL BOLT — INSPECTION	325575			X
32-3-0	MAIN GEAR DOOR TRUNNION / SWIVEL / ROD END — INSPECTION (A/C 1 — 111 and 114 Not having ASC 164)			X2	
	Left	320140			
	Right	320540			
32-3-0	MAIN GEAR DOOR TRUNNION / SWIVEL / ROD END — INSPECTION (A/C 1 — 111 and 114 having ASC 164 and A/C 112 — 200)			X2	
	Left	320141			
	Right	320541			
32-1-0	MAIN GEAR DRAG BRACE — INSPECTION			X2	
	Left	320150			
	Right	320550			
32-1-0	MAIN GEAR DRAG BRACE, ROD END BEARING — INSPECTION			X2	
	Left	320152			
	Right	320552			
12-0	MAIN GEAR SWIVEL FITTING — LUBRICATION		X		
	Left	320155			
	Right	320555			
12-0	MAIN GEAR — LUBRICATION		X		
	Left	320160			
	Right	320560			
32-2-0	NOSE GEAR SHOCK STRUT — SERVICE	321015		X2	

"Reference to manufacturers in this Tech. Pub. are for identification purposes only and are neither specified nor furnished as a source for obtaining such parts."

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
32-2-0	NOSE GEAR OUTER CYLINDER — NDT INSPECTION	321025		X2	
12-0	NOSE GEAR - LUBRICATION	321035	X		
32-0	LANDING GEAR SYSTEM — FUNCTIONAL TEST	321045		X2	
32-2-0	NOSE GEAR STRUT ACTUATOR MOUNTING ADAPTER — NDT INSPECTION	321050		X2	
32-13-0	MAIN GEAR WHEEL — INSPECTION				X
	Left Outboard	321520			
	Left Inboard	321525			
	Right Outboard	321540			
	Right Inboard	321545			
32-13-0	NOSE GEAR WHEEL — INSPECTION (Every other tire change after 1500 ldgs, every tire change after 2000 ldgs)			1500 Ldgs (←SEE NOTE)	
	Left	322015			
	Right	322025			
32-14-1	ANTI-SKID / VALVE FILTER — INSPECTION			X2	
	Left	322054			
	Right	322555			
32-14-0	ANTI-SKID SYSTEM — OPERATIONAL TEST (Aircraft having ASC 206)	322556		X2	
32-12-0	BRAKE PRESSURE LINE RESTRICTOR / FILTER — INSPECTION	323046		X4	
32-12-0	WHEEL BRAKE SYSTEM — OPERATIONAL TEST	323050		X2	
32-11-0	NOSE WHEEL STEERING PIN — NDT INSPECTION	323516			X2
32-11-0	NOSE WHEEL STEERING LUG LOWER PIN — NDT INSPECTION	323518			X
32-11-0	NOSE WHEEL STEERING — OPERATIONAL TEST	323530		X2	
32-7-0	LANDING GEAR EMERGENCY AIR BOTTLE — HYDROSTATIC TEST (See Hydrostatic Testing of Cylinders/Bottles (Charts and Notes), Section 5-0)	323545		60 Months	
32-7-0	LANDING GEAR EMERGENCY AIR BOTTLE — INSPECTION	323550		X2	
32-5-1	MAIN GEAR ACTUATOR — INSPECTION			X2	
	Left	324536			
	Right	325016			

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	INSPECTIONS B	INSPECTIONS C
32-5-5	MAIN / NOSE GEAR TIMER VALVE -- FUNCTIONAL TEST (N/A to A/C having Valves P/N 1159H20265-3)			X2	
	Left Main Up	324560			
	Left Main Down	324570			
	Right Main Up	325035			
	Right Main Down	325045			
	Nose Up	325535			
	Nose Down	325545			
32-5-5	MAIN GEAR TIMER VALVE BUNGEE LINKAGE -- INSPECTION			X2	
	Left	324571			
	Right	325046			
32-2-0	NOSE GEAR STRUT TRUNNION -- NDT INSPECTION (P/N 171467 Only)	326005	100 Hours		
32-1-0	MAIN GEAR UPLOCK BEAM SUPPORT ANGLES -- NDT INSPECTION (A/C 1-83 Not having ASC 179 only)		100 Hours		
	Left	326010			
	Right	326015			
	CHAPTER 33 — LIGHTS				
33-2-3	EMERGENCY EXIT LIGHTS -- OPERATIONAL TEST	331006		X	
33-2-3	EMERGENCY EXIT LIGHTS INERTIA SWITCH -- INSPECTION			X2	
	Main Door	331012			
	Overhead Hatch	331022			
	Left Cabin Emergency Exit No. 2	331032			
	Left Cabin Emergency Exit No. 3	331037			
	Right Cabin Emergency Exit No. 2	331042			
	Right Cabin Emergency Exit No. 3	331047			
	Baggage Door Emergency Exit	331052			
	Forward Cabin Emergency Exit	331057			
	Aft Cabin Emergency Exit	331062			
	External Exit Light	331075			
	Left Cabin Emergency Exit No. 1	331082			
	Lavatory Emergency Exit	331085			
33-3-0	EXTERIOR LIGHTS -- OPERATIONAL TEST			X	
	Top Anti-Collision	331515			
	Aft Anti-Collision	331525			
	Bottom Anti-Collision	331535			
	Left Landing Light	332015			
	Right Landing Light	332515			
	Strobe Lights	333060			
	Taxi Lights	335015			
	Fuselage/Nacelle Wing Inspection Lights	335525			
	Identification Lights (A/C Having ASC 208 Only)	335530			

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
12-0	LANDING LIGHT [illegible] VICE / LUBRICATION	332520	X2		
	CHAPTER 34 — NAVIGATION				
34-1-0	OVERSPEED WARNING SYSTEM — FUNCTIONAL TEST	347030		24 Months	
34-1-0	PITOT / STATIC SYSTEM — LEAK CHECK / FUNCTIONAL TEST			24 Months	
	No 1 Pitot	348510			
	No 2 Pitot	349025			
	No 1 Static	348515			
	No 2 Static	349030			
34-1-0	ALTERNATE STATIC SYSTEM — LEAK CHECK / FUNCTIONAL TEST			24 Months	
	No 1	348520			
	No 2	349035			
34-1-0	ATC TRANSPONDER — TEST (FAR 91 413)			24 Months	
	No 1	342030			
	No 2	342035			
34-1-0	ALTIMETER — TEST (FAR 91 411)			24 Months	
	No 1	347515			
	No 2	349015			
	No 3	349018			
	CHAPTER 35 — OXYGEN				
35-1-1	OXYGEN CYLINDER — HYDROSTATIC TEST (3AA only) OXYGEN CYLINDER — HYDROSTATIC TEST (3HT and DOT-E8162-1850) (See Hydrostatic Testing of Cylinders/Bottles (Charts and Notes), Section 5-0)			60 Months X4	
	No 1	351015			
	No 2	351016			
	No 3	351046			
35-0	OXYGEN SYSTEM — OPERATIONAL TEST	354010		X2	
35-0	PASSENGER OXYGEN SYSTEM — OPERATIONAL TEST	355010		X2	
	CHAPTER 49 — AIRBORNE AUXILIARY POWER				
49-0	APU ENCLOSURE — INSPECTION	490105 494574 494575		250 APU Hours	
49-0	APU AIR INLET DOOR ACTUATOR — OPERATIONAL TEST	494006		X2	

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
49-0	APU FUEL PRESS'..'E REGULATOR / SHUT-OFF VALVE — FUNCTIONAL TEST			X2	
	APU External Shut-off Valve	494016			
	APU Pressure Regulator	494025			
	CHAPTER 52 — DOORS				
52-1-0	MAIN ENTRANCE / BAGGAGE DOOR -- OPERATIONAL TEST	521010		X2	
52-1-0	MAIN ENTRANCE DOOR HINGE PINS — INSPECTION	521015		X4	
12-0	MAIN ENTRANCE DOOR — SERVICE / LUBR'CATION	522010		X	
52-1-0	MAIN ENTRANCE DOOR — INSPECTION	522020		X2	
52-1-0	MAIN ENTRANCE DOOR CABLE — TENSION CHECK	522025		X2	
12-0	BAGGAGE DOOR — SERVICE / LUBRICATION	523010		X	
52-5-0	OVERHEAD ESCAPE HATCH — OPERATIONAL TEST (N/A for A/C having ASC 197)	525010		X2	
52-5-0	OVERHEAD ESCAPE HATCH STRUCTURE — INSPECTION (N/A for A/C having ASC 197)	525015		X2	
52-5-1	EMERGENCY ESCAPE DOOR -- OPERATIONAL TEST (Aircraft Having ASC 240 Only)	527005		X2	
5-5-1	EMERGENCY ESCAPE DOOR - INSPECTION (Aircraft Having ASC 240 Only)	527010	X		
	CHAPTER 53 — FUSELAGE				
53-0	MAIN GEAR WHEEL WELL — INSPECTION			X	
	Left Main	320165			
	Right Main	320565			
53-0	NOSE GEAR WHEEL WELL -- INSPECTION	321040		X	
53-0	COCKPIT ABOVE FLOOR -- INSPECTION	531010		X	
53-0	COCKPIT BELOW FLOOR / PEDESTAL — INSPECTION			X2	
	Cockpit	531015			
	Pedestal	531020			
53-0	COCKPIT BEHIND PANEL -- INSPECTION	531025		X2	

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
53-0	ENTRANCE COMPARTMENT ABOVE FLOOR -- INSPECTION	531510		X	
53-0	FUSELAGE STATION 169 BULKHEAD -- INSPECTION (N/A for A/C not having entrance way threshold light)	531511		X2	
53-0	ENTRANCE COMPARTMENT BELOW FLOOR -- INSPECTION	531515		X2	
53-0	CABIN ABOVE FLOOR -- INSPECTION Fwd Aft	 532010 532510		X	
53-0	CABIN BELOW FLOOR — INSPECTION Fwd Mid Aft	 532015 532020 532515		X2	
53-0	BAGGAGE COMPARTMENT ABOVE FLOOR -- INSPECTION	533010		X	
53-0	BAGGAGE COMPARTMENT BELOW FLOOR -- INSPECTION	533015		X2	
53-0	TAIL INTERIOR (ROUTINE) — INSPECTION	533510		X	
53-0	TAIL INTERIOR (DETAILED) — INSPECTION	533515		X2	
53-0	FUSELAGE / WING CENTER SECTION (EXTERIOR) — INSPECTION	534010		X	
53-0	FUSELAGE / WING CENTER SECTION (INTERIOR) — INSPECTION	534015		X2	
53-0	LAVATORY COMPARTMENT BELOW FLOOR -- INSPECTION	534020		X2	
53-0	WING / FUSELAGE FITTING -- INSPECTION	534040		X2	
53-0	AFT FUSELAGE — NDT INSPECTION	535015		X2	
53-0	PRESSURE DOME -- INSPECTION	534050			X4
	CHAPTER 54 — NACELLES				
54-0	FIREWALL RECEPTACLE -- INSPECTION Left Right	 711022 713022	Eng Removal		
54-0	ENGINE GRUMMAN MOUNT -- INSPECTION (More than 7500 landings) Left Right	 711500 713500	12 Months		

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
54-0	ENGINE GRUMMAN MOUNT INSPECTION (Less than 7500 landings)			72 Months	
	Left	711535 thru 711595			
	Right	713535 thru 713585			
54-0	ENGINE MOUNT CHAFE GUARD INSPECTION			2500 Hours	
	Left	715010			
	Right	715015			
54-0	NACELLE INSPECTION			X2	
	Left	541010			
	Right	541510			
54-0	ENGINE MOUNT TRUSS INSPECTION			Eng Removal	
	Left	541015			
	Right	541515			
54-0	ENGINE ACCESSORY SECTION / NACELLE INSPECTION			X	
	Left	711025			
	Right	713025			
	CHAPTER 55 — STABILIZERS				
55-1-0	VERTICAL STABILIZER / RUDDER (EXTERIOR) INSPECTION	551015		X	
55-1-0	VERTICAL STABILIZER / RUDDER (INTERIOR) INSPECTION	551020		X2	
55-1-0	VERTICAL STABILIZER / FUSELAGE FITTING X-RAY INSPECTION (Initial inspection at 9000 Hours)			4000 Hours	
	Fwd	551025			
	Mid	551030			
	Aft	551035			
55-1-0	VERTICAL STABILIZER HINGE SUPPORT FITTING INSPECTION			2000 Hours	
	Lower	551511			
	Mid	551516			
	Upper	551521			
55-1-0	VERTICAL STABILIZER NDT INSPECTION	551005		X2	
55-2-0	HORIZONTAL STABILIZER / ELEVATOR EXTERIOR INSPECTION			X	
	Left	552015			
	Right	552515			

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	B	C
55-2-0	HORIZONTAL STABILIZER / ELEVATOR INTERIOR -- INSPECTION			X2	
	Left	552020			
	Right	552520			
55-2-0	HORIZONTAL STABILIZER -- NDT INSPECTION	552005		X2	
55-2-0	ELEVATOR HINGE FITTING -- INSPECTION		20,000 Hours		
	Left Inboard	552023			
	Left Mid	552028			
	Left Outboard	552033			
	Right Inboard	552523			
	Right Mid	552528			
	Right Outboard	552533			
55-2-0	HORIZONTAL STABILIZER ATTACH FITTING -- X-RAY INSPECTION (Initial inspection at 9000 Hours)		4000 Hours		
	Left Fwd	552025			
	Left Mid	552030			
	Left Aft	552035			
	Right Fwd	552525			
	Right Mid	552530			
	Right Aft	552535			
55-2-0	ELEVATOR STOP FITTING -- INSPECTION		20,000 Hours		
	Left	552540			
	Right	552545			
	CHAPTER 56 -- WINDOWS				
56-0	EMERGENCY EXIT WINDOW RELEASE -- OPERATIONAL TEST			X2	
	Left	562510			
	Right	564010			
	CHAPTER 57 -- WINGS				
57-0	WING INTERIOR -- INSPECTION			X2	
	Left	571015			
	Right	572015			
57-0	WING (UNDER HEAT SHIELD) -- INSPECTION		Eng Chg		
	Left	571016			
	Right	572016			
57-0	OUTER WING FWD BEAM -- VISUAL OR X-RAY CORROSION INSPECTION			X2	
	Left	571017			
	Right	572017			

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	INSPECTIONS B	INSPECTIONS C
57-0	INBOARD WING BEAM / WING PANEL -- INSPECTION			X2	
	Left	571020			
	Right	572020			
57-0	LOWER WING PLANK CORROSION – INSPECTION		60 Months		
	Left	571021			
	Right	572021			
57-0	LOWER WING PLANK SPLICE – NDT INSPECTION	573515		X2	
57-0	OUTBOARD WING BEAM / WING PANEL – INSPECTION			X2	
	Left	571025			
	Right	572025			
57-0	AILERON – INSPECTION			X2	
	Left	571030			
	Right	572030			
57-0	FUEL TANK CONTAMINATION – INSPECTION			X	
	Left	571035			
	Right	572035			
57-0	WING / FLAP EXTERIOR – INSPECTION	571010		X	
	Left	571040			
	Right	572010			
		572040			
57-0	WING / FUSELAGE ATTACH FITTING – INSPECTION (Initial inspection at 9500 Hours)		4000 Hours		
	Left 159WM10016-1	571510			
	Left 159WM10017-1	571515			
	Left 159WM10018-1	571520			
	Left 159WM10019-1	571525			
	Left 159BM10000-1	571530			
	Left 159BM10001-1	571535			
	Right 159WM10016-2	572510			
	Right 159WM10017-2	572515			
	Right 159WM10018-2	572520			
	Right 159WM10019-2	572525			
	Right 159BM10000-2	572530			
	Right 159BM10001-2	572535			
57-0	LOWER WING CENTER SECTION – INSPECTION	573510	12 Months		

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	INSPECTIONS B	INSPECTIONS C
	CHAPTER 71 — POWER PLANT				
71-0	POWER PLANT EXTERIOR — INSPECTION		X		
	Left	711015			
	Right	713015			
71-0	POWER PLANT INTERIOR — INSPECTION		X		
	Left	711020			
	Right	713020			
71-1-0	ENGINE MOUNT (VICKERS) — INSPECTION		72 Months		
	Left	711506			
	Right	713506			
71-1-0	ENGINE MOUNT (VICKERS) ATTACHMENT HARDWARE — NDT INSPECTION		Eng Removal		
	Left Engine Mount Hollow Retaining Bolts	711510			
	Left Engine Locking Bolts	711515			
	Left Engine Mount Pin Assemblies	711520			
	Right Engine Mount Hollow Retaining Bolts	713510			
	Right Engine Locking Bolts	713515			
	Right Engine Mount Pin Assemblies	713520			
	CHAPTER 75 — AIR				
75-0	ENGINE HOT AIR GATE VALVE — OPERATIONAL TEST		X2		
	Left	750515			
	Right	750525			
	CHAPTER 76 — ENGINE CONTROLS				
12-0	ENGINE CONTROLS — LUBRICATION		X		
	Left	760510			
	Right	760520			
76-0	ENGINE CONTROLS — INSPECTION		X		
	Left	760510			
	Right	760520			
76-0	ENGINE POWER LEVER / HP COCK CABLE — TENSION CHECK			X2	
	Left Power Lever	761010			
	Right Power Lever	761020			
	Left HP Cock	761015			
	Right HP Cock	761025			
	CHAPTER 77 — ENGINE INDICATING				
77-5-0	ENGINE TGT INDICATION SYSTEM (EMF) — FUNCTIONAL TEST			X2	
	Left	772515			
	Right	772535			

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS		
			A	B	C
77-1-0	ENGINE RPM INDICATOR -- FUNCTIONAL TEST			X2	
	Left	773050			
	Right	773055			
77-3-0	FUEL TRIM SYSTEM -- OPERATIONAL TEST	732036		X2	
	CHAPTER 78 — EXHAUST				
78-0	ENGINE TAILPIPE BELLOWS -- INSPECTION			X2	
	Left	781510			
	Right	782510			
78-0	ENGINE TAILPIPE AND TAILPIPE BLANKET -- INSPECTION		Eng Chg		
	Left Fwd Tailpipe	781520			
	Left Aft Tailpipe	781525			
	Right Fwd Tailpipe	782520			
	Right Aft Tailpipe	782525			
	Left Blanket	781515			
	Right Blanket	782515			
	CHAPTER 82 — WATER INJECTION				
82-0-1	WATER/METHANOL CELL/TANK PUMP -- INSPECTION			X2	
	Left Cell	821515			
	Tank	822015			
	Right Cell	822010			
	Tank	822025			
82-6-1	WATER/METHANOL QUANTITY INDICATION -- OPERATIONAL TEST	822535		X4	
82-2-0	WATER/METHANOL LOW PRESSURE SWITCH -- FUNCTIONAL TEST			X2	
	Left	823015			
	Right	823025			
82-0-1	WATER/METHANOL SYSTEM CHECK VALVE -- OPERATIONAL TEST			X2	
	Left	823035			
	Right	823045			
82-5-0	WATER/METHANOL FILTER -- INSPECTION			X2	
	Left	824025			
	Right	824525			
82-2-0	WATER/METHANOL CROSS FLOW CHECK VALVE -- OPERATIONAL TEST			X2	
	Left	825020			
	Right	825025			
82-0-1	WATER/METHANOL LINES -- PRESSURE TEST / INSPECTION		3000 Hours		
	Left	827040			
	Right	827045			

MAINTENANCE OF COMPONENTS AND TIME BETWEEN OVERHAUL — SECTION 5-2

1. **General**

This section list all maintenance operations for items of equipment and provides their frequencies. For equipment requiring a major overhaul, it also states the time between each overhaul.

The following Notes are used throughout this Section 5-2 and are denoted as applicable in COLUMN 8.

NOTES COLUMN 8

(1) Installed on Main Landing Gear 173576 or 2570894

(2) Installed on Main Landing Gear 172242 or 2570893

(3) Part No. 7507212 and 57734-2 used in 64684 assembly. Part No. 573545 used in 032678 assembly.

(4) On Aircraft 1 - 93, not having CB 266.

(5) Cartridges have a life limit of 72 months from date stamped on cartridge and storage temperatures do not exceed 100° F. replacements and expense. Removal of the vertical stabilizer may be coordinated with horizontal stabilizer removal at a major structural inspection (4000 Ldg) not later than 12 years.

(6) See Hydrostatic Testing of Cylinder/Bottles (Charts and Notes), Section 5-0, for additional information on oxygen cylinders/bottles, fire extinguishers/bottles and/or air bottles.

2. **Component Tables**

PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
	CHAPTER 21 AIR CONDITIONING						
139153	Supercharger (Blower)	HT	O/H Inspect Oil Filter	2500 hrs B (X) B (X2)	21-1-0 71-0 21-1-0	210505 713025 210510	
205710	Bootstrap Unit	HT	O/H Oil Change Oil Service O/T	4500 hrs A (X2) 50 hrs	21-1-3 21-1-3 21-1-3 21-1-2 21-1-2	211505 211525 211520 213055 213060	
HYCL6629-1	Actuator, Ram Air Limiter	HT	O/H O/T	5000 hrs B (X2)	21-1-2 21-1-2	212530 213055 213060	
107312-1	Relay, Pneumatic	HT	O/H Inspect	6000 hrs B (X)	21-2-8	213560 213580 531510	
103262-1	Valve, Outflow	HT	O/H Inspect	6000 HRS B (X2)	21-2-9 21-2-9	213570 213590 213571 213591	

PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
137847-1	Filter, Outflow Valve	HT	Scrap	5000 rs	21-2-9	213575 213595	
	CHAPTER 24 ELECTRICAL POWER						
30E20-41A	Generator, DC	HT	O/H	1800 hrs	24-2-1	242505 243005	
			Insp Brushes	A (X2)	24-2-1	242515 243015	
			Lube Splines	A (X)	12-0	242520 243020	
30007-003	Generator, DC	HT	O/H	1000 hrs	24-2-1	242505 243005	
			Insp Brushes	A (X2)	24-2-1	242515 243015	
			Lube spline	A (X)	12-0	242520 243020	
BAI00IEXP2	Alternator	HT	O/H	1250 hrs	24-4-1	244505 245005	
			Insp Brushes	A (X2)	24-4-1	244510 245010	
			F/T	A (X)	24-4-1	244535 245035	
SE-20-2	Inverter (1500 VA)	HT	O/H	4000 hrs	24-3-1	245510 246005	
MGE-23-3	Inverter (2500 VA)	HT	O/H	4000 hrs	24-3-1	245510 246005	
SE-16-3 MGH-229-100	Inverter (250 VA)	HT	O/H	4000 hrs	24-3-1	247005	
MGE-23-400	Inverter (2500 VA)	HT	O/H	4000 hrs	24-3-1	245510 246005	
	CHAPTER 26 FIRE PROTECTION						
890275	Extinguisher H_2O	HT	O/H	6 Months		262510 262530 262545	
892480	Extinguisher H_2O	HT	O/H	24 Months		262510 262530 262545	

Reference to manufacturers in this Tech. Pub. are for identification purposes only and are neither specified nor furnished as a source for obtaining such parts.

PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
841155	Extinguisher Cartridge	HT	Scrap	24 Months		260515 thru 261015	(5)
	CHAPTER 27 FLIGHT CONTROLS						
159SCC100-1 159SCC100-5	Actuator, Trim Tab - Aileron	HT	O/H Inspect	120 Months B (X2)	27-4-0	271030 571030	
159SCC100-1 159SCC100-5	Actuator Trim Tab - Elevator	HT	O/H Scrap	120 Months 20,000 hrs	27-4-0	272025 272525 272026 272526	
1159SCC100-11 1159SCC100-13 1159SCC100-15 1159SCC100-17 1159SCC100-19	Actuator, Trim Tab - Elevator	HT	O/H Inspect	120 Months B (X2)	27-4-0	272025 272525 552020 552520	
159SCC100-1 159SCC100-5 159RDC105-1	Actuator Trim Tab - Rudder	HT	O/H Inspect	120 Months B (X2)	27-4-0	273030 551020	
159RDCS174 (GIC ONLY)	Tab, Rudder Spring/Trim	HT	Scrap Insp	20,000 hrs B (X2)	27-4-0	273020 551015	
159C10307-1	Cable S/Tab Lockout - Rudder (A/C not having ASC 211)	HT	Replace Inspect	3000 hrs A (X2)	27-4-0 27-4-0	273045 273050	
159AM10004-1 159AM10003-1 159AM10002-1 159AM10001-1	Carriage - Flap	HT	Scrap Inspect	35,000 ldgs B (X)		274540 274545 274550 274555 571010	
159AM10004-2 159AM10003-2 159AM10002-2 159AM10001-2	Carriage - Flap	HT	Scrap Inspect	35,000 ldgs B (X)		275040 275045 275050 275055 572010	

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PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
159CSM10054 159CSM10053 159CSM10052 159CSM10051	Fitting - Flap Carriage	HT	Scrap	20,000 ldgs		274520 274525 274530 274535 275520 275525 275530 275535	
159SCAM100-1 A/C 1-5 159SCAM100-3 A/C 6 & Sub	Gear Box - Flap Drive	HT	Scrap	20,000 ldgs	27-6-0	275510	
	CHAPTER 28 FUEL						
123571-110-01 RR12030B1 159P10023-S3	Pump - Fuel Boost Normal	HT	O/H	2000 hrs	28-2-1	281015 281010	
123571-110-01 RR12030B1 159P10023-S3	Pump - Fuel Boost Aux	HT	O/H	2000 hrs	28-2-1	281030 281530	
	CHAPTER 29 HYDRAULIC POWER						
57735-2	Element - Filter	HT	Scrap	12 Months	29-4-0	291021	
AN6237-1	Element,Filter - Reservoir / Relief	HT	Replace	8000 hrs	29-4-0	291040	
159SCH100-3 159SCH100-5	Pump - Hydraulic	HT	O/H Inspect Coupling Lube Coupling	4000 hrs A (X4) A (X)	29-2-0 29-1-0 12-0	291510 293510 291520 293520 291515 293515	
B574615 B57737-2	Element, Filter - Main Press	HT	Scrap	12 Months	29-1-0	291531 293531	
7507212 573545	Element Filter Aux Press	HT	Scrap	12 Months	29-2-0	294021	

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PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
	CHAPTER 30 ICE AND RAIN PROTECTION						
A706-1	Controller, Windshield / DV / Side - Window Heat	HT	O/H	6000 HRS	30-3-0	301015 301515 301030 302015 302515	
			O/T	B (X2)	30-3-0	303510	
B708-1C	Power Supply, D.V. Window Heat	HT	O/H	7500 HRS	30-3-0	301020 302010	
			O/T	B (X2)	30-3-0	303510	
38E01-7	Valve De-Ice Safety	HT	O/H	6000 hrs	30-1-0	307020 307520	
38E35-1 38E32-2	Press Regulator Valve De-Ice	HT	O/H	6000 hrs	30-1-0	307015 307515	
			O/T	B (X2)	30-1-0	309010	
38E20-2	Regulator, Pressure De-Ice	HT	O/H	1500 hrs	30-1-0	307075 307580	
			F/T	B (X2)	30-1-0	307080 307585	
	CHAPTER 32 LANDING GEAR						
173625 172252	Adaptor, Actuator, Main Gear	HT	Scrap	2500 ldgs	32-1-0	320125 320525	(1) (2)
			Insp NDT	B (X2)	32-1-0	320130 320530	
	Hardware, Actuator Main Gear	HT	Scrap	2500 ldgs	32-1-0	320127 320527	
171552	Adapter, Nose Strut Actuator (A/C 1 - 23 only)	HT	Scrap Insp	5000 ldgs B (X2)	32-2-0	321029 321050	
159L10003-7 159L10003-11	Drag Brace Assy, Main Gear	HT	O/H	6000 ldgs	32-1-0	320145 320545	
			Insp	B (X2)	32-1-0	320150 320550	

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PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
159SCL103	Strut, Shock Main Gear	HT	O/H	6000 ldgs	32-1-0	320110 320510	
			Insp NDT			320110 320510	
			Service	B (X2)	32-1-0	320115 320515	
			F/T	B (X2)	32-0	321045	
159SCL104	Strut, Nose Gear Shock	HT	O/H Insp NDT Service F/T	6000 ldgs B (X2) B (X2) B (X2)	32-2-0 32-2-0 32-2-0 32-0	321010 321025 321015 321045	
	Hardware, Actuator Nose Gear	HT	Scrap	2500 ldgs		321060	
11606 159SCH108	Valve, Brake Shuttle	HT	O/H	7000 ldgs	32-12-0	322040 322525 322055 322545	
			F/T	B (X2)	32-12-0	323050	
	Valve, Nose Gear Door Shuttle	HT	O/H F/T	7000 ldgs B (X2)	32-5-3 32-0	324515 321045	
159SCH119 32351	Steering Unit, Nose Wheel	HT	O/H F/T	7000 ldgs B (X2)	32-11-0 32-11-0	323510 323530	
171619	Pin, Lower Steering Lug	HT	Scrap Insp	4000 ldgs 1000 ldgs	32-11-0	323535	
891104 159SCH131-1	Air Bottle, Landing Gear Emergency	HT	Scrap Insp Hydro Test	15 years B (X2)	32-7-0 32-7-0 32-7-0	323540 323550 323545	(6) (6)
159SCH112	Valve, Manual Reset Dump	HT	O/H F/T	10,000 ldgs B (X2)	32-7-0 32-0	324010 321045	
159SCH123	Valve, Hydraulic Dump	HT	O/H F/T	10,000 ldgs B (X2)	32-5-0 32-0	324015 321045	
159H10043-1	Actuator, Main Gear Door	HT	O/H	6000 ldgs	32-3-2	324025 324040	
			F/T	B (X2)	32-0	321045	
12654	Valve, Main Gear Door Act Shuttle	HT	O/H	7000 ldgs	32-5-3	324030 324045	
			F/T	B (X2)	32-0	321045	

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PART NUMBER	NOMENCLATURE	REQMTS		FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
159SCH115	Valve, Main Gear Door Act Select	HT	O/H F/T	4000 ldgs B (X2)	32-3-0 32-0	324035 324050 321045	
159H10029	Actuator, Nose Gear Door	HT	O/H F/T	5000 ldgs B (X2)	32-3-2 32-0	324510 321045	
159SCH114	Valve, Selector, Nose Gear Door	HT	O/H F/T	4000 ldgs B (X2)	32-2-0 32-0	324520 321045	
159H10026-5 159H10026-9 159H10026-11 159H10026-12 159H10026-15	Actuator, Main Gear	HT	O/H Insp	7000 ldgs B (X2)	32-5-1	324535 325015 320165 320565	
159H10026-7	Actuator, Main Gear	HT	O/H Insp	3000 ldgs B (X2)	32-5-1	324535 325015 320165 320565	
159H10042	Actuator, Gear Uplock	HT	O/H F/T	7000 ldgs B (X2)	32-5-2 32-0	324540 325020 321045	
12681	Valve, Gear Uplock Shuttle	HT	O/H F/T	7000 ldgs B (X2)	32-5-3 32-0	324545 325025 321045	
159H10035-1 159H10035-3	Valve, Gear Timer Main / Nose	HT	O/H F/T Scrap	2000 ldgs B (X2) 6000 ldgs	32-5-5 32-0	324550 324565 325030 325040 325530 325540 321045 324551 325031 324566 325041 325531 325541	
7507212 57734-2 573545	Filter Elements, N/W Steering		Scrap	12 Months		323517	(3)
159H10027	Actuator, Nose Gear	HT	O/H Insp F/T	7000 ldgs B (X) B (X2)	32-5-1 32-0	325510 321040 321045	

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PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
HP644100	Valve, Gear Act, Shuttle - Nose	HT	O/H F/T	7000 ldgs BX2	32-5-3 32-0	325515 321045	
159H10032	Actuator, Gear Uplock, Nose	HT	O/H F/T	7000 ldgs B (X2)	32-5-2 32-0	325520 321045	
HP644100	Valve, Gear Uplock Shuttle - Nose	HT	O/H F/T	7000 ldgs B (X2)	32-5-2 32-0	325525 321045	
159SCH110	Valve, Gear Selector - Nose	HT	O/H F/T	10,000 ldgs B (X2)	32-2-0 32-0	325560 321045	
159SCH126	Valve, Main / Speed Brake Sel.	HT	O/H F/T	10,000 ldgs B (X2)	32-1-0 32-0	325565 321045	
	CHAPTER 33 LIGHTS						
950 or Equiv	Battery Emergency Exit Light Non-chargeable	HT	Scrap	6 Months	33-2-3	331061	
	CHAPTER 35 OXYGEN						
ALAR-AI2A	Regulator, Oxygen	HT	O/H	36 Months	35-2-1	353010 353510 353520	
			Insp O/T	B (X) B (X2)	35-0	531010 354010	
ZC268-48-7 ZC381-64	Cylinder, Oxygen	HT	Scrap	24 Years	35-1-1	351010 351011 351045	(6)
			Hydro Test	36 Months		351015 351016 351046	(6)
176203-50 176203-115 89518015	Cylinder, Oxygen	HT	Scrap	15 Years	35-1-1	351010 351011 351045	(6)
			Hydro Test	36 Months		351015 351016 351046	(6)

PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
	CHAPTER 57 WINGS						
159BM10000 159BM10001 159WM10016 159WM10017 159WM10018 159WM10019	Fittings, Fuselage to Wing at Fuselage Station 288.5 and 353.0	HT	Scrap	50,000 hrs	53-0	573010	
			Insp	4000 hrs	57-0	571510 571515 571520 571525 571530 571535 572510 572515 572520 572525 572530 572535	
159WM10079 159WM10080 159WM10081 159WM10082 159WM10083 159WM10084	Track, Wing Flap	HT	Scrap Insp	35,000 ldgs B (X)	57-0 57-0	573015 571010 572010	
	CHAPTER 61 PROPELLERS						
6042H	Relay, Prop Feathering Pump	HT	Scrap	10,000 hrs	61-0	612520	
FC400	Relay, Propeller Control		Scrap	3000 hrs	61-0	616510 616515	
			F/T	A (X4)	61-0	616010 616510	
	CHAPTER 73 ENGINE FUEL AND CONTROL						
162C25	Indicator, Fuel Temp	HT	O/H	16,000 hrs	73-3-0	730520 731020	
	CHAPTER 76 ENGINE CONTROLS						
159P10078-1 159P10078-39	Cable, Power Lever	HT	Scrap	1200 hrs	76-0	761030 761035	(4)
			Tension Test	B (X2)	76-0	761010 761020	

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PART NUMBER	NOMENCLATURE		REQMTS	FREQ	MM	CMP	SEE NOTE 5-2 PAGE 1
1CZ135470/B 500/1/00925/001 500/1/04446/001	Actuator, Fuel Trim	HT	O/H O/H O/H O/T	1200 hrs 1800 hrs 5000 hrs B (X2)	76-1-0 77-3-0	732025 732040 732036	
	CHAPTER 77 **ENGINE INDICATING**						
KB153/1 KGA0401	Generator, Tachometer	HT	O/H	5000 hrs	77-1-0	773010 773020	
159SCF104 (Not Having Mod 6)	Indicator, Tachometer	HT	O/H Insp / Service	1500 hrs 500 hrs	77-1-0	773015 773025	
159SCF104 (Having Mod 6)	Indicator, Tachometer	HT	O/H Insp	5000 hrs B (X)	77-1-0	773015 773025 531010	
	CHAPTER 80 **STARTING**						
NB34	Starting Ignition Unit	HT	O/H O/T	3000 hrs 300 hrs	80-1-1 80-0-1	802510 803010 802530 803030 802550 803050	
	CHAPTER 82 **WATER INJECTION**						
RG-11870	Pump, Water/Methanol (Aircraft Not Having ASC 125)	HT	O/H Insp O/T	3000 hrs B (X2) B (X2)	82-3-1	820510 821010 821515 822010 825020 825025	
RG-11870	Pump, Water/Methanol (Aircraft Having ASC 125)		O/H Insp O/T	3000 hrs B (X2) B (X2)	82-3-2	820520 821020 822015 822025 825020 825025	

SECTION 5-3 — NOT USED

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LIFE LIMITED COMPONENTS — SECTION 5-4

1. **General**

   This Section lists all components that have retirement (scrap) times controlled by FAA Engineering. The retirement times (frequency) are not subject to change by Gulfstream operators.

   Aircraft without a record of the number of landings, a figure of one landing per 1.24 flying hours is considered acceptable.

2. **Life Limited Components Tables**

PART NUMBER	NOMENCLATURE	LIMITED SERVICE LIFE (SCRAP)	MAINT MANUAL CHAPT / SECT	CMP CODE
	CHAPTER 27 FLIGHT CONTROLS			
159RDCS174 (GIC ONLY)	Rudder Spring/Trim Tab	20,000 hrs	27-4-0	273020
	CHAPTER 32 LANDING GEAR			
172252 173625	Main Gear Strut Actuator Mounting Adapter	2500 ldgs	32-1-0	320125 / 320525
	CHAPTER 53 FUSELAGE			
	Wing - Fuselage Connection	50,000 hrs	53-0	573010
	CHAPTER 54 NACELLES			
1159W10432-1 1159W10432-2	Engine Mount (Category A Grumman Mount)	20,000 ldgs	54-4-0	711530 / 713530 711540 / 713540 711550 / 713550 711560 / 713560 711570 / 713570 711590 / 713580
1159W10432-1 1159W10432-2	Engine Mount (Category B Grumman Mount)	7500 ldgs	54-4-0	711530 / 713530 711540 / 713540 711550 / 713550 711560 / 713560 711570 / 713570 711590 / 713580
	CHAPTER 57 WINGS			
	Flap Tracks	35,000 ldgs	57-0	573015

PART NUMBER	NOMENCLATURE	LIMITED SERVICE LIFE (SCRAP)	MAINT MANUAL CHAPT / SECT	CMP CODE
159AM10004-1	Flap Carriages	35,000 ldgs	27-6-0	274540 / 275040
159AM10003-1				274545 / 275045
159AM10002-1				274550 / 275050
159AM10001-1				274555 / 275055
159AM10004-2				
159AM10003-2				
159AM10002-2				
159AM10001-2				
159CSM10054	Flap Carriage Fittings	20,000 ldgs	27-6-0	274520 / 275020
159CSM10053				274525 / 275025
159CSM10052				274530 / 275030
159CSM10051				274535 / 275035

SPECIAL INSPECTIONS — SECTION 5-5

1. General

This section list those maintenance checks and inspections on the aircraft which are dictated by special or unusual conditions which are not related to the time limits specified in Section 5-1.

2. Hard and/or Overweight Landing — Inspection Procedure

A. Main gear attachment fittings and the supporting ribs inside the wing. Drag strut attachment fittings at rear beam.

B. Front and rear beams inboard of nacelles for permanent buckling of webs. Front and rear beam splices at the dihedral break.

C. Wing to fuselage main attachment fittings and bolts.

D. Fuselage in area of wing for permanent buckling of skin.

E. Look for fuel leaks from integral tank as possible evidence of excessive deformation.

NOTE: Because the wing skin is of integral design, it is unlikely that any deformation would be visible in the skin.

F. Nose gear and drag strut attachments to fuselage.

G. Engine mounting structure and its attachment to wing. The top tubes of the engine mounting should be given a careful examination for over tension and the welds examined for cracks.

H. Chin fairing (on lower front of engine firewall) inspect attaching fasteners.

3. Aircraft Severe Turbulence Reported — Inspection

A. Front and rear beams inboard of nacelles for permanent buckling of webs. Front and rear beam splices at the dihedral break.

B. Wing to fuselage main attachment fittings and bolts.

C. Fuselage in area of wing for permanent buckling of skin.

D. Look for fuel leaks from integral tank as possible evidence of excessive deformation.

NOTE; Because the wing skin is of integral design, it is unlikely that any deformation would be visible in the skin.

E. Engine mounting structure and its attachment to the wing. The top tubes of the engine mounting should be given a careful examination for over tension and the welds examined for cracks.

F. Chin fairing (on lower front of engine firewall). Inspect attaching fasteners.

G. Outer wing splice. Check fasteners both externally and internally.

H. Inner wing skin splice at dihedral break. Check fasteners both externally and internally.

I. Fuselage skin for permanent buckles.

J. Control surfaces and connections to main structure.

4. Reported Lightning Strike — Inspection procedure

Structure and equipment in the vicinity of a strike should be inspected.

On the Gulfstream the strike pattern usually begins at radome and continues down the fuselage. An exit is found most often in the form of a hole at the top of the rudder. However, all extremities should be examined.

The radome honeycomb will delaminate if lightning conductors have not been installed. if lightning conductors have been installed, a check for delamination is still recommended. Inspect the fuselage and wing skins, the propeller, antennas, and lead in cables for signs of pitting or overheating.

5. Rudder Following A Gust Lock Failure — Inspection

A Gulfstream I may experience a failure of its rudder gust lock latch and rudder lower hinge attaching bolt. The most plausible cause is that it was exposed to jet blast from a passing aircraft or high gusting wind while parked with the gust lock on

Excessive torque applied to rudder while parked with gust lock on is likely to fail the gust lock latch, P/N 159CM10521-1 (See Figure 1).

If this failure should occur, lower half of rudder and fin structure supporting the two lower hinges must be inspected before further flight

Inspect rudder skin aft of the rudder front beam for distortion and loose or working rivets and the vertical tail overhang structure for distortion.

Only one side needs a close inspection, the side, one would have to push on, to produce the latch failure (tension or compression). Inspect vertical stabilizer skin just forward of the rear beam in vicinity of the two lower hinges for local buckles and loose or working rivets.

These will occur on the opposite side to possible damage on the rudder and fin trailing edge overhang.

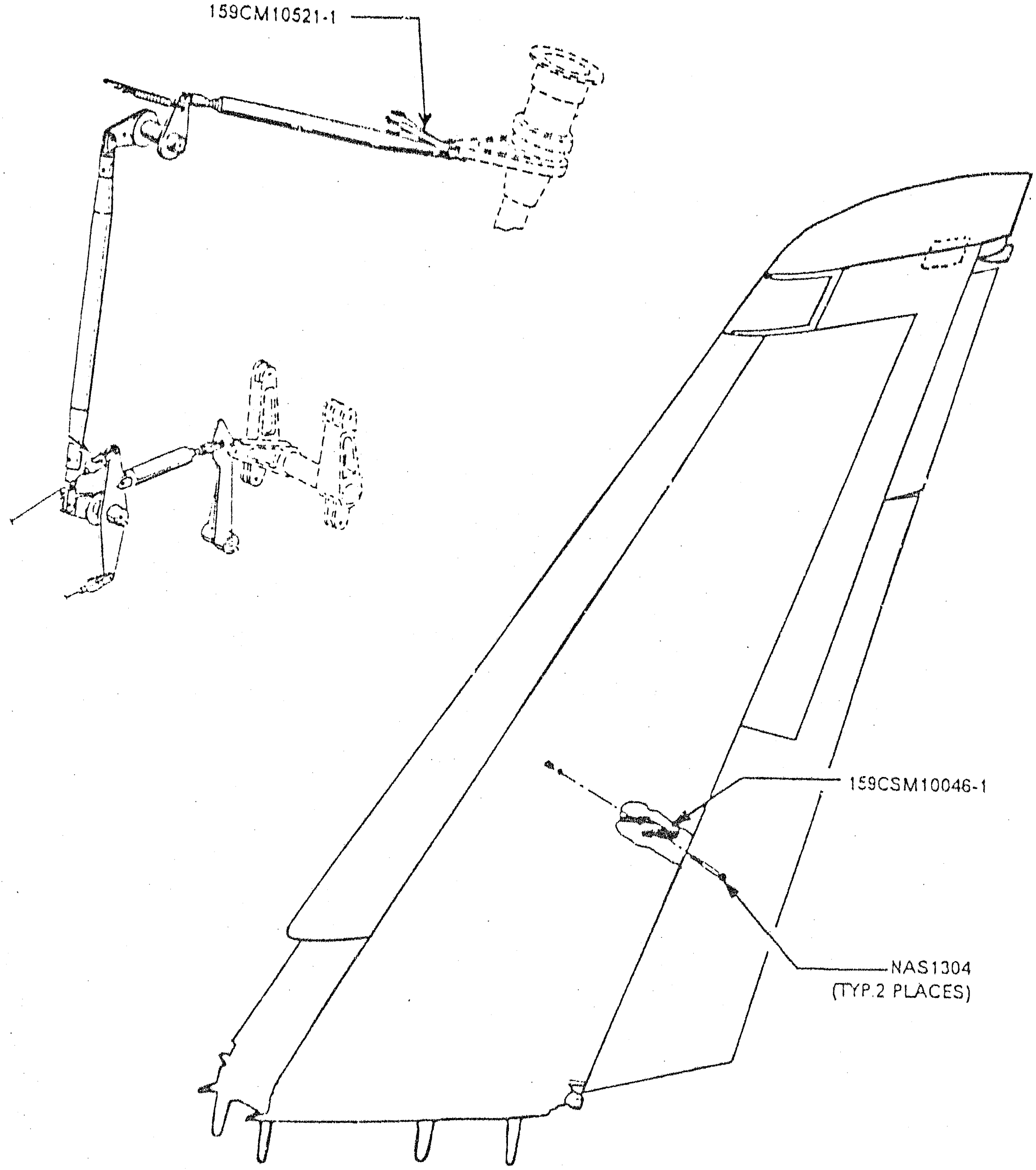
Open lower two removable trailing edge overhang panels, and check the structure supporting two lowest rudder hinges

Check rivets attaching fittings, P/N 159CSM10046-1 and 159CSM10048-1, (See Figure 1), to the sheet metal support diaphragms

Withdraw NAS1304-92D and NAS1304-120D Bolts and inspect them visually for signs of overload. In case a hole is worn or damaged, a 1/64 inch oversize bolt (NAS2904) may be used.

CAUTION: ENSURE BOLT IS NOT DAMAGED IN THE PROCESS OF WITHDRAWAL.

NOTE: It is recommended that when parking the aircraft at high density airports, operators take appropriate precautions to avoid exhaust gas impingement.



Gust Lock Latch and Bolt - Installation
Figure 1

MAINT MANUAL CHAPT / SECT	DESCRIPTION OF MAINTENANCE PROCEDURES	CMP CODE	INSPECTIONS A	INSPECTIONS B	INSPECTIONS C
82-6-1	WATER/METHANOL QUANTITY INDICATION — OPERATIONAL TEST	822535		X4	
82-2-0	WATER/METHANOL LOW PRESSURE SWITCH — FUNCTIONAL TEST			X2	
	Left	823015			
	Right	823025			
82-0-1	WATER/METHANOL SYSTEM CHECK VALVE — OPERATIONAL TEST			X2	
	Left	823035			
	Right	823045			
82-5-0	WATER/METHANOL FILTER — INSPECTION			X2	
	Left	824025			
	Right	824525			
82-2-0	WATER/METHANOL CROSS FLOW CHECK VALVE — OPERATIONAL TEST			X2	
	Left	825020			
	Right	825025			
82-0-1	WATER/METHANOL LINES — PRESSURE TEST / INSPECTION		3000 Hours		
	Left	827040			
	Right	827045			

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LIST OF EFFECTIVE PAGES

CHAPTER 6

DIMENSIONS AND AREAS

* Pages revised by Revision 48
** Pages added by Revision 48
*** Pages deleted by Revision 48

TABLE OF CONTENTS

CHAPTER 6

DIMENSIONS AND AREAS

DIMENSIONS AND AREAS — DESCRIPTION / OPERATION

1. **General**

The Gulfstream I is a pressurized passenger transport aircraft, powered by two Rolls-Royce Dart Mk 529 (RDa-7/2) turbo propeller engines.

The fuselage is divided into six separate compartments: nose, cockpit, entrance, cabin, baggage, and tail. (See Figure 1)

The nose compartment is at the forward end of the fuselage and begins at Fuselage Station 8 and ends at Fuselage Station 63, which is the forward pressure bulkhead location.

The cockpit begins at Fuselage Station 63 and ends at Fuselage Station 133. It is designed to carry a minimum crew of two.

The entrance compartment is just aft of the cockpit. It begins at Fuselage Station 133 and ends at Fuselage Station 181. The entrance door is on the left side of the entrance compartment, and a radio equipment area is located at the forward right side of the entrance compartment.

The passenger cabin is just aft of the entrance compartment. It begins at Fuselage Station 181 and continues on to Fuselage Station 525. This compartment is arranged for a maximum of 19 passengers. Space is also provided for a galley, clothes closet, and lavatory.

The baggage compartment begins at Fuselage Station 525 and ends at Fuselage Station 561, which is the aft pressure bulkhead location.

The tail compartment is an unpressurized area that begins at Fuselage Station 561 and ends at Fuselage Station 745. Components found in various sub-systems are located in this compartment. A tail compartment door is provided for easy access to these components. All compartments with the exception of the nose and tail compartments are pressurized to maintain a cabin altitude of 8000 feet with an aircraft altitude of 30,000 feet.

Two integral fuel tanks, one in each wing, contain a total usable capacity of 10,462.4 pounds (1550.0 US gallons). On Aircraft 1 - 200, 322 and 323 having ASC 125 the total fuel capacity is 12,106.8 pounds (1793.6 US gallons). Each tank is divided into compartments to minimize shifts in c.g. with changes in aircraft attitude, a crossflow system connecting the two pumping systems is provided to allow the fuel to flow from one tank to the opposite engine or both engines.

Bladder type cells in each wing outer panels contain the water methanol. Aircraft 1 - 106 and 114 not having ASC 165, have dual cells with a total capacity of 84 gallons. Aircraft 107 - 200, 322 and 323 and Aircraft 1 - 106 having ASC 165 have single cells with total capacity of gallons.

Two integral tanks (one each side) located in the wing root fillet area contain the water methanol, for Aircraft 1 - 200, 322 and 323 having ASC 125 with a total capacity of 47.4 gallons.

The elevators, ailerons, and rudder are conventionally operated and are manually controlled through the use of cables, bellcranks, and push rods. Fowler-type flaps are provided on the trailing edge of each wing.

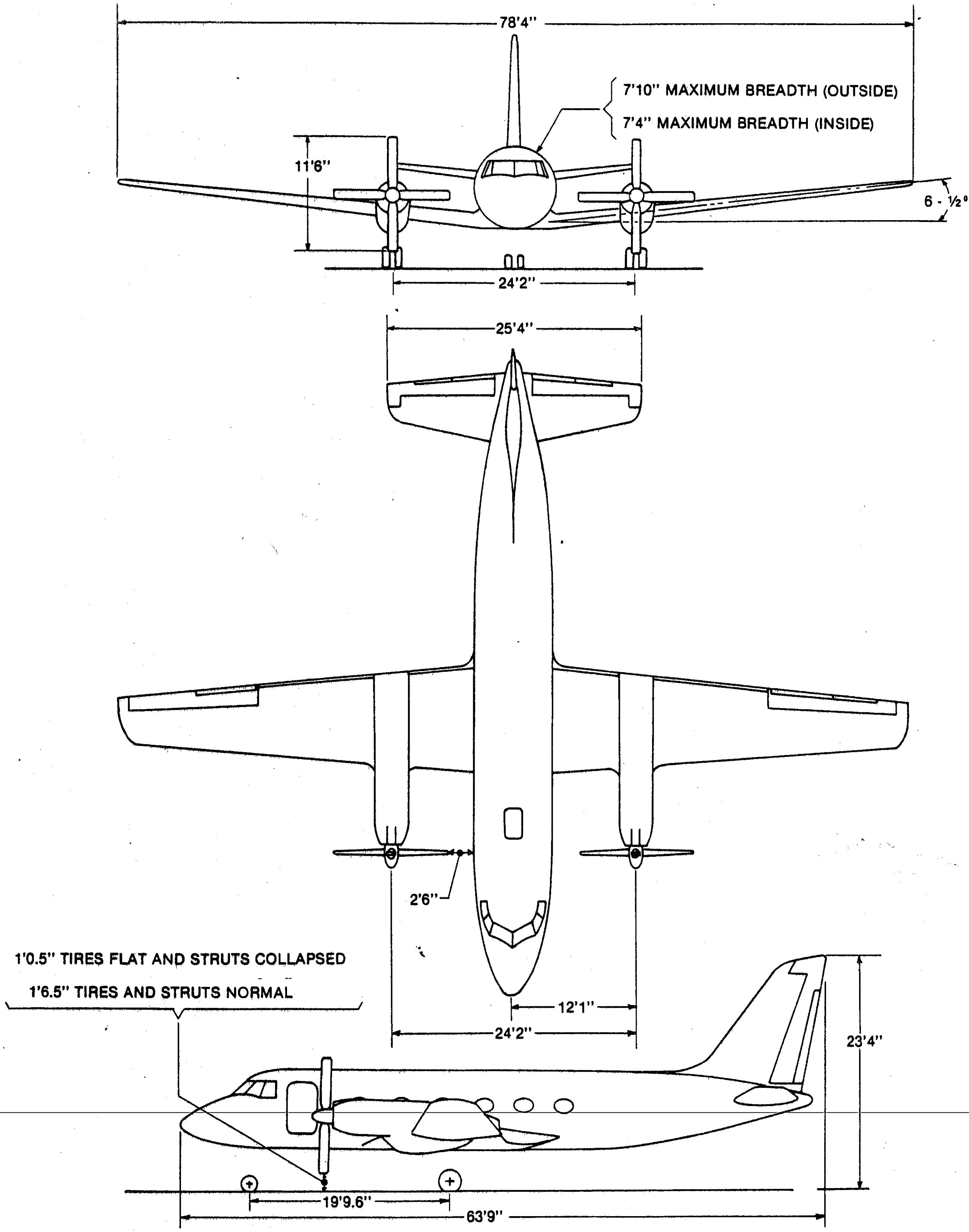
A tricycle landing gear has dual nose and main gear wheels. The main gear retracts forward into the engine nacelles and the nose gear retracts forward into the fuselage. The nose gear is steered hydraulically.

An auxiliary power unit, located in the tail of the aircraft, is installed to provide full pressurization in flight in the event of engine-driven blower failure. In addition, the APU drives a dc generator which is used to recharge the batteries. The APU can also be used for air conditioning when the aircraft is on the ground.

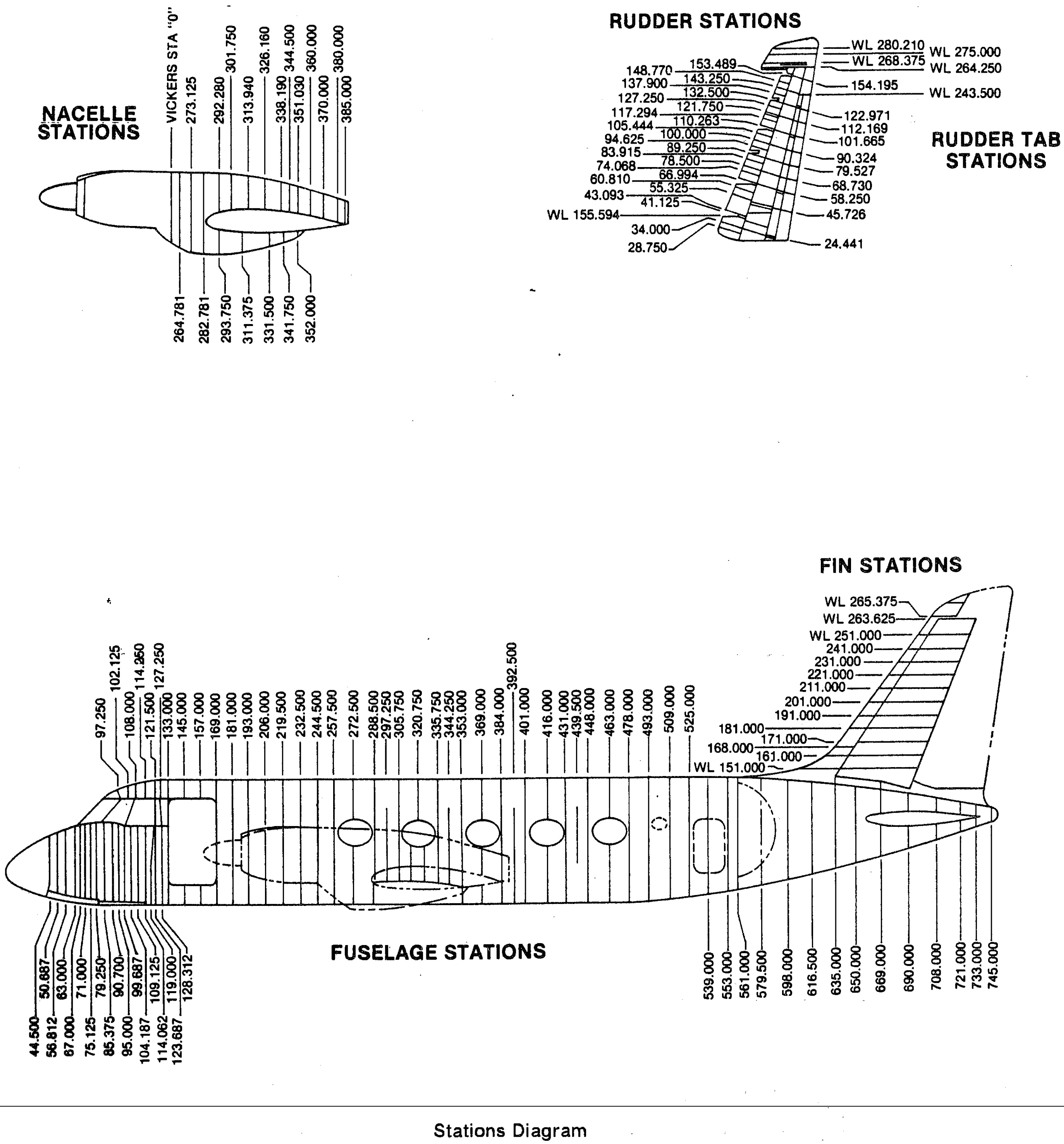
General dimensions and stations are shown in Figure 2 and Figure 3.

2.	**General Dimensions**	
A.	Span	78 ft 4 in.
B.	Length, Maximum	63 ft 9 in.
C.	Height of Vertical Fin tip to ground	23 ft 4 in.
D.	Propeller ground clearance	18 1/2 in.
E.	Design maximum takeoff gross weight	33,500 lbs. - Aircraft 1-60 not having ASC 69. 35,100 lbs. - Aircraft 1-60 having ASC 69 and Aircraft 61-162 not having ASC 175. 36,000 lbs. - Aircraft 1-60 having ASC 69 and 175, Aircraft 61-162 having ASC 175 and Aircraft 163-200, 322 and 323.
F.	Design maximum landing weight	33,000 lbs. - Aircraft 1-60 not having ASC 69. 33,650 lbs. - Aircraft 1-60 having ASC 69 and Aircraft 61-162 not having ASC 175. 34,285 lbs. - Aircraft 1-60 having ASC 69 and 175, Aircraft 61-162 having ASC 175 and Aircraft 163-200, 322 and 323.
3.	**Wing**	
A.	Type	Low wing
B.	Root airfoil section	63A214
C.	Tip airfoil section	63A314
D.	Root chord	13.8 in.
E.	Tip chord	53 in.
F.	Mean aerodynamic chord	99.35 in.
G.	Dihedral	6° 30′
H.	Aspect ratio	10
I.	Wing area, projected, exposed including area through fuselage	609.7 sq. ft
J.	Aileron area, projected, exposed	36.7 sq. ft
K.	Wing flap area projected total	110.8 sq. ft
4.	**Fuselage**	
A.	Maximum breadth, outside	94 in.
B.	Maximum breadth, inside	88 in.
5.	**Landing Gear**	
A.	Tread	24 ft 2 in.
B.	Wheel Base	19 ft 10 in. (static)
C.	Main gear tire size (tubeless)	7.50-14 Type III
D.	Nose gear tire size (tubeless)	6.50-8 Type III

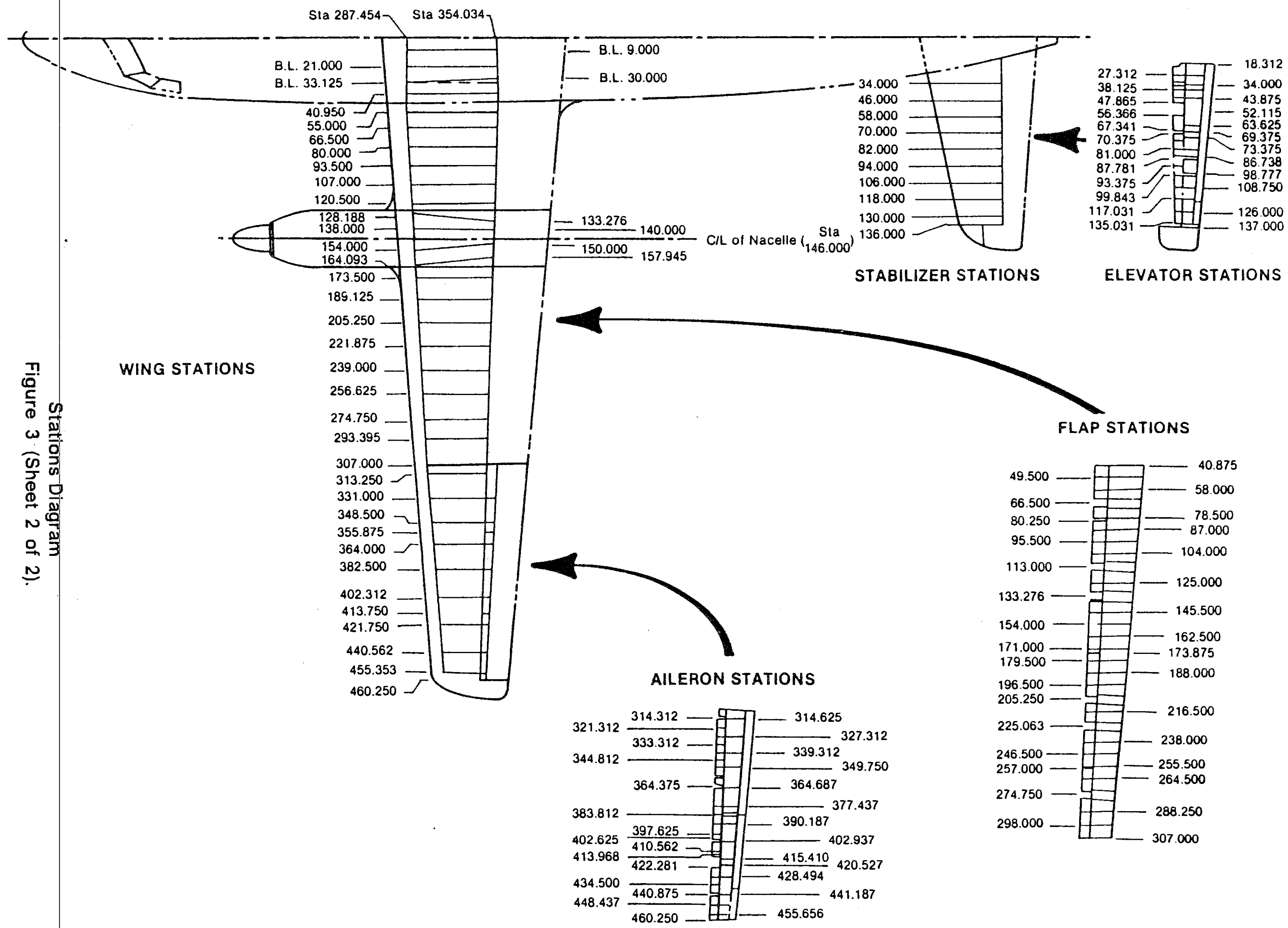
33,000 no tiene ASC 69
33,650 si tiene ASC 69



General Dimensions
Figure 2.



Stations Diagram
Figure 3 (Sheet 1 of 2).



Stations Diagram
Figure 3 (Sheet 2 of 2).

LIST OF EFFECTIVE PAGES

CHAPTER 7

LIFTING AND SHORING

* Pages revised by Revision 48
** Pages added by Revision 48
*** Pages deleted by Revision 48

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CHAPTER 7

LIFTING AND SHORING

LIFTING AND SHORING — DESCRIPTION / OPERATION

1. General

Jacking points are provided on the landing gear, fuselage and wing.

No provisions have been made for hoisting the entire aircraft. For hoisting of individual aircraft assemblies, refer to the appropriate chapter of this Maintenance Manual.

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AIRCRAFT JACKING — DESCRIPTION / OPERATION

1. General

See Figure 201

The jacking procedure for the Gulfstream is the same as that for any other aircraft with a tricycle landing gear. The aircraft is raised off the ground by jacks located at three points. The nose or fuselage jack point is located at Fuselage Station 120.10. This is just aft of the nosewheel fairing. Two holes have been provided in the aircraft structure at this point for a jack pad. This pad is Gulfstream Aerospace designed tool part number 159GT1002. The main or wing jack points are located on each wing at station 158.3. These jack pads are structural parts of the aircraft and are accessible through access covers located at the aft, outboard end of each nacelle. These access covers are triangular in shape and are held in position by nine screws. A jack having a capacity of 11,000 pounds and capable of extending from 40 to 60 inches is required to lift the fuselage (Regent part number 2921 or Malabar part number 515). Two jacks of 24,000 pounds capacity, capable of extending from 50 to 75 inches are required for the wings (Regent part number 9725). When all three jacks are in place, the aircraft may be lifted evenly and cautiously off the ground. A jack point has been provided for jacking the main dual wheels. This jack point is located directly between the two wheels. A jack of at least 16,500 pounds capacity, capable of extending from 6 to 14 3/4 inches, is required at this point (Regent part number 3999). If both nosewheel tires are flat, the height between the ground and the jack point is only four inches, therefore a jack with at least a 5000 pound capacity, capable of retracting to a height of 4 inches, is required. A crocodile jack, 10 ton capacity, with an extension from 2 1/2 inches to 23 1/2 inches can be used (Regent part number 1988S). If a jack of this type is not available, the nose must be raised by means of the forward fuselage jack.

2. Jacking Weight Limitations

When jacking the main landing gear, the aircraft weight must not exceed 36,000 pounds.

When jacking at three points (each wing and the nose) to raise the entire aircraft, a maximum weight of 36,000 pounds must not be exceeded.

3. Tail Clearances for Tilted Hangaring

The vertical fin reaches a height of 23 feet 1 1/2 inches from the ground when the aircraft is in the static ground attitude (3/4° nose up, all struts with three inch stroke). The skin surface of the aircraft at the fuselage jack pad, station 120.1 is 49 1/2 inches from the ground in the same condition. For each one inch the nose jack pad is raised from the static position, the vertical fin will be lowered approximately two inches. Before using this relationship to establish a nose raising distance, allow for any overall increase in vertical fin height caused by the use of dollies under the main gear.

A. Example of Tilted Hangaring Calculation

(1) Hangar door beam height above level ground: 20 feet

(2) Gulfstream tail height, static attitude: 23 feet 4 inches.

(3) Main gear hangaring dolly platform height: 3 1/2 inches

(4) Net tail height: 23 feet 4 inches (tail height)
+ 3 1/2inches (dolly platform)
23 feet 7 1/2 inches

(5) Tail lowering required: 23 feet 7 1/2 inches (net tail height)
− 20 feet 0 inches (hangar door beam height)
3 feet 7 1/2 inches (=43 1/2 inches)

(6) Safety clearance: 6 inches

(7) Total tail lowering desired: 43 1/2 inches (actual lowering required
+ 6 inches (Safety clearance)
49 1/2 inches

(8) Jack pad to ground: 49.6 inches (static)

(9) Jack pad raising distance required to produce 47 inches of tail movement:

$$\frac{49.5 \text{ inches}}{2} \quad + 24.75 \text{ inches}$$

(10) Dolly factor applied to jack pad to ground distance = 3.5 inches

(11) Jack pad to ground distance at end of tilt to clear 20 feet hangar door by 6 inches:

49.6 inches (jack pad to ground)
24.75 inches (jack pad raising distance)
+ 3.5 inches (dolly factor)
77.85 inches

(12) This amount of tilt gives the aircraft approximately a 7° nose up attitude. (3/4° static + 6 1/4° (tilt)

CAUTION: AS A GENERAL RULE, THE LESS FUEL THE BETTER WHEN HANDLING A TILTED AIRCRAFT FROM BOTH THE WEIGHT ON THE GEAR AND SLOSHING FUEL WEIGHT ON THE CG STANDPOINTS. HOWEVER, LESS THAN 400 POUNDS OF FUEL PER WING COULD RESULT IN EXPOSURE OF THE FORWARD FUEL BOOST PUMPS AT HIGH TILT ANGLES, THEREBY ALLOWING AIR TO ENTER THE SYSTEM. THIS CONDITION COULD CAUSE HOT STARTS AND/OR A BLEEDING PROCEDURE.

MAXIMUM ALLOWABLE STATIC LOAD ON THE NOSE GEAR IS 3480 POUNDS.

MAXIMUM ALLOWABLE LOAD ON THE NOSE GEAR TO REMAIN WITHIN CG LIMITS IS 3380 POUNDS.

TO AVOID UPSETTING THE AIRCRAFT, ESPECIALLY WHEN PARTIAL FUEL LOADS ARE ABOARD, THE CENTER OF GRAVITY (H-ARM) OF THE STATIC AIRCRAFT SHOULD NOT BE AFT OF FUSELAGE STATION 307.0.

THE PRACTICAL LIMIT FOR TILTING THE AIRCRAFT SHOULD BE HELD TO NINE DEGREES NOSE UP. THIS IS EQUIVALENT TO RAISING THE NOSE JACK PAD TO 78 INCHES ABOVE THE GROUND, STARTING FROM STATIC POSITION WITH MAIN WHEELS ON THE GROUND, OR RAISING THE JACK PAD 28.4 INCHES.

AIRCRAFT JACKING — MAINTENANCE PRACTICES

1. Jacking Aircraft

(See Figure 201)

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

WALKING ON EMPENNAGE OR OUTER WING WHILE AIRCRAFT IS ON JACKS MAY CAUSE MOVEMENT OF AIRCRAFT ON JACKS. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

CAUTION: DO NOT JACK AIRCRAFT IF MAXIMUM WIND GUST EXCEED 20 MILES PER HOUR.

THE LOWER SECTION OF AIRSTAIR DOOR IS NORMALLY SUPPORTED ON GROUND. IF STAIRWAY IS USED FOR ANY REASON WHILE AIRCRAFT IS ON JACKS, THIS STAIRWAY MUST BE SUPPORTED.

A. General Jacking Practices

(1) Operate all jacks evenly so aircraft remains as nearly level as possible.

(2) Set locking devices on jack stands to prevent accidental lowering of aircraft.

(3) Head aircraft into wind if it is to be jacked outdoors. Do not jack if wind velocity exceeds 10 miles per hour unless aircraft is snubbed at all mooring points.

(4) When jacking from a single point, keep lift to an absolute minimum and always within safe limits of the jack.

(5) The following jacks are to be used for jacking the aircraft:

• Main landing gear jacks	8.5 tons
• Nose landing gear jacks	3.0 tons
• Wing jacks	8.0 tons
• Fuselage nose jacks	3.5 tons

B. Jacking from Wing/Nose Jack Points

(1) Remove access plate located at rear outboard underside of each engine nacelle at Wing Station 158.3.

(2) Place an eight ton capacity wing jack in position and run up to contact wing jack pads on each wing.

(3) Attach fuselage nose jack pad, part number 159GT1002, to two pickup holes in forward fuselage section at Fuselage Station 120.10.

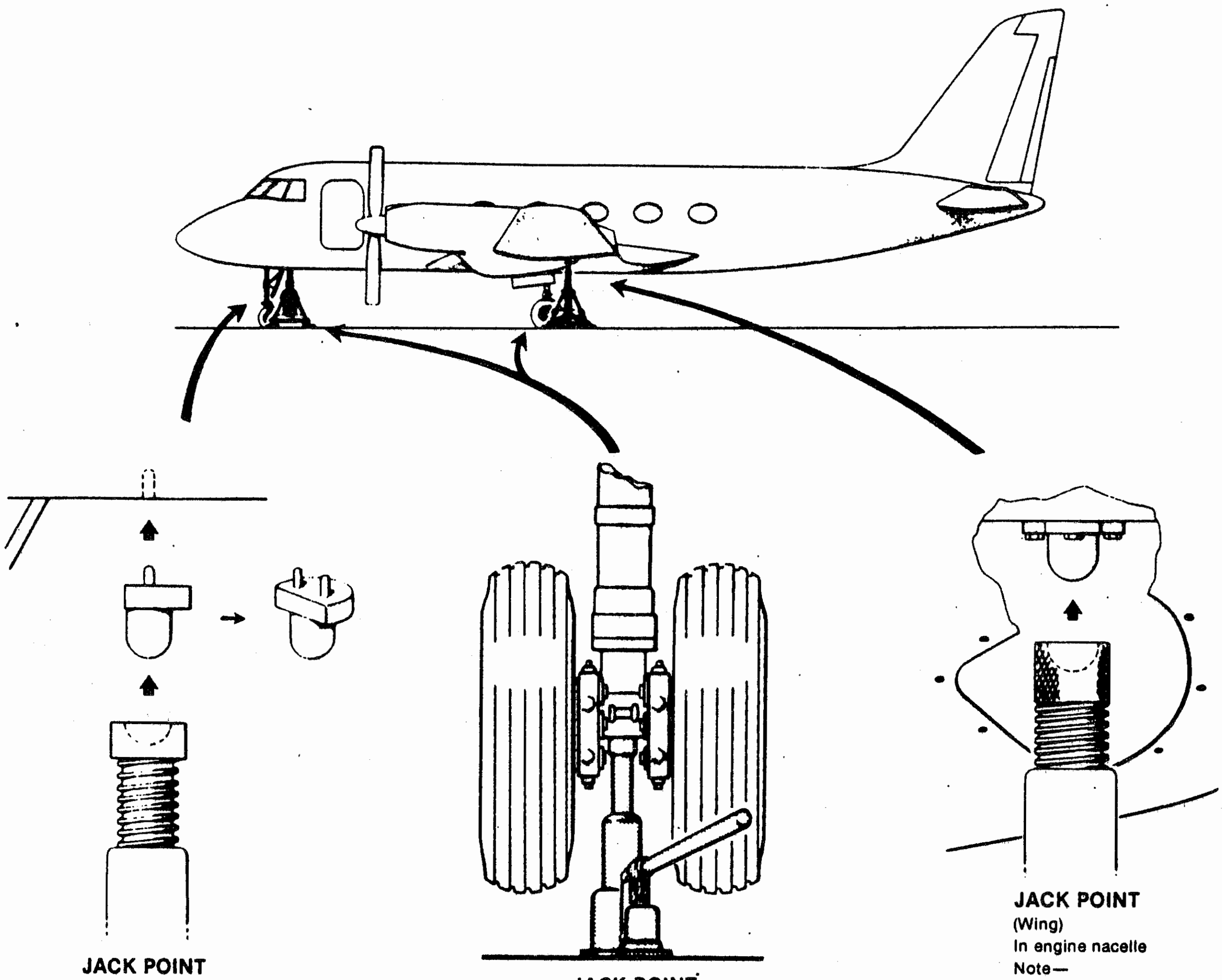
(4) Place a 3.5 ton capacity jack in position and run up to contact nose jack pad.

C. Jacking with Engines Removed

(1) To prevent a critical rearward center of gravity shift with engines removed, approximately 1000 pounds of ballast should be added to forward baggage compartment and entrance area when jacking from wing and nose jack points.

(2) When aircraft is on its wheels or wheel jacks, 1500 pounds of ballast must be added to forward baggage compartment and entrance area. When removing only one engine the ballast required may be reduced to 750 pounds with the aircraft on its wheels.

NOTE: The maximum ballast weight required to maintain correct center of gravity for all conditions of engine and other equipment removal is 1500 pounds. No ballast is required for propeller removal alone.



JACK POINT	CLOSED HEIGHT (Fully compressed shock strut and flat tires)	LIFT REQUIRED (For extended shock strut and two inch tire clearance)	LIFT REQUIRED (For one inch tire clearance)	WEIGHT (Pounds maximum)
MAIN WHEELS	6.0 in.		8.0 in.	16,756
NOSE WHEEL	4.0 in.		6.0 in.	5,000
FUSELAGE JACK PAD	43.1 in.	16.0 in.		6,608
WING JACK PADS	54.8 in.	20.0 in.		15,210

Jacking Provisions
Figure 201.

NOSE GEAR JACKING — DESCRIPTION / OPERATION

1. **General**

   (See Section 7-1-0 Figure 201)

   A jack cone on the bottom of the nose gear shock struts provides a jacking pad for nose gear wheel removal. A 3 ton capacity jack is required.

   If both nose wheel tires are deflated, it may be necessary to jack the nose of the aircraft.

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MAIN GEAR JACKING — DESCRIPTION / OPERATION

1. **General**

(See Section 7-1-0 Figure 201)

CAUTION: CARE SHOULD BE TAKEN WHEN APPLYING THE JACK TO THE MAIN GEAR, SINCE THE JACK POINT IS LOCATED BETWEEN TWO BRAKE CARRIERS AND SHUTTLE VALVES WHICH MAY BE EASILY DAMAGED.

A jack cone for use as a jack pad is provided for main gear wheel removal, on the bottom of each main gear strut. Use an 8.5 ton capacity landing gear jack.

When both tires on one main landing gear are deflated, wing and nose jacks must be used.

NOTE: A special jack (Regent part number 3999) has been designed for this condition and may be obtained from the Regent Jack Mfg. Co., Inc., if desired.

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HOISTING AND LIKING — DESCRIPTION / OPERATION

1. General

It is essential that personnel engage in raising the Gulfstream be familiar with the aircraft structure, its limitations, and the use of raising equipment. The aircraft can be raised by means of hoists or lifting bags as shown in Figure 1 and Figure 1

NOTE: A survey to determine the structural condition of the aircraft should be made before raising is attempted to ensure that the raising and stabilizing points to be used can take and transfer the loads to be imposed on them.

Lifting bags cannot be placed at areas of extensive damage, however, local damage can be bridged to sound structure.

Owing to the unpredictability of aircraft attitude, the degree of damage, the terrain and its surface conditions, the success of any lifting operation will depend on the ingenuity and care of the personnel in the field. The important thing to watch is the attitude of the bags. Since each bag is supporting a large weight, any inclination from the vertical could produce lateral forces far in excess of those allowable in stabilizing lines. For this reason, the aircraft should be kept in equilibrium by inflating the appropriate bags rather than relying on the stabilizing lines. One of the advantages of using more than the minimum number of bags is that they can be shifted individually during the raising process to improve their positions.

It is advisable, although not necessary, to defuel the aircraft, remove the cargo, and purge the tanks, to lessen the possibility of additional damage and reduce the fire hazard during the raising procedure. Cargo that is not removed should be tied down to prevent shifting. Control surfaces should be locked to prevent their banging against the stops. Landing gear that are extended should have ground locks installed, wheels chocked and brakes released.

CAUTION: TO MAINTAIN STABILITY, AIRCRAFT SHOULD ONLY BE RAISED WHEN THERE IS LITTLE OR NO WIND.

When aircraft is being raised with bags, stabilizing lines must be used to maintain stability in a lateral plane. The aircraft stabilizing tie-down points are shown in Figure 201.

2. Pneumatic Lifting Bags

Although a fully loaded Gulfstream I (Gross Weight 36,000 lbs.) with all landing gear up can be raised with one bag under each wing and one bag under fuselage (located either fore or aft as required to level aircraft), it is preferable to use two bags under each wing to reduce the possibility of damage to leading and trailing edges. It is also preferable to use two bags under fuselage, one forward and one aft, to prevent tipping due to a shift of the center of gravity relative to wing bags. Where only one bag is used under each wing, they must be placed symmetrically, that is, either both inboard or both outboard of engine nacelles.

The physical characteristics of the lifting bags as follows:

- Lifting Bags (Type F-2 MIL-B-6640A)
- Length: 7 1/2 feet, width: 6 1/2 feet
- Depth minimum: 6 inches
- Depth maximum: 6 feet
- Lifting capacity: 12 tons
- Pressure: 2 1/2 psi

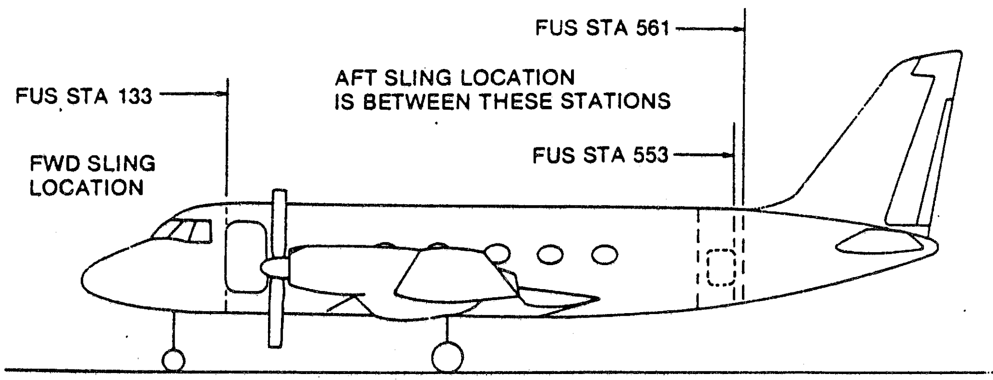
Before bags are set in place, the ground area should be cleared of all sharp or projecting objects. The structure to be supported, and that adjacent to it, should also have all projections and sharp edges removed or covered. Any oil or grease present should be removed. The bags, which come covered in a tarpaulin, should be laid on the ground on top of the tarpaulin. Check air outlets to ensure that they are secure before inflating. Bags should be inflated so as to keep the aircraft level and the lifting force vertical.

3. Stabilizing Lines

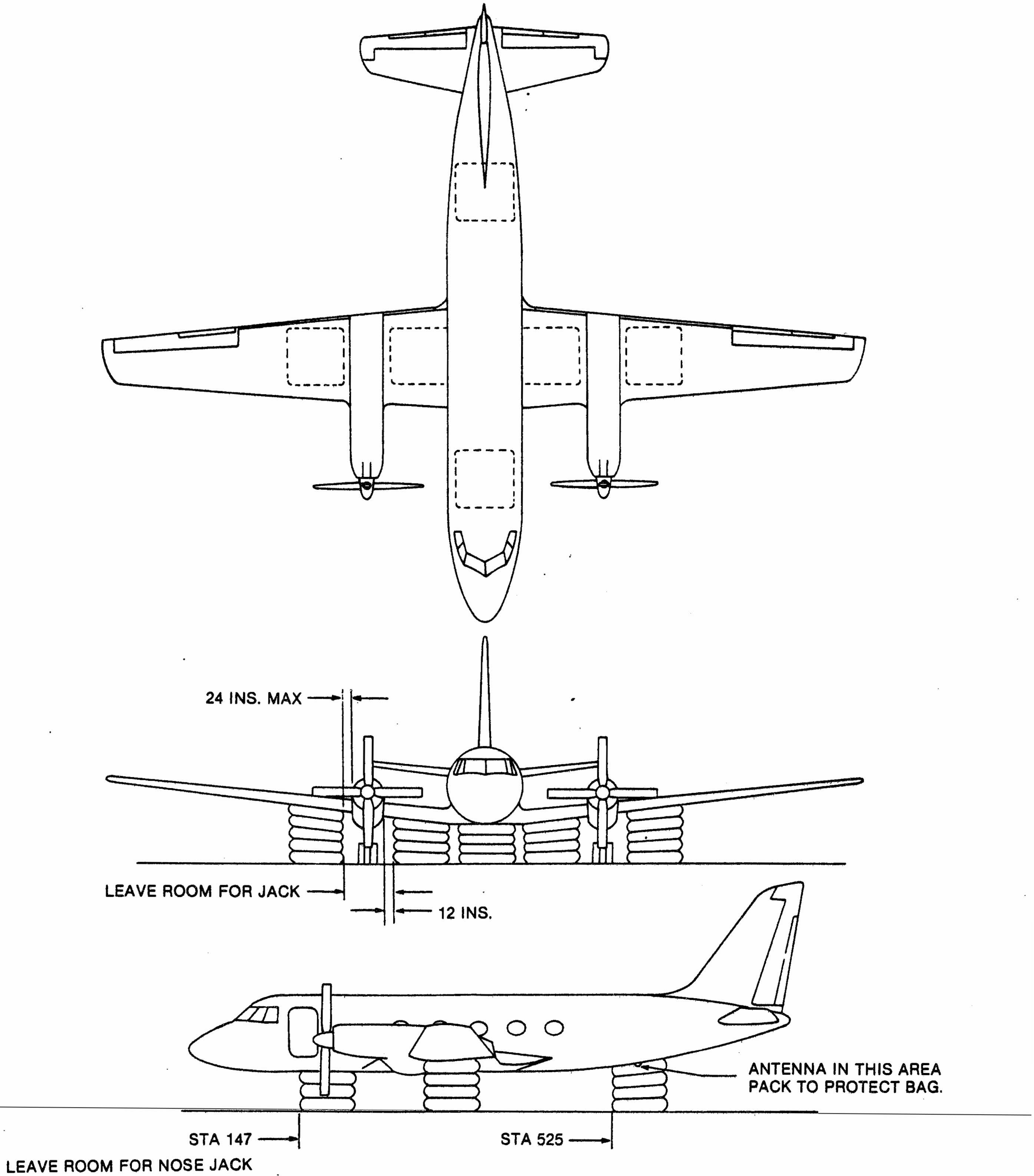
When the aircraft is being raised with bags, stabilizing lines must be used to maintain stability in a lateral plane. Loads in excess of maximum line load capabilities can cause permanent damage to structure in the vicinity of the attachment points. All loads are based on a factor of three to ultimate failure and a factor of two to yield conditions.

Maximum line loads are as follows:

- Tail line: 610 pounds each
- Wing line: 1000 pounds each



Sling Locations
Figure 1.



Locations — Lifting Bags
Figure 2.

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HOISTING AND LIKING — MAINTENANCE PRACTICES

1. **Tail Lines Attachment — Procedure**

To attach tail lines, proceed as follows:

A. Remove tail cone.

B. Loop rope around each inboard end of elevator torque tube.

C. Cross each rope so as to pull from the opposite side.

NOTE: Aircraft that have riveted tail cones could be stabilized by attaching ropes around fuselage forward of tail surfaces as shown in Figure 201

2. **Wing Lines Attachment — Procedure**

CAUTION: AIRCRAFT PROPELLER AND SHAFT MUST NOT BE USED.

To attach wing lines, process as follows:

A. Remove eight screws in lower water methanol pump covers on outer wing panel.

B. Attach 159GT1069-T1 and 159GT1069-T2 fittings to cover. (eights screws each).

C. Loop two ropes through each fitting.

D. Run one line forward.

E. Run second line aft.

NOTE: Manila rope sizes required are as follows:

For tail stabilizing lines use 7/16 minimum diameter, breaking strength 1750 lbs. For wing stabilizing lines use 9/16 minimum diameter breaking 3450 lbs. Where it is impossible to use stabilizing locations shown, other points will have to be chosen.

3. **Emergency Condition — Procedures**

The following paragraphs list several emergency conditions which could be encountered and how each should be handled.

A. Belly Landing. All Gear Retracted (See Figure 202

(1) Position bags under wings outboard of nacelles and under aft fuselage.

(2) Attach stabilizing lines, two on wing and two on tail.

(3) Inflate bags evenly until additional bags may be placed under wing inboard of nacelles.

NOTE: Use of bags under wings, outboard of nacelles only, will not raise aircraft sufficiently to lower main gear. In this case, wing jacks would have to be used to gain extra height required.

(4) Inflate all bags evenly until bags can be places under forward fuselage.

(5) Inflate all bags evenly until gear can be lowered or jacks installed.

B. Nose Gear Retracted Main Gear Extended

(1) Nose can be raised by hoisting up on nose or pulling down on aft fuselage at sling stations shown sufficiently to enable nose gear to be lowered, as shown in Figure 203

CAUTION: IF NOSE IS RAISED TOO FAR, AIRCRAFT CG MAY MOVE BEHIND MAIN GEAR AND CAUSE AIRCRAFT TO FALL ON ITS TAIL. IT IS ADVISABLE TO HAVE A BAG OR SUPPORT OF SOME KIND UNDER AFT FUSELAGE TO PREVENT AIRCRAFT FROM TIPPING. IT IS ALSO ADVISABLE AFTER RAISING NOSE TO SUPPORT IT PRIOR TO LOWERING NOSE WHEEL, IN THE EVENT HOIST SHOULD FAIL.

NOTE: Stabilizing lines are unnecessary when hoists are used.

(2) Nose can be raised sufficiently with bags as shown in Figure 204, to enable nose gear to be lowered. Place bags under rear fuselage and inflate. Bags under rear fuselage will prevent aircraft from falling on its tail. Position bags under forward fuselage and inflate. Raise nose of aircraft until nose wheel can be lowered and locked.

NOTE: Stabilizing lines are not normally required if wheels are firmly chocked.

C. One Main Gear Retracted, One Main Gear and Nose Gear Extended.

(1) Place bags under lower wing, inboard and outboard of nacelle as shown in Figure 205.

(2) Attach tail stabilizing lines and chock main gear.

NOTE: A stabilizing line may be tied to nose gear and run off laterally.

(3) Inflate bags under wing until fuselage is raised sufficiently to place bags at rear fuselage position.

(4) Inflate all bags to level of aircraft, and raise until main gear can be lowered or jacks installed.

D. Both Main Gear Retracted, Nose Gear Extended

CAUTION: FLAPS SHOULD BE IN THE ZERO (0) DEGREE POSITION IF POSSIBLE, TO AVOID DAMAGE.

NOTE: Aircraft weight distribution is roughly equal between extended gear and aft fuselage. Nose gear strut will probably be bottomed out.

(1) Place bags inboard and outboard of each engine nacelle, even with wing trailing edge, flap, and overhang skin.

(2) Place panel of quarter-inch plywood or equivalent between bags and the wing skin to avoid damage to trailing edge overhang skin and flaps, and to prevent damage to bags.

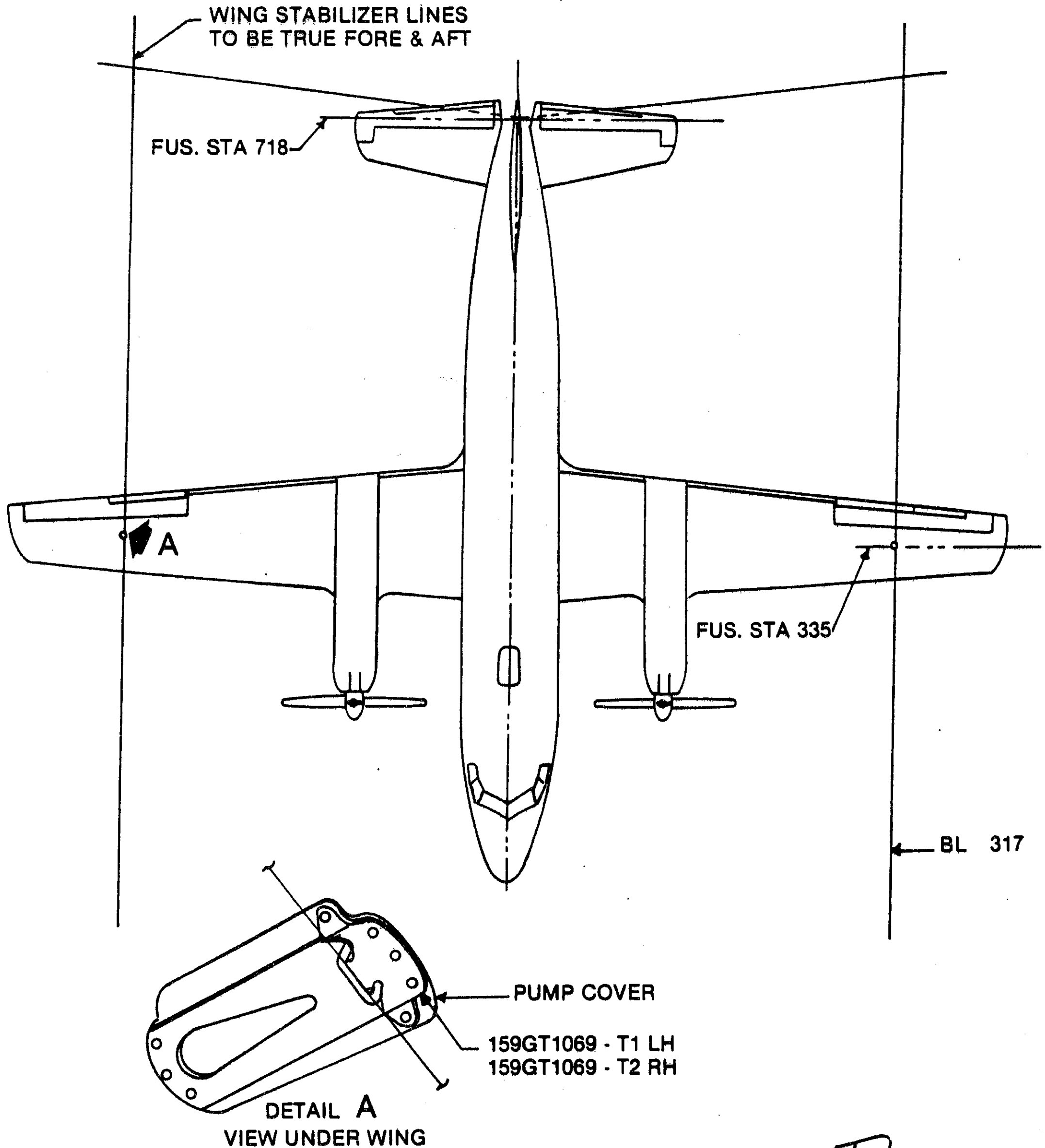
(3) Place one bag underneath each horizontal tail and release gust locks to avoid damage to elevators. Do not chock nose wheel.

(4) Provide guy lines for stabilization as shown in Figure 201, Paragraphs 1. and 2. above.

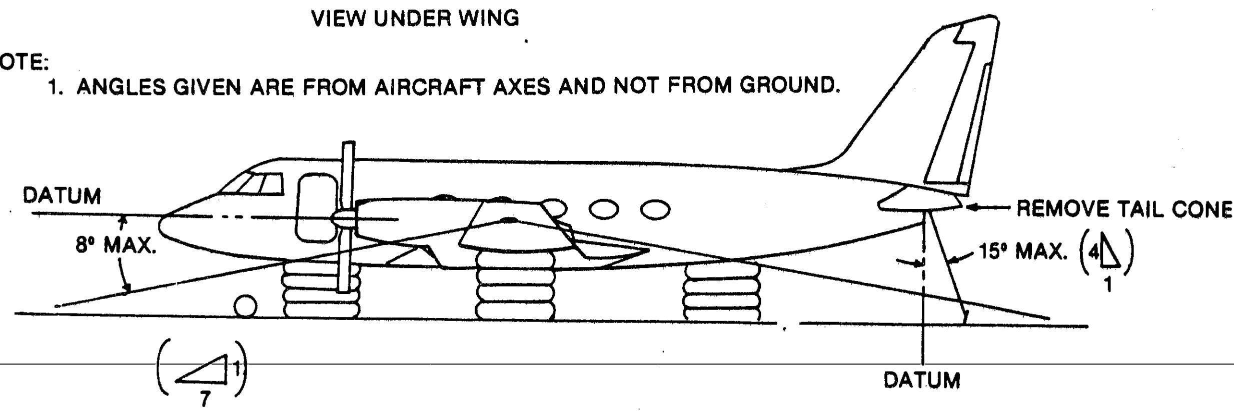
CAUTION: DURING INFLATION OF BAGS, KEEP BAGS BENEATH THE TAIL SOFT TO AVOID DAMAGE TO TAIL OR AFT FUSELAGE.

(5) Inflate bags beneath wings evenly to maintain the aircraft wings level. Inflate bags beneath horizontal tail as required for stabilization but keep them soft. These bags are only to be used for stabilizing during initial lifting process and will become useless as aircraft tail rises above the bags' maximum height.

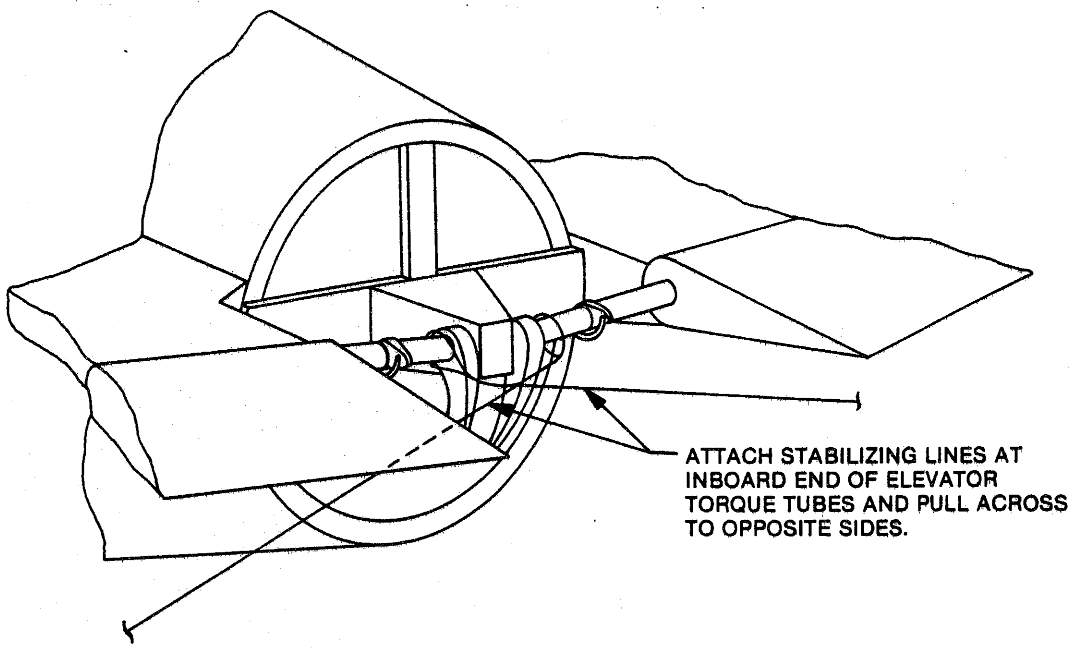
(6) Continue inflation until main gear can be lowered or aircraft transferred to jacks. Inflate nose strut before moving aircraft.

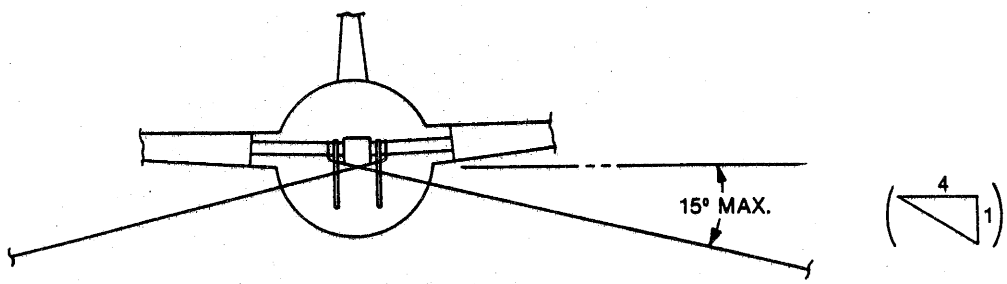


NOTE:
1. ANGLES GIVEN ARE FROM AIRCRAFT AXES AND NOT FROM GROUND.

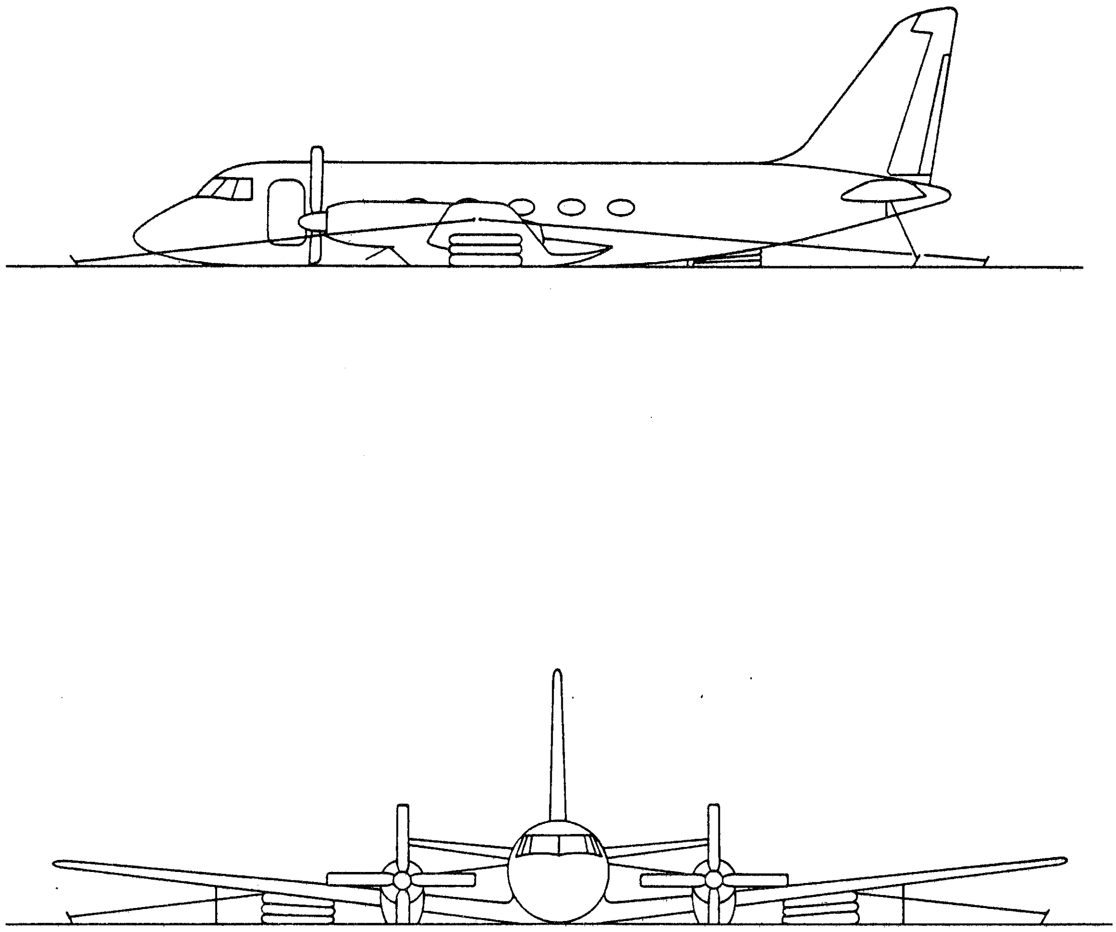


Stabilizing Line Attachment Locations Diagram (Angles Given Are From Aircraft Axes)
Figure 201 (Sheet 1 of 2).

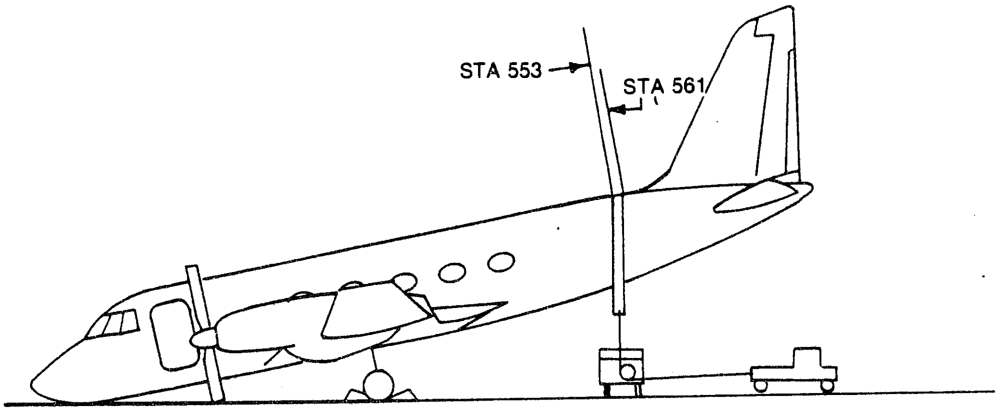




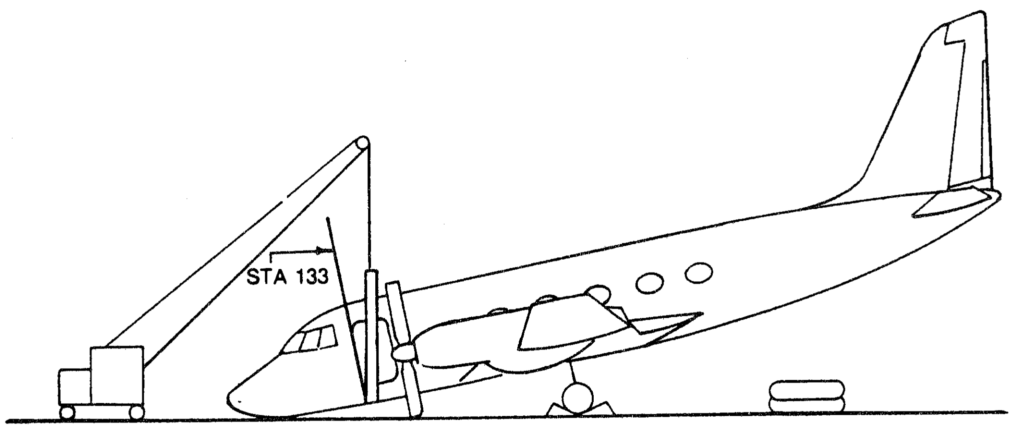
Stabilizing Line Attachment Locations Diagram (Angles Given Are From Aircraft Axes)
Figure 201 (Sheet 2 of 2).



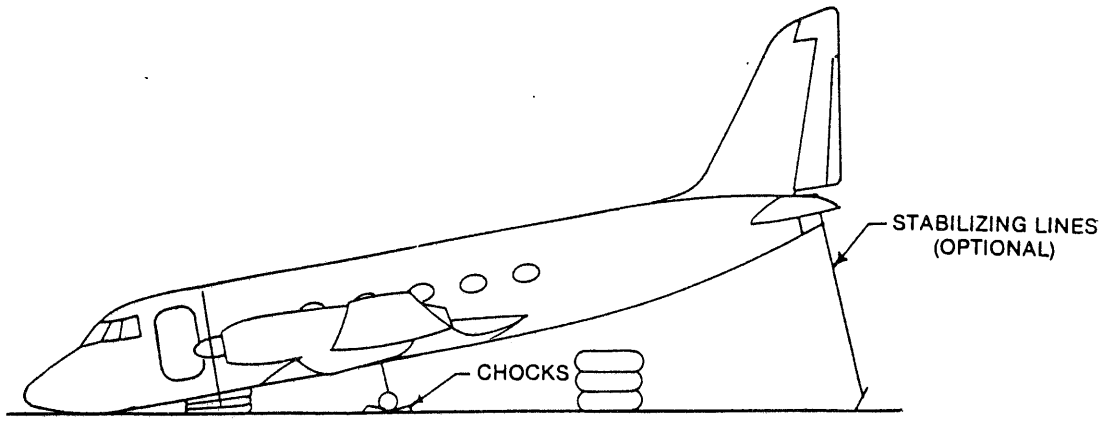
Belly Landing
Figure 202.



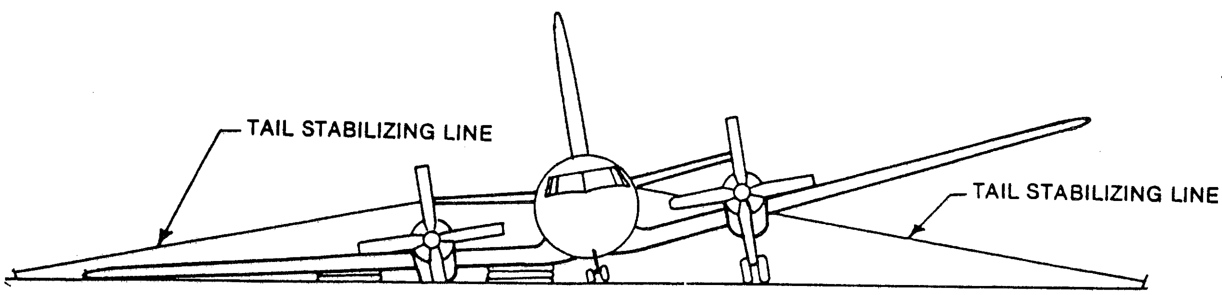
CAUTION:
SLINGS MUST BE RESTRAINED AGAINST SLIPPING FROM STATIONS SHOWN.



Nose Gear Up Landing — Hoisting Methods
Figure 203.



Nose Gear Up Landing — Raising With Bags
Figure 204.



One Main Gear Collapsed — Lifting With Bags
Figure 205.

LIST OF EFFECTIVE PAGES

CHAPTER 8

LEVELING AND WEIGHTING

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*8	i	September 11
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*8	2	September 11/92
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*8-1-0	2	September 11/92
*8-2-0	1	September 11/92
*8-2-0	2	September 11/92
*8-2-0	201	September 11/92
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CHAPTER 8

LEVELING AND WEIGHTING

LEVELING — DESCRIPTION / OPERATION

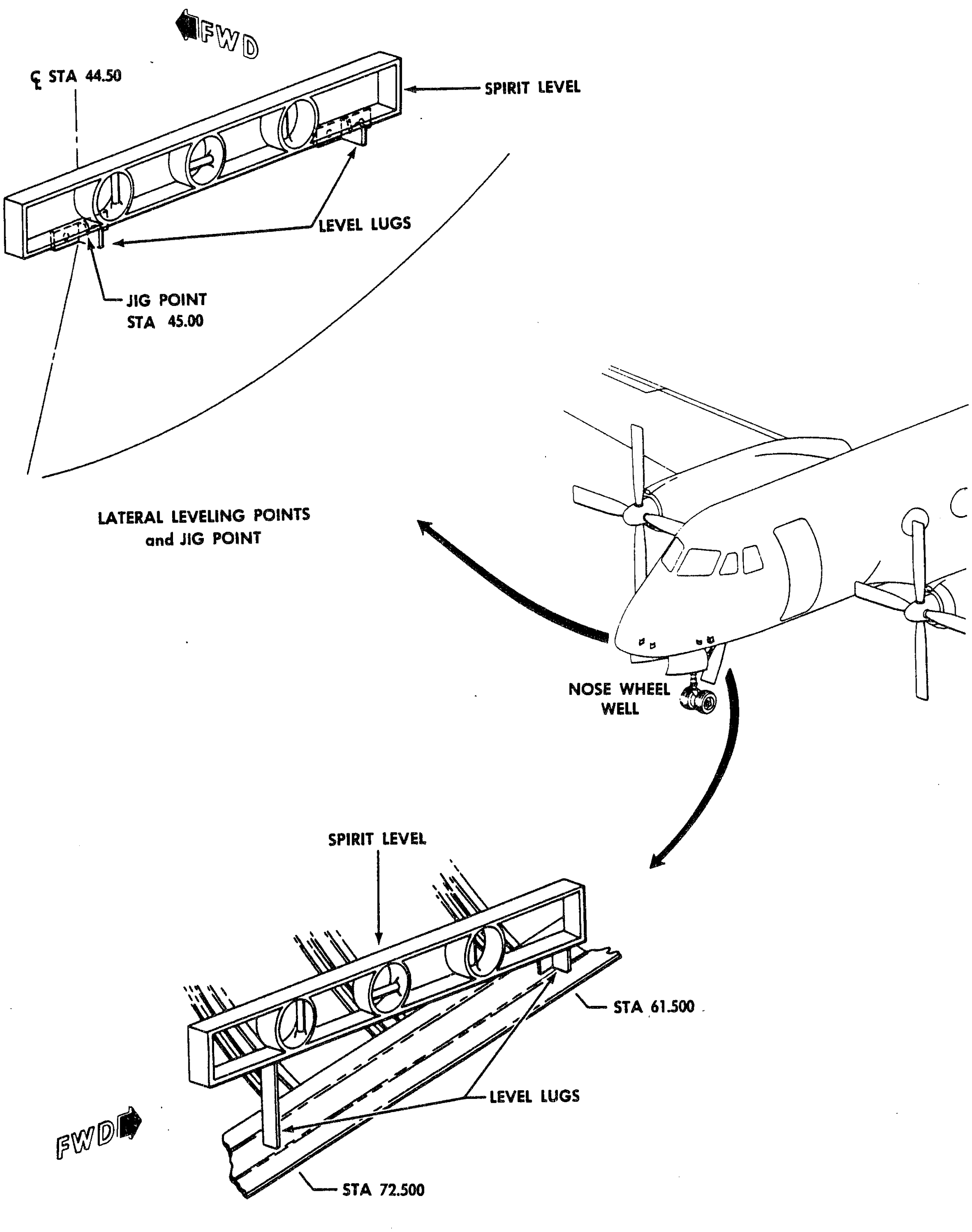
1. General

(See Figure 1)

Determination of level position is accomplished by the use of a level and leveling brackets located in the nose wheel well. The longitudinal leveling brackets are located along the left nose wheel door longeron (Fuselage Station 72.500 and 61.500). The lateral leveling brackets are located on the rear face of the bulkhead station 44.5. The lateral bracket at the centerline of the aircraft is also the jig point (Fuselage Station 45.00).

With aircraft on wheels, the aircraft can be leveled by inflating or deflating the shock struts.

With aircraft on jacks, the aircraft can be leveled by raising or lowering of the jacks as required.



Leveling Aircraft
Figure 1.

WEIGHTING — DESCRIPTION / OPERATION

1. **General**

Weighing of the aircraft is accomplished by two methods:

- Weighing on wheels
- Weighing on fuselage and wing jacks.

In both cases, the aircraft must be level.

2. **Aircraft Condition and Definitions**

A. Empty Weight Condition.

The empty weight condition is established with wheels down and wing flaps retracted and includes the following fluids:

- Full hydraulic fluid.
- Full oil in auxiliary power unit.
- Full oil in accessory gearbox.
- Unusable fuel (includes undrainable fuel).
- Unusable water-methanol.
- Full engine oil.

B. Fuel and Water-Methanol.

Unusable fuel is the quantity of fuel that is unavailable to the fuel pumps which is 72 pounds per aircraft (6.75 pounds per gallon @ 59°F). This fuel level corresponds to the zero point on the fuel gage. Unusable fuel includes 16 pounds of undrainable fuel. The aircraft should be weighed with undrainable fuel remaining. This is done by first defueling through the nacelle fuel drain valve per normal defueling practice and then draining through the water drain valves in the inboard bottom wing skin. The difference between unusable fuel and undrainable fuel (i.e., 72 pounds − 16 pounds = 56 pounds) should be added to the **ACTUAL WEIGHT** to determine **EMPTY WEIGHT**.

NOTE: Due to large volume of the fuel tanks, it is not recommended that aircraft be weighed with full fuel.

Unusable water-methanol is the quantity remaining after the system has been drained through the nacelle water-methanol drain valve by running the boost pumps to cavitation. This quantity corresponds to the zero point on the water methanol gage.

It may be desirable to weigh the aircraft with full water-methanol tanks, because it is easier to fill the tanks than to drain them. In this case, the aircraft is weighed with full tanks, and the weight of the usable water-methanol is deducted from the **ACTUAL WEIGHT** to determine **EMPTY WEIGHT**. (Weight variation caused by temperature is negligible.)

For aircraft with the water-methanol tanks located in the wing outer panel, total usable water-methanol equals 660 pounds (cg 321.0), and unusable water-methanol equals 27 pounds (cg 325.0).

For aircraft with water-methanol tanks located in the wing fillet, total usable water-methanol equal 372 pounds (cg 384.1) and unusable water-methanol equals 14 pounds (cg 345.5).

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WEIGHTING — MAINTENANCE PRACTICES

1. **Weighing Aircraft**

(See Figure 201)

A. Preparation for Weighing.

(1) Conduct a complete inventory check of the aircraft to determine that all required equipment for the empty weight condition is installed or properly accounted for.

NOTE: It is recommended that a checklist be used as an aid to consistent and thorough check-out. The equipment list section of the weight report furnished with the aircraft may be used for this purpose if it is supplemented by the operator to reflect changes made to the aircraft after delivery.

(2) The following fluid tanks and systems must be filled to normal operating level:

- Engine oil
- Hydraulic fluid
- Oil in auxiliary power unit
- Oil in accessory gearboxes

(3) The following fluids and/or items should be drained or removed: Fuel (defueled, then drained to undrainable level with aircraft level)

- Wash water
- Toilet
- Galley and or bar items not on equipment list (liquids, food, etc).

(4) The water-methanol system should be drained, with the aircraft level, to the unusable quantity.

NOTE: If draining of water-methanol is not desired, the tanks may be filled, and the weight of the usable water-methanol deducted from the **ACTUAL WEIGHT** to determine **EMPTY WEIGHT**.

(5) Remove work equipment, protective material, etc. from interior.

(6) Exterior must be dry and clean, and protective covers, etc. removed.

(7) Provide scales of the following capacities:

- 8.5 tons at each main gear
- 8 tons at each wing jack
- 3 tons at the nose gear
- 3.5 tons at fuselage nose jack

(8) Close entrance door and stairway.

B. Weighing Procedures.

(1) Level aircraft as follows:

(a) Determination of level position is accomplished by use of a level and the leveling brackets in the nose wheel well.

(b) For weighing on wheels and if the scales cannot be vertically adjusted, the aircraft can be leveled longitudinally by inflating the main struts, or deflating the nose strut, or both.

(c) For weighing on fuselage and wing jacks, the aircraft must be approximately leveled by adjusting landing gear struts prior to jacking and all jacks should be raised at approximately the same rate. This prevents side load components which could cause scale errors from being transmitted to the scales.

(2) Determine the following measurements:

(a) For weighing on wheels, measure dimensions 'a' and 'b' with the aircraft level. This is done by projecting main wheel centers and jig point centers to the floor and measuring the average longitudal distances. These distances are slightly variable as a function of strut

extension. A typical dimension for 'a' is 283.12 inch and a typical dimension for 'b' is 237.62 inch.

(b) For weighing on fuselage nose and wing jacks, no measurements are necessary. See Figure 201 for fixed dimensions.

C. Example of a Typical Weighing

(1) Condition of aircraft

(a) Aircraft inventoried with equipment list and is complete.

(b) Wash water and toilets drained. Miscellaneous galley food and liquids removed.

(c) All tools and work equipment removed, protective covers removed from engines and propellers.

(d) Fuel tanks drained to undrainable level (16 pounds of fuel remaining per aircraft).

(e) Water-methanol tanks drained to unusable level.

(f) All other fluids full to normal operating level.

(g) Entrance door and stairway closed.

(h) Weighed on wheels, with aircraft level and flaps retracted.

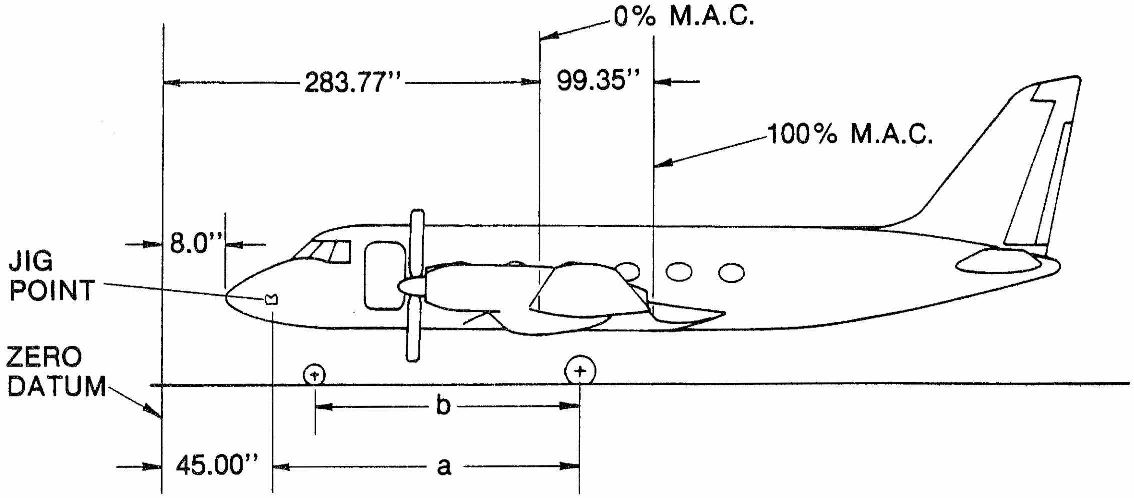
(2) Data (See Figure 201)

(a) Items to be added to **ACTUAL WEIGHT** to determine **EMPTY WEIGHT**.

	Weight	CG
Unusable Fuel	72	325.0
Undrainable Fuel	−16	325.0
Additions Total	56	325.0

(b) There are no items to be subtracted from **ACTUAL WEIGHT** to determine **EMPTY WEIGHT**.

ACTUAL WEIGHT & BALANCE DATA



DISTANCE	FT	& INS.	INS.
a	23	7-1 8	283.12
b	19	9- 5 8	237.62
a – b			45.50

MAIN GEAR HORIZONTAL ARM = a + 45.00 = 328.12 INCHES

NOSE GEAR HORIZONTAL ARM = (a – b) + 45.00 = 90.50 INCHES

C.G. IN PERCENT M.A.C. = $\frac{\text{(HOR. ARM IN INCHES)} - (283.77)}{(0.9935)}$

NOTE: ALL C.G.'s ARE FOR THE AIRPLANE WITH LANDING GEAR EXTENDED

AIRPLANE ON WHEELS

WHEEL LOCATION	GROSS	TARE	ERROR	TOT. Δ	NET	HOR. ARM	MOMENT
LEFT MAIN	9502	– 2	± 0	– 2	9500		
RIGHT MAIN	9601	– 2	± 0	– 2	9599		
SUBTOTAL MAINS	19,103	– 4	± 0	– 4	19,099	328.12	6,266,764
NOSE	2,254	– 2	± 0	– 2	2,252	90.50	203,806
TOTALS	21,357	– 6	± 0	– 6	21,351	303.06	6,470,570

SCALE LOCATION	GROSS	TARE	ERROR	TOT. Δ	NET	HOR. ARM	MOMENT
LEFT WING							
RIGHT WING							
SUBTOTAL WINGS						347.00	
NOSE						120.10	
TOTALS							

EMPTY WEIGHT & C.G. DETERMINATION

ITEMS	WEIGHT	C.G.IN HOR. ARM	C.G. IN % M.A.C.	MOMENT
TOTAL ACTUAL NET	21,351	303.06		6,470,570
TOTAL ADDITIONS	56	325.00		18,200
~~TOTAL SUBTRACTIONS~~	NONE			
EMPTY WEIGHT	21,407	303.11	19.47	6,488,770

WEIGHED BY: ____________ WITNESSED BY: ____________ DATE: ____________

Weighing Aircraft
Figure 201.

LIST OF EFFECTIVE PAGES

CHAPTER 9

TOWING AND TAXIING

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CHAPTER 9

TOWING AND TAXIING

TOWING — DESCRIPTION / OPERATION

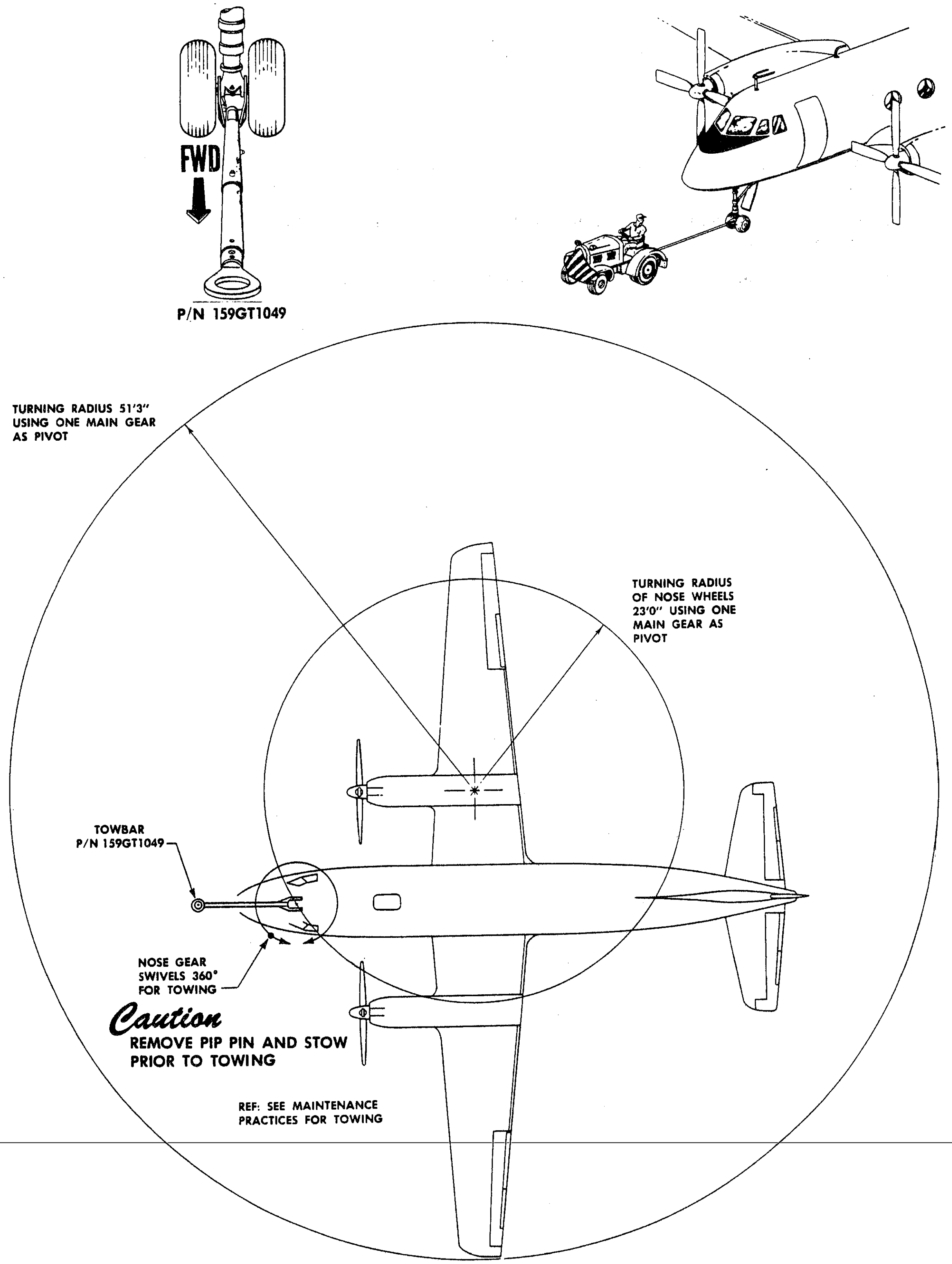
1. General

(See Figure 1)

Provision is made on the nose gear for the installation of tow bar, part number 159GT1049, for towing forward or pushing aft.

NOTE: It is recommended that only towing equipment designed or approved by Gulfstream Aerospace Corporation be used when towing aircraft.

The nose gear provides for an applied towing load not to exceed 7000 pounds.



Towing Provisions and Turning Radii
Figure 1.

TOWING — MAINTENANCE PRACTICES

1. **Aircraft Towing**

   (See Figure 1

   CAUTION: NOSE WHEEL STEERING MECHANISM MUST BE DISCONNECTED BEFORE TOWING AIRCRAFT. ROTATION OF NOSE GEAR BEYOND ITS NORMAL LIMITS OF TRAVEL (60°) CAN CAUSE DAMAGE TO STEERING MECHANISM. WITH STEERING MECHANISM DISCONNECTED, THE NOSE GEAR IS FREE TO ROTATE 360°.

2. **Steering Pin — Removal / Installation**

   A. Removal

   CAUTION: PLIERS, LEVERS, ETC. MUST NOT BE USED FOR REMOVAL OF PIN.

   (1) Remove pip pin safety clip.

   (2) Depress pip pin button while removing pip pin by hand.

   (3) Raise steering unit.

   (4) Insert pip pin through hole in trunnion pin assembly and in lug on strut assembly. This prevents damage to steering assembly during towing.

   B. Installation

   NOTE: Bolt NAS1104-22, Nut MS20365-428, and 2 washers AN 960-416L can be used in place of pip pin. No other combination can be used. Replace nut at each installation.

   (1) Inspect pip pin for proper functioning of plunger and locking balls as follows:

   (a) Check that balls work freely when plunger is depressed.

   (b) Check that balls are immovable when plunger is released.

   (c) Discard pins which fail this inspection.

   (d) Discard pins with a hole (0.050 dia) visible in end of plunger.

   (2) Install pin.

   (3) Inspect for position of locking balls by attempting to withdraw pin without depressing button.

   (4) Install pip pin safety clip.

3. **Towing Procedure and Precautions**

   CAUTION: DURING ALL TOWING OPERATIONS, HAVE A MAN AT THE FLIGHT STATION TO OPERATE BRAKES WHEN NECESSARY.

   A. Ensure that gust lock is in LOCKED position, and that all three control systems are locked in their neutral position.

   B. Check hydraulic brake accumulator pressure gage on copilots flight panel for an indication of at least 1000 psi. If below this pressure proceed as follows:

   (1) Connect batteries. (R and L Nacelle.)

   (2) Place BATT switch on overhead panel in NORMAL.

   (3) Place either pilots or copilots emergency hydraulic pump switch to ON. Pilots switch is just forward of nosewheel steering tiller bar on pilots console. Copilots switch is located on copilots outboard skirt panel (Pump will be heard running).

   (4) Observe hydraulic brake accumulator gage until it indicates that accumulator is fully charged (1425-1500 psi.)

   (5) Turn OFF emergency hydraulic pump switch.

   (6) Turn off battery master switch.

(7) Disconnect batteries.

NOTE: Five full applications of parking and emergency brake system can be made with a fully charged accumulator (1500 psi).

C. Aircraft may be moved by nose landing gear only, on hard surfaces which are approximately level. If conditions require towing up an incline greater than 5°, ensure pull on nose landing gear does not exceed 4655 pounds.

CAUTION: TO PREVENT POSSIBLE DAMAGE TO NOSE WHEEL SELF CENTERING CAMS, DO NOT TOW AIRCRAFT IF DIMENSION "X", AS SHOWN ON NOSE GEAR STRUT INFLATION INSTRUCTION PLATE, EXCEEDS 12.80 INCHES.

TOW BAR, PART NUMBER 159GT1049 MUST BE USED WHEN TOWING AIRCRAFT FROM NOSE GEAR. DO NOT ATTEMPT TO USE TOW BARS DESIGNED FOR OTHER AIRCRAFT.

(1) Ensure landing gear is free of obstructions and brakes are released.

(2) Install pip pin in drag brace (Aircraft having ASC 226).

(3) When towing with engines removed, add 1500 pounds ballast to forward baggage compartment and entrance area to prevent a critical rearward shift of center of gravity.

CAUTION: AIRSTAIR DOOR SHOULD BE CLOSED OR SUPPORTED WITH JURY CABLE 159GT1054 BEFORE TOWING. DAMAGE TO AIRSTAIR DOOR WILL RESULT IF THIS PROCEDURE IS NOT ADHERED TO.

(4) When towing is complete remove pip pin installed in Step C.(2) above.

TAXIING - DESCRIPTION — DESCRIPTION / OPERATION

1. General

NOTE: Taxi with gust lock engaged to prevent damage to flight control stops and to prevent forward motion of the propeller fine pitch lock lever.

Refer to Gulfstream Flight Manual for additional Taxiing information.

LIST OF EFFECTIVE PAGES

CHAPTER 10

PARKING AND MOORING

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CHAPTER 10

PARKING AND MOORING

PARKING AND MOORING — DESCRIPTION / OPERATION

1. **General**

Provisions are incorporated on the main and nose landing gear for mooring the aircraft. Spring-loaded mooring rings are arranged as follows:

- Main gear — one each strut.
- Nose gear — two, one on each side of the strut, just below the taxi lights.

All mooring rings are made to pivot, and when not in use for mooring are held down by a torsion spring to a position against the strut in order to clear lines and structure in the wheel wells when the gear is retracted.

Parking the aircraft involves the use of certain ground safety locks, parking brakes, chocks, and certain protective covers depending on conditions and duration of parking.

It is strongly recommended that the aircraft be parked in a laterally level condition. If the parking area is pitched, it is recommended that the lateral axis be level, and the nose positioned so that it is pitched down.

2. **Static Electricity — Bonding / Grounding**

Static electricity may be generated by the contact and separation of dissimilar material. For example, static electricity is generated when a fluid flows through a pipe or from an orifice into a tank.

The principal hazards created by static electricity are those of fire and explosion caused by spark discharges of accumulated electrical charges in the presence of flammable or explosive vapors, gases, or dust. A spark between two bodies occurs when there is not a good electrical conductive path between them. Hence grounding and bonding of flammable liquid containers is necessary to prevent static electricity from causing a spark.

A point of greater danger from a static spark is the place where a flammable vapor may be present in the air, such as the outlet of a flammable liquid fill pipe, hose nozzle, near an open flammable liquid container, and around a tank truck fill opening or barrel bunghole. Static spark ignition sources are prevented by bonding or grounding or both.

The terms BONDING AND GROUNDING often have been mistakenly used interchangeably. BONDING is done to eliminate a difference in potential between objects. GROUNDING is done to eliminate a difference in potential between an object and ground. Bonding and grounding are effectively applied to conductive bodies.

Permanent bonding and grounding shall be done with screw type clamps. When using alligator-type clamping for temporary grounding and bonding, be sure to attach the clamp by working it from side to side so that positive contact is made with the metal to be grounded. Painted surfaces may insulate and cause poor bonding between metals. Paint should be removed to expose the metal at grounding points. Usually the teeth of the alligator clamp, worked side to side, will accomplish this.

3. **Aircraft Storage, General**

The following procedures are intended as a guide to use in maintaining the aircraft and its equipment in satisfactory condition during an extended lay up period of up to six months.

Safety, especially with regards to fueling and grounding, should never be compromised. Fire hazards should be reduced to a minimum. It is recommended that all operators have on hand a copy of "National Fire Codes-Volume 10 (Transportation)." See, in particular, NFPA No. 411, "Airport Ramp Hazards," a section of this book. It may be purchased from:

National Fire Protection Association
60 Batterymarch Street
Boston, Massachusetts 02110

Listed below are the documents for vendor preservation and storage instructions:

Rolls-Royce Dart Maintenance Manual (M-Da7-G)

Garrett GTC85-37-2 Maintenance Manual (49-21-51)
Dowty Rotol Propeller & Ancillary Maintenance Manual (865/1)
Dowty Rotol Accessory Gearbox & Driveshaft Maintenance Manual (865A/1)
National Fire Codes - Volume 10 (Transportation)

PARKING AND MOORING — MAINTENANCE PRACTICES

1. **Parking and Mooring Equipment**

   A. Ground Safety Locks

   To prevent inadvertent retracting of the landing gear while on the ground, the following ground safety locks are available and should be installed during parking and mooring operations:

   (1) The main gear ground safety lock is a pip pin that is placed in the lower end of the combination side brace, retraction cylinder. The pip pin prevents the internal locking feature of the retraction cylinder from unlocking.

   (2) The nose gear ground safety lock is a pip pin inserted through the lock hook on the trunnion which prevents the nose gear overcenter downlock linkage from unlocking.

   The nose and main landing gear ground lock safety pins are designed to withstand full hydraulic system pressure.

   B. Parking Brake

   The PARK/EMER brake is utilized to apply main gear wheel brakes during parking procedures. It is operated by a red and white striped handle located on pedestal in the cockpit. Application of the parking brakes is accomplished by pulling the handle aft and rotating it 1/4 turn to lock it in the aft position. Parking brakes will then be applied provided that there is sufficient pressure available, as indicated by the brake accumulator pressure indicator gage on the copilots inboard skirt panel. At least 1000 psi must be indicated on the gage for brake application. The PARK/EMER brake is released by rotating it back to the horizontal position, and a spring load will return it to the brakes released position.

   Do not set brakes while they are in a heated condition. Prevention of brake leakage and longer seal life can be achieved if the brake seals are not held under pressure during the cooling down period. After the brakes have sufficiently cooled, the parking brake can be set regardless of local temperature.

   C. Gust Lock

   A gust lock has been provided to enable the crew to lock all flight control surfaces without the use of external locking devices. The gust lock lever is located on pedestal in the cockpit. It is red, with the words GUST LOCK engraved on the head in white lettering. This is a double detented, two position lever. If released, it is in the forward or OFF position. If applied, it is in the aft or ON position. Detents incorporated in the linkage will lock the lever in the position selected. To apply or release the gust lock, the aft end of the lever head must be depressed in order to release the detent, and the lever moved to the appropriate position until the opposite detent is reached. On application of the gust lock lever to the ON, each flight control will be protected in gusts up to 60 knots.

   D. Tiedown Lines, Chocks, Etc.

   (1) Sufficient tiedown lines are necessary if the aircraft is to be moored. The size of the lines depend on the expected weather conditions and is specified in the mooring procedure.

   (2) Wheel chocks are needed, depending on duration and condition of the parking and mooring.

   E. Protective Covers

   Certain external openings and areas must be protected by the use of special protective devices when parked or moored, depending on duration and weather condition.

2. Parking Procedures

A. Ensure gust lock lever is ON.

B. Check brake accumulator pressure gage on copilots inboard skirt panel for an indication of at least 1000 psig. If accumulator pressure is below the 1425 psig pressure, recharge accumulator.

CAUTION: DO NOT SET BRAKES PERMANENTLY WHILE THEY ARE IN A HEATED CONDITION.

C. Set parking brakes by pulling PARK/EMER brake handle in cockpit.

D. Chock main landing gear wheel fore and aft.

E. Release parking brakes after wheel chocks are in place by releasing PARK/EMER brake handle in cockpit.

F. Protect tires with covers.

G. Install the following protective covers.

(1) Install Ram scoop (blower)plug.

(2) Install cover on ram air exhaust.

(3) Install ram air, air conditioning duct cover.

(4) Install plug in APU exhaust.

(5) Install tailpipe duct cover.

(6) Install oil inlet duct plug.

(7) Install engine cover.

(8) Install cover on wing anti-ice exhaust port.

(9) Install pitot tube cover.

(10) Install propeller blade cover.

NOTE: Engine cover eliminates need for 1, 6 and 7 above.

H. Close, latch, and lock main and baggage doors as required.

I. Remove battery disconnects from batteries. This action will remove battery power from the coils of left and right feeder #1 and #2 relays. If batteries are left connected for more than 12 hours, battery voltage may be less than 24 volts.

J. Moor aircraft if weather conditions make it advisable.

3. Mooring Procedures

(See Figure 201.)

A. If wind is expected to exceed 30 knots due to a severe storm or wind condition, aircraft should be housed. If housing the aircraft, or flying the aircraft to a safe location is not possible, the following procedure should be used:

(1) Space aircraft from other aircraft or structures so that a minimum clearance around the aircraft is equivalent to its maximum length plus 15 feet.

(2) Park aircraft on level ground, with nose into wind. Follow Parking Procedure, this Section.

(3) Attach tiedown lines to mooring rings on both the main and nose gear.

NOTE: The prevailing weather conditions will determine the number of tiedown lines.

(4) Ensure main and baggage doors are latched and locked, landing gear doors are closed, and control surface gust lock is ON.

CAUTION: ENSURE TIEDOWN FITTINGS ARE FREE TO PIVOT, SPRINGHELD DOWN, AND PREFERABLY SAFETY WIRED BEFORE FLIGHT, AFTER MOORING, LINES ARE REMOVED.

CAUTION: A QUALIFIED PERSON MUST BE IN THE COCKPIT WHEN AIRCRAFT IS BEING TOWED IN ORDER TO MAINTAIN CONTROL BY USE OF BRAKES. IN ADDITION, A PERSON SHOULD BE STATIONED AT EACH WING TIP DURING AIRCRAFT MOVEMENT.

(5) If possible, choose an area for storage away from runways, taxiways, and other areas where dust and dirt from prop wash and jet exhaust may be a problem. The aircraft should be moored heading into prevailing wind direction, and mooring provisions should take into consideration possibility of high winds.

(6) The availability of towing equipment and servicing facilities should be considered in picking the time for placing aircraft in storage area. Lubrication, preservation, cleaning, servicing, and installation of covering and seals may be performed before aircraft is towed to storage site, however, drainage of aircraft systems and other equipment, and draining and inerting of the fuel system are best performed at storage site.

(7) Ensure brake accumulator has sufficient pressure. If necessary, the auxiliary hydraulic pump may be run to recharge the brake accumulator. Electrical power must be available to operate the pump.

(8) Disconnect nose landing gear torque links before towing.

(9) Place nose wheel in fore and aft position and reconnect torque links after towbar has been removed.

(10) Jack aircraft to relieve weight on tires when feasible. If facilities are not available for blocking, place chocks fore and aft of each wheel with each pair of chocks tied together with rope, wood cleats, or any other means which will prevent chocks from slipping away from the tires. Sand bags may be used for aircraft moored on steel mats.

CAUTION: DO NOT SET THE PARKING BRAKES AT ANY TIME DURING STORAGE. DO NOT USE AUTOMATIC PILOT TO LOCK THE CONTROL SURFACES.

(11) Lock control surfaces with internal locks.

CAUTION: WITH THE REMOVAL OF EQUIPMENT PRIOR TO STORAGE, THE POSSIBILITY OF AIRCRAFT SQUATTING OR SITTING DOWN ON ITS TAIL EXISTS. AS A PRECAUTIONARY MEASURE, ADD BALLAST IN FORWARD SECTION OF AIRCRAFT (TIED DOWN IF AIRCRAFT IS TO BE TAXIED) WHENEVER EQUIPMENT IS TO BE REMOVED FOR STORAGE.

REMOVING EQUIPMENT FROM THE FORWARD PORTION OF A DEFUELED AIRCRAFT WILL MOVE THE AIRCRAFT'S C.G. AFT, WHICH IS PARTICULARLY DANGEROUS IF, AT THE SAME TIME, CARGO IS LOADED INTO THE BAGGAGE COMPARTMENT OR A FEW MEN ARE WORKING IN BOILER ROOM OR ON THE STABILIZER.

NOTE: Prior to delivery, the aircraft has ballast weights installed. Do not remove these weights without ensuring that the aircraft center of gravity is forward of Fuselage station 463. In locating ballast weight, be sure not to block exits and maintain a clear aisle space.

(12) Ensure a good electrostatic ground is available and aircraft is properly connected to it.

B. Landing Gear

(1) Install all ground safety lockpins. Check for leaks and correct as required.

(2) Lubricate and grease landing gear system. (See Chapter 12.)

(3) With aircraft on jacks, perform a functional test of landing gear (gear swing) every 60 days.

NOTE: The gears must be properly inflated with no protective paper attached.

(4) If aircraft is to be stored for more than 30 days, with the aircraft on jacks, preserve shock struts as follows:

(a) Deflate struts. (See Chapter 32 for Servicing.)

(b) Do not inflate struts during storage period unless the aircraft is to be moved. Prior to moving aircraft, or performing a landing gear functional test (gear swing), inflate the struts

with nitrogen. (See Chapter 32 for Servicing.) After moving aircraft or performing a "gear swing", jack the aircraft and deflate struts.

(5) Coat exposed portion of piston with a soft film corrosion preventive compound or petroleum jelly, Federal Specification VV-2-236.

(6) Wrap the exposed portion of the piston with creped greaseproof paper. Secure with cloth-backed adhesive tape.

(7) Protect perserved strut with a loose fitting waterproof paper or canvas boot.

C. Tire Care

(1) Check tires for proper inflation prior to storage. Refer to the Goodyear Tire Information Booklet, 700-862-931-538 and 821-932-538, for storage instructions.

(2) Aircraft tires should be rotated every seven days. Mark the date and position on the tire with chalk to ensure tires are not reparked on same tire area. This is also beneficial to wheel bearings. (Tire manufacturer's recommendations may be used.)

(3) Tires and brakes should be completely covered.

(4) The preferred method is to equip aircraft with unserviceable tires and bearings. Tire pressures need not be checked in this case.

NOTE: Equip aircraft with serviceable tires prior to flight.

D. Nose Wheel Steering Unit

(1) Cover nose wheel steering unit, including taxi lights.

(2) Perform an operational check of steering unit with each gear swing operation.

(3) Cover wheel well openings.

E. Hydraulic Fluid System

(1) Inspect hydraulic lines and components for leakage. Repair as required.

(2) Drain system at its lowest point until clean fluid is obtained. Fill reservoir to proper level.

(3) Wipe exposed surface of actuating rods with a clean cloth and coat with preservative Hi-Lo MS No.1 grease.

(4) Cover exposed surfaces of actuating rods with greaseproof paper, then tape with cloth-backed tape.

(5) Cover oil vents to exclude dust and dirt. Do not make airtight.

(6) Bleed air from accumulator.

(7) Operate external hydraulic pump or electric driven pump to force all the air from accumulator.

(8) Dissipate system pressure.

(9) Remove pressure line in air side and force fluid out by operating external hydraulic pump or electric driven pump in aircraft until piston or diaphragm is forced completely to bottom of the accumulator.

(10) Relieve system pressure.

(11) Recap air side of accumulator.

F. Lubrication

Perform a complete aircraft lubrication. (See Chapter 12.) Spray all moveable joints with oil per Federal Specification VV-L-800.

G. Batteries

(1) Check cockpit switches for proper position.

(2) Disconnect batteries from aircraft electrical system, then vent system. Remove batteries from aircraft if aircraft is to be inactive for 15 days or more.

(3) Batteries should be discharged, shorted, and stored at room temperature.

H. Electronic Equipment

CAUTION: WITH THE REMOVAL OF EQUIPMENT PRIOR TO STORAGE, THE POSSIBILITY OF THE AIRCRAFT SQUATTING OR SITTING DOWN ON ITS TAIL EXISTS. AS A PRECAUTIONARY MEASURE, BALLAST SHOULD BE ADDED IN THE FORWARD SECTION OF THE AIRCRAFT (TIED DOWN IF THE AIRCRAFT IS TO BE TAXIED) WHENEVER EQUIPMENT IS TO BE REMOVED FOR STORAGE.

The following electronic equipment should be removed from the aircraft, boxed, and stored if aircraft storage time is to exceed 90 days. If storage is less than 90 days, equipment may remain installed but must be operated for at least one hour every week.

(1) All equipment mounted in radio rack.

(2) All electronic equipment mounted in cockpit side consoles and center console.

(3) Vertical gyros.

(4) Directional gyros.

(5) DC voltage regulators.

(6) AC voltage regulators.

(7) Inverters.

(8) All similar equipment installed by outfitter or operator.

NOTE: Removal of DC generators is not necessary. Plug inlet and exhaust ports on engine cowl, and inspect for obstruction prior to activation of aircraft.

I. Exterior Protection

(1) For aircraft to be stored two months or less:

(a) Periodic rinsing with tap water should be accomplished as necessary to prevent deterioration of aircraft exterior surfaces that are not protected with an organic coating. If excessive streaking or corrosion is encountered, the aircraft should be cleaned and the rinsing frequently increased.

(b) Remove grease, oil, or exhaust deposits after completion of ground runs and flight tests.

(2) For aircraft to be stored more than two months:

(a) Touch up painted surfaces with both primer and paint in areas where needed.

(b) Apply a preservative coating to all windows, (a wax coating is sufficient).

NOTE: If the aircraft is scheduled for repair or maintenance, or must be stored where windows will be subjected to possible abrasive damage, clean external surfaces thoroughly and apply a layer of Protex-10V paper or equivalent.

J. Interior Protection

(1) Remove debris such as paper, rags, oil, water, etc.

(2) Clean interior in accordance with outfitter's recommendations to remove dirt and organic matter. Allow interior to air dry completely before storing aircraft.

(3) Clean toilets and waste water systems by flushing and draining.

(4) Drain all water and other liquids from tanks, galley, etc.

(5) Install dust covers as needed. Use only cloth or other porous materials. Do not use plastic film.

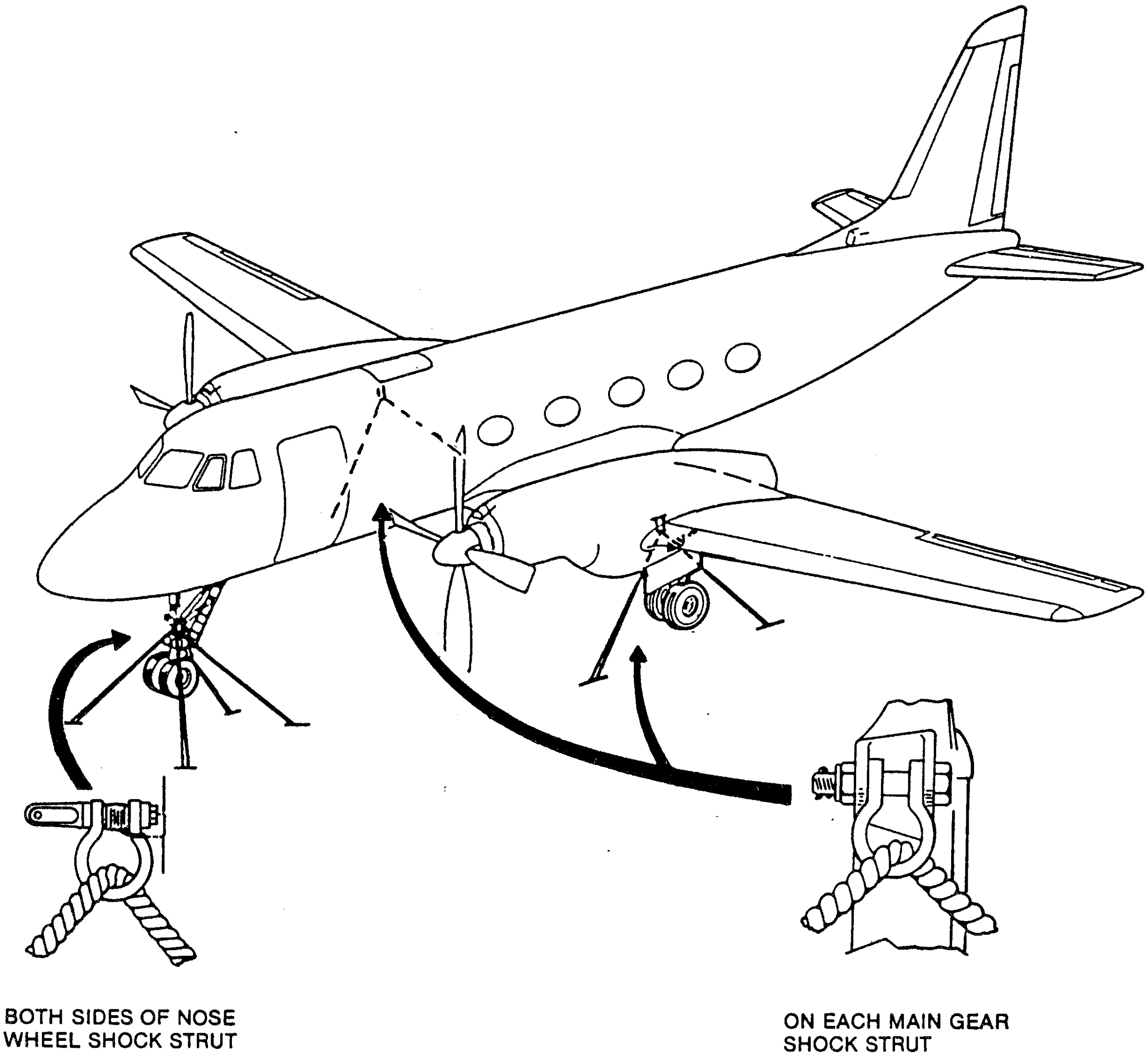
(6) Cover, but do not make airtight, all exterior openings to prevent entry of rodents, birds, etc.

4. Final Inspection Before Storage

Before final storage, inspect wings, aft interior fuselage, wheel wells, tail surfaces, and all moveable control surfaces for entrapped water or other fluids. Pools of hydraulic fluid should be cleaned up as they may cover trapped water.

	INSPECTION INTERVAL						
	24 HOURS	48 HOURS	7 DAYS	15 DAYS	30 DAYS	60 DAYS	90 DAYS
Walkaround inspection							
(a) All doors are properly closed	X						
(b) Safety ropes and signs are intact and in place	X						
(c) Inspect external cables	X						
Inspect oxygen content of vapors in the fuel tank		X					
Check tire pressure and rotate tires			X				
Engine run-up (if not preserved). (See Rolls-Royce Maintenance Manual.)			X				
Inspect the following:							
(a) Safety pins and red streamers are in place and intact				X			
(b) Check areas where rodents or insects may habitate.				X			
Operate all control surfaces and where possible move cockpit controls through full travel.					X		
Check landing gear struts for proper inflation (if left inflated).						X	
Perform a functional test of the following:							
(a) Landing gear (gear swing)						X	
(b) Steering unit						X	
Inspect all engine and flight controls.							X
Run engines. (See Rolls-Royce Maintenance Manual.)							X

TYPICAL INSIDE STORAGE INSPECTION AND SERVICING CHART



NOTE:
USE 1-INCH DIAMETER ROPE WHEN WIND SPEED IS 0-17 MPH.
USE 1-1/2 INCH DIAMETER ROPE WHEN WIND SPEED IS 17-75 MPH.

CAUTION: AFTER MOORING AND BEFORE FLIGHT, CHECK THAT THE MOORING RINGS ARE FREE TO PIVOT AND THAT THE TORSION SPRINGS ARE HOLDING THE RINGS IN CONTACT WITH THE MAIN AND NOSE STRUTS. AS AN ADDED PRECAUTION IT IS STRONGLY RECOMMENDED THAT THEY BE LOCK-WIRED DOWN TO THE STRUT AFTER USE.

Mooring Provisions
Figure 201.

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(3) Wipe the exposed surface of actuating rods with a clean cloth and coat with preservative Hi-Lo Ms No.1 grease.

(4) Cover exposed surfaces of actuating rods with greaseproof paper, then tape with clothbacked tape.

(5) Cover oil vents to exclude dust and dirt. Do not make airtight.

(6) Bleed air from accumulators.

(7) Operate external hydraulic pump or auxiliary pump to force all the air from the accumulators.

(8) Dissipate the system pressure.

(9) Remove the pressure line in the air side and force the air out by operating the external hydraulic pump or electric driven pump in the aircraft until the piston or diaphragm is forced completely to the bottom of the accumulator .

(10) Relieve system pressure. Install cap on air side of accumulator, and cap air line.

N. Lubrication

Perform a complete aircraft lubrication. Refer to Chapter 12-Servicing. Spray all clevises, rod ends, and other moveable joints with Preservative Oil, Federal Specification VV-L-800A or equivalent.

O. Batteries

Disconnect batteries from the aircraft electrical system, then vent the system. Remove batteries from the aircraft if the aircraft is to be inactive for 15 days or more. Refer to Chapter 24 for Battery Storage Procedures.

P. Electronics Equipment

The following electronic equipment should be removed from the aircraft, boxed, and stored if aircraft storage time is to exceed 90 days. If storage is less than 90 days, equipment may remain installed but must be operated for at lease one hour every week. (See "Outside Storage Inspection Requirements".)

CAUTION: WITH THE REMOVAL OF EQUIPMENT PRIOR TO STORAGE THE POSSIBILITY OF THE AIRCRAFT SQUATTING OR SETTING DOWN ON ITS TAIL EXISTS. AS A PRECAUTIONARY MEASURE, BALLAST SHOULD BE ADDED IN THE FORWARD SECTION OF THE AIRCRAFT (TIED DOWN IF THE AIRCRAFT IS TO BE TAXIED) WHENEVER EQUIPMENT IS TO BE REMOVED FOR STORAGE.

(1) All equipment mounted in the radio rack.

(2) All electronic equipment mounted in the cockpit side consoles and center console.

(3) Vertical gyros

(4) Directional gyros

(5) DC Voltage regulators

~~(6) AC Voltage regulators~~

(7) Inverters

(8) All similiar equipment installed by outfitter or operator.

Q. Exterior Protection

(1) For aircraft to be stored two months or less:

(a) Where required, prime and paint.

(b) Remove grease, oil and exhaust deposits after completion of ground runs.

(c) Depending on ambient atmospheric conditions, rinsing with tap water should be accomplished as necessary to prevent deterioration of aircraft exterior surfaces that are not protected with an organic coating. If excessive streaking or corrosion is encountered, the aircraft should be cleaned and the rinsing frequency increased.

(2) For aircraft to be stored more than two months:

(a) Touch up painted surfaces with both primer and paint in areas where needed.

(b) Apply a preservative coating to all windows. (A wax coating is sufficient.)

NOTE: If the aircraft is scheduled for repair or maintenance, or must be stored where windows will be subjected to possible abrasive damage, clean external surfaces thoroughly and apply a layer of Protex-10V paper or equivalent.

R. Interior Protection

(1) Remove debris such as paper, rags, oil, water, etc.

(2) Clean interior in accordance with outfitters recommendations to remove dirt and organic matter. Allow interior to air dry completely before storing aircraft.

(3) Clean toilets and waste systems by flushing and draining.

(4) Drain all water and other liquids from tanks, galley, etc.

(5) Install dust covers as needed. Use only cloth or other porous materials. Do not use plastic film.

(6) Cover, but do not make airtight, all exterior openings to prevent entry of rodents, birds, etc.

S. Water Methanol System

(1) Drain and flush water methanol system.

(2) Fill systems with fresh water methanol up to filler cap. Make sure that lines and filters are full.

T. Final Inspection Before Storage

Before final storage,inspect wings, aft interior fuselage, wheel wells, tail surfaces, and all moveable control surfaces for entrapped water or other fluids.

5. **Outside Storage Inspection Requirements**

A. Daily Inspection

Do a walk around inspection making sure:

(1) Wheel chocks are in place and secured together

(2) Tie-down lines are secure. (If manila rope is used ensure tie-down rope has slack.)

(3) Various protective covers are secure and intact (Refer to 2G and 4B(7) in this section.)

(4) All doors and windows are properly closed

(5) All safety ropes and caution signs are intact and still in place.

B. 48 Hour Inspection

Make sure that the oxygen content of the vapor in the fuel tanks is at a safe level. Additional nitrogen should be added as required to maintain a safe tank atmosphere.

C. Seven Day Inspection

(1) Run engines unless they are preserved, nine minutes at high ground idle, and one minute at cruising speed. Activate and operate engine synchronizing equipment.

(2) Check tire pressure and rotate tires. Mark and date the position of the tires.

NOTE: If aircraft is equipped with unserviceable tires and bearings, rotation is not required.

(3) If electronic equipment is still installed, operate for one hour. Observe for proper operation.

(4) Inspect propeller blades and spinners for corrosion.

(5) Operate propeller pitch through full range.

D. 15 Day Inspection

Inspect as follows:

(1) Make sure the gust lock is properly engaged and all control surfaces are locked.

(2) Make sure safety pins and red streamers are in place and intact.

(3) Inspect drain holes to be sure they are free of obstructions.

(4) Inspect for accumulations of water. If drainage holes are necessary, contact Gulfstream American.

(5) Inspect for signs of corrosion. If found, take corrective measures.

NOTE: The first appearance of corrosion on unpainted aluminum surfaces is in the form of a whitish deposit on the surface, especially along seam lap joints or in niches and crevices where traces of cleaning compound may have accumulated. Particular attention should be given to the underside of fuselage, wings, wing flaps, actuating mechanisms, tubular type structures, etc., since moisture deposited in these locations does not evaporate as rapidly as areas exposed to the sun and air. Corrosion on painted surfaces, although not as prevalent as on unpainted surfaces, will be characterized by blisters or a scaly appearance of the paint:

(6) Inspect areas where birds, mice, or insects may deposit materials that would interfere with the proper operation of the aircraft or would trap moisture and cause corrosion.

E. 30 Day Inspection

Inspect as follows:

NOTE: Service all control surface lubrication points before and after operation. Refer to Chapter 12-Servicing.

(1) Disengage gust lock and operate all control surfaces listening for any abnormal sounds.

(2) Feather and unfeather propellers if engine has not been run, using the feathering unit.

F. 60 Day Inspection Interval

Jack aircraft and perform the following:

(1) Check the strut chrome areas for corrosion and recoat if necessary.

(2) If the landing gear struts have been left inflated, deflate, compress strut several times, service, and inflate. (See Chapter 32 for procedures.) Recoat strut.

(3) Perform a functional test of the landing gears and steering unit.

G. 90 Day Inspection

(1) Inspect all engine and flight controls. Preserved engines shall be ground run unless stipulated otherwise by Rolls-Royce. Run engines nine minutes at high idle and one minute at cruising speed.

(2) Carpeting and other interior furnishings should be inspected for musty odors, mildew, or other problems. On a fair day with moderate temperatures and low humidity, shampoo or otherwise clean the carpet and furnishings if required. Open the main entrance door and baggage doors and let the aircraft be aired out. The use of a large circulating fan at the baggage door to pull the air out is adequate. Make sure the carpet and furnishings are completely dry before closing doors.

NOTE: The opening of cockpit windows is not recommended due to the proximity of instruments and electronic equipment and the possibility of sudden rain showers.

6. Inside Storage Procedures

The requirements for inside storage procedures are the same as outside storage procedures except for electronic equipment, which need only be operated at six month intervals unless stipulated otherwise by the manufacturer or installer. Removal for storage and operation at less than six month intervals is not necessary within a controlled constant temperature atmosphere which will prevent condensation.

NOTE: A hangar area where large doors are continuously opened is not a constant temperature atmosphere even when air conditioned.

7. Inside Storage Inspection Procedures

A. Daily Inspection

Do a walk around inspection making sure:

(1) All doors are properly closed

(2) All safety ropes and caution signs are intact and still in place

(3) External cables should be checked for damage and to determine if they are properly located and marked to prevent damage to the equipment or injury to personnel.

B. 48 Hour Inspection

(1) Make sure that the oxygen content of the vapor in the fuel tanks is at a safe level. Additional nitrogen should be added as required to maintain a safe tank atmosphere.

C. Seven Day Inspection

(1) Run engines unless they are preserved, nine minutes at high ground idle and one minute at cruising speed.

NOTE: If aircraft is equipped with unserviceable tires and bearings, rotation is not required.

(2) Check tire pressure and rotate tires. Mark and date the position of the tires.

(3) Inspect propeller blades and spinners for corrosion.

D. 15 Day Inspection

(1) Inspect to make sure safety pins and red streamers are in place and intact.

(2) Check areas where rodents or insects may habitate.

E. 30 Day Inspection

Operate all control surfaces.

F. 60 Day Inspection

Jack the aircraft and perform the following:

(1) Check the strut chrome areas for corrosion and recoat if necessary.

(2) If the landing gear struts have been left inflated, deflate, compress several times, service, and inflate. (See Chpater 32 for servicing procedures.)

(3) Perform a functional test of the landing gears and steering unit. (See Chapter 32 for procedures.)

G. 90 Day Inspection Interval

Inspect all engine and flight controls. Preserved engines shall be run unless stipulated otherwise by Rolls-Royce. Run engines nine minutes at high idle and one minute at cruising speed.

8. **Activation of Aircraft from Storage**

A suitable manned fire truck should stand by the aircraft during fueling, engine run, and electrical system check as the aircraft is returned to flight status. A water/methanol specific gravity test should be performed. See Dart Maintenance Manual Chapter 89.

The checklist prepared at the time of storage should be reviewed as the aircraft is returned to flight status. During this review, an operational or functional test should be performed on all systems. A visual inspection during operational and functional tests should also be made to note and correct any binding or other irregularities noted.

If the storage period exceeds 90 days, a complete inspection of the aircraft is recommended before returning to active status.

Before flight, remove any loose equipment and ballast. Make sure the aircraft center of gravity is within prescribed limits.

	INSPECTION INTERVAL							
	24 Hours	48 Hours	7 Days	15 Days	30 Days	60 Days	90 Days	
Walk around Inspection (a) Wheel chocks in place and secured together (b) Security of tie down lines (If manila rope is used ensure tie down rope has slack) (c) Security of protective covers (d) All doors and windows are properly closed (e) Safety ropes and signs are intact and in place	X							
Inspect Oxygen content of vapors in the fuel tank.		X						
Check tire pressure and rotate tires. Mark and date tire position.			X					
NOTE: If aircraft is equipped with unserviceable tires and bearings, rotation is not required								
Engine run-up (if not perserved)-See Rolls-Royce Maintenance Manual.			X					
Operate electronic equipment for one hour (if left installed).			X					
Inspect propeller blades and spinners for corrosion. Operate propeller through full range.			X					
Ensure the gust lock is properly engaged and all control surfaces are locked.				X				
Inspect the fuselage, wings, tail section, (including all moveable control surfaces) wheel wells, aft interior fuselage, etc., for: (a) Freedom of drain holes (b) Accumulation of trapped water (c) Safety streamers are in place and intact (d) Corrosion (e) Areas accessible to birds, insects, etc.				X				
Operate all control surfaces (lubricate before and after operation). Feather and unfeather propeller.					X			
Check landing gear strut chrome areas for corrosion and recoat if necessary.						X		
Check landing gear struts for proper inflation pressure (if left inflated).						X		
Perform a functional test of the following: (a) Landing gears (b) Steering unit						X		
Inspect all engine and flight controls							X	
Run engines (see Rolls-Royce Maintenance Manual)							X	
Inspect aircraft interior furnishings for odors, mildew or other problems.							X	

TYPICAL OUTSIDE STORAGE INSPECTION AND SERVICING CHART

	INSPECTION INTERVAL						
	24 Hours	48 Hours	7 Days	15 Days	30 Days	60 Days	90 Days
Walk around inspection (a) All doors are properly closed (b) Safety Ropes and signs are in tact and in place. (c) Inspect external cables	X						
Inspect oxygen content of vapors in the fuel tank		X					
NOTE: If aircraft is equipped with unserviceable tires and bearings, rotation is not required.							
Check tire pressure and rotate tires and inspect propeller blades and spinner			X				
Engine run-up (if not preserved) - See Rolls-Royce Maintenance Manual			X				
Inspect for the following: (a) Safety pins and red streamers are in place and intact (b) Check areas where rodents or insects may habitat.				X			
Operate all control surfaces and where possible move cockpit controls through full travel.					X		
Check landing gear struts for proper inflation (if left inflated)						X	
Perform a functional test of the following: (a) Landing gear (b) Steering unit						X	
Inspect all engine and flight controls							X
Run Engines (See Rolls-Royce Maintenance Manual)							X

TYPICAL INSIDE STORAGE INSPECTION AND SERVICING CHART

OUTSIDE STORAGE — DESCRIPTION / OPERATION

1. **General**

Outside storage procedures are based on a humid tropical atmosphere (above 80 percent humidity) or a marine environment. Based on this somewhat severe atmosphere, a layup period exceeding 15 days is considered as having the aircraft in storage. (See Rolls-Royce Dart Maintenance Manual for engine requirements.) Deviations should not be arbitrary, but based on your experience with local climatic conditions.

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OUTSIDE STORAGE — MAINTENANCE PRACTICES

1. **Outside Storage Procedures**

A. Preparation of Power Plant and Engine Fuel System.

Preserve power plant and engine fuel system for storage in accordance with Rolls-Royce Maintenance Manual (M-Da7-G) Chapter 72.

B. Fuel System

(1) If aircraft is to be stored for 30 days or more, purge aircraft fuel tanks with nitrogen gas in accordance with procedures given in this manual, Chapter 28. Install outboard wing vent plug and pressurizing adapter.

NOTE: If storage is to be less than 30 days and engine run-up is to be accomplished every seven days, fuel tanks need not be drained.

(2) Attach red streamers to plugs on defueling drains, outboard wing vent plugs, pressurizing adapter, and fuel gravity filler with the warning that they are not to be opened while the tanks are inerted.

C. Fire Extinguishing System

(1) Inspect engine and APU fire extinguishing systems in nacelles and tail compartment for proper pressure prior to storage.

(2) Inspect all hand fire extinguishers for weight or pressure prior to storage.

D. Environmental Control and Oxygen Systems

(1) Auxiliary Power Unit (APU)

Preserve APU in accordance with Garrett GTC85-37-2 Maintenance Manual (49-21-51.

(2) Aircraft System

(a) Install Ram air scoop inlet/oil cooler air intake cover.
(b) Install cover on nacelle cooling air exhaust.
(c) Install blower air inlet cover.
(d) Install DC generator exhaust cover.
(e) Install DC generator inlet cover.
(f) Install AC generator exhaust cover.
(g) Install AC generator inlet cover.
(h) Install tailpipe outlet plug.
(i) Install nose cowl inlet plug.

(3) Oxygen System

(a) Remove oxygen bottle from aircraft if aircraft is to be out of service for a period exceeding 30 days.

(b) Cap all fittings and tube ends opened by removal of oxygen cylinder to prevent dirt from entering and contaminating the system.

E. Controls

(1) Lubricate flight controls, engine controls, and airstair door in accordance with Chapter 12.

(2) Cycle all flight controls, engine controls, and the airstair door prior to storage to ensure they are operating.

(3) Place flaps to 10°.

F. Furnishings

(1) Cover pitot heads with covers.

(2) Remove overspeed warning switch and place switch in a suitable container for storage. Cap lines in aircraft.

(3) Cover all static air ports, but do not make airtight.

2. **Outside Storage Inspection Requirements**

A. Daily Inspection

Do a walkaround inspection ensuring:

(1) Wheel chocks are in place.

(2) Tiedown lines are secure and tight.

(3) Various protective covers are secure and intact.

(4) All doors are properly closed.

(5) All safety ropes and caution signs are intact and still in place.

B. 48 Hour Inspection

Ensure oxygen content of vapor in the fuel tanks is not above minimum requirements. Add nitrogen as required to maintain a safe tank atmosphere.

C. Seven Day Inspection

(1) Run engines unless they are preserved.

(2) Check the tire pressure and rotate tires.

(3) If electronic equipment is still installed, operate for at least 1 hour every week. Check for proper operation.

D. 15 Day Inspection

Inspect as follows:

(1) Ensure gust lock is on and functioning.

(2) Ensure safety pins and red streamers are in place and intact.

(3) Inspect drain holes to ensure they are free of obstructions.

(4) Inspect for accumulations of water. If drainage holes are necessary, contact Gulfstream Aerospace Engineering.

(5) Inspect for signs of corrosion. If found, take corrective measures.

NOTE: The first appearances of corrosion on unpainted aluminum surfaces is in the form of a whitish deposit on the surface, especially along seam lap joints or in niches and crevices where traces of cleaning compound may have accumulated. Particular attention should be given to the underside of fuselage, wings, wing flaps, actuating mechanisms, tubular type structures, etc., since moisture deposit in these locations does not evaporate as rapidly as areas exposed to the sun and air. Corrosion on painted surfaces, although not as prevalent as on unpainted surfaces, will be characterized by blisters or a scaly appearance of the paint.

(6) Inspect areas where birds, mice, or insects may deposit materials that would interfere with the proper operation of the aircraft or would trap moisture and cause corrosion.

E. 30 Day Inspection

Operate all control surfaces listening for any abnormal sounds. Lubricate before and after operation. (See Chapter 12.)

F. 60 Day Inspection

~~(1) If landing gear struts have been left inflated, check for proper pressure.~~

(2) Check the strut chrome areas for corrosion and recoat if necessary.

(3) Do a functional test of the following:

(a) Landing gear (gear swing)

(b) Steering unit

G. 90 Day Inspection

Inspect all engine and flight controls. Engines shall be run unless stipulated otherwise by Rolls-Royce.

	INSPECTION INTERVAL						
	24 HOURS	48 HOURS	7 DAYS	15 DAYS	30 DAYS	60 DAYS	90 DAYS
Walkaround inspection (a) Wheel chocks in place No (b) Security of tiedown lines (c) Security of protective covers (d) All doors are properly closed (e) Safety ropes and signs are intact and in place	X X X X X						
Inspect oxygen content of vapors in the fuel tank.		X					
Check tire pressure and rotate tires. Mark and date tire position.			X				
Engine run-up (if not preserved) - See Rolls-Royce Maintenance Manual. NO			X				
Operate electronic equipment for one hour (if left installed).			X				
Inspect propeller blades and spinners for corrosion. Operate propeller through full range.			X				
Inspect gust lock for security.				X			
Inspect the fuselage, wings, tail section, (including movable control surfaces) wheel wells, aft interior fuselage, etc., for: (a) Freedom of drain holes (b) Accumulation of trapped water (c) Safety streamers are in place and intact No (d) Corrosion (e) Areas accessible to birds, insects, etc.				X X X X X			
Operate all control surfaces (lubricate before and after operation). Feather and unfeather propeller.					X		
Check landing gear strut chrome areas for corrosion and recoat if necessary.						X	
Check landing gear struts for proper inflation pressure (if left inflated).						X	
Perform a functional test of the following: (a) Landing gears (gear swing) No (b) Steering unit						X X	
Inspect all engine and flight controls.							X
Run engines. (See Rolls-Royce Maintenance Manual.) No							X
Inspect aircraft interior furnishings for odors, mildew or other problems.							X

TYPICAL OUTSIDE STORAGE INSPECTION AND SERVICING CHART

ACTIVATION FROM STORAGE — DESCRIPTION / OPERATION

1. **Activation of Aircraft from Storage**

   It is recommended that a fire truck stand by during fueling, engine run and electrical system check when reactivating an aircraft from storage. The checklist prepared at time of storage should be reviewed as aircraft is being reactivated. During this review, an operational or functional test should be performed on all systems. During these tests a visual inspection should be made to note and correct any binding or other problems. If storage period exceeds 90 days, a complete inspection of aircraft is recommended before returning to active status.

   Before flight, remove all loose equipment and ballast. Ensure aircraft center of gravity is within prescribed limits.

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CHAPTER 12

SERVICING

SERVICING — DESCRIPTION / OPERATION

1. General

A. Servicing/Replenishing Fluid Systems

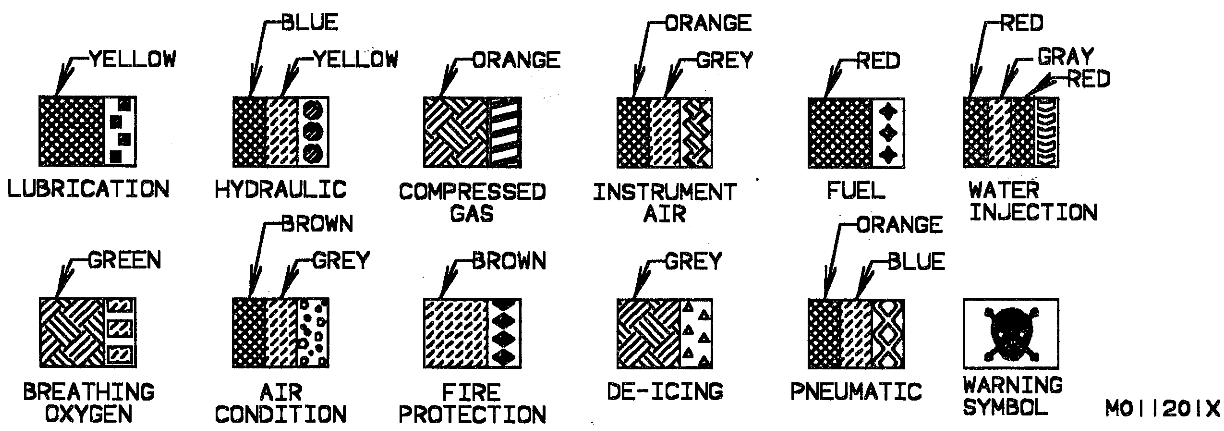
When servicing and replenishing the various systems, the mechanic or serviceman must be very observant, always on the lookout for leaks in plumbing, frayed or broken strands in visible control cables, security of all components and access doors and panels. Pay particular attention to the following:

- Drip pans under accessories.
- Fire extinguisher pressure gages.
- Tire condition.
- Tire slip marks.
- Oxygen pressure indicator.
- Steering control cables on nose gear.
- Hydraulic lines on all landing gear.
- Access doors closed and secure.
- Fuel stains caused by leaks from fuel tanks, plumbing, or components.
- All visible plumbing for leaks.

A COLOR CODING SYSTEM HAS BEEN DEVISED TO AID THE MECHANIC IN SERVICING THE AIRCRAFT AND TO FACILITATE THE IDENTIFICATION OF TUBING OF A PARTICULAR SYSTEM. THIS CODING SYSTEM CONSISTS OF A SERIES OF SYMBOLS AND COLORS WHICH WHEN PLACED IN A CERTAIN ORDER WILL DENOTE AN AIRCRAFT SYSTEM. IN ADDITION TO THE COMBINATIONS OF COLORS AND SYMBOLS THERE IS A WARNING SYMBOL THAT IS USED WHEN THE CONTENTS OF THE LINE IS CONSIDERED TO BE DANGEROUS TO MAINTENANCE PERSONNEL. THIS SYMBOL IS PLACED ADJACENT TO THE SYSTEM SYMBOL.

FOR FURTHER INFORMATION SEE MIL-STD-1247.

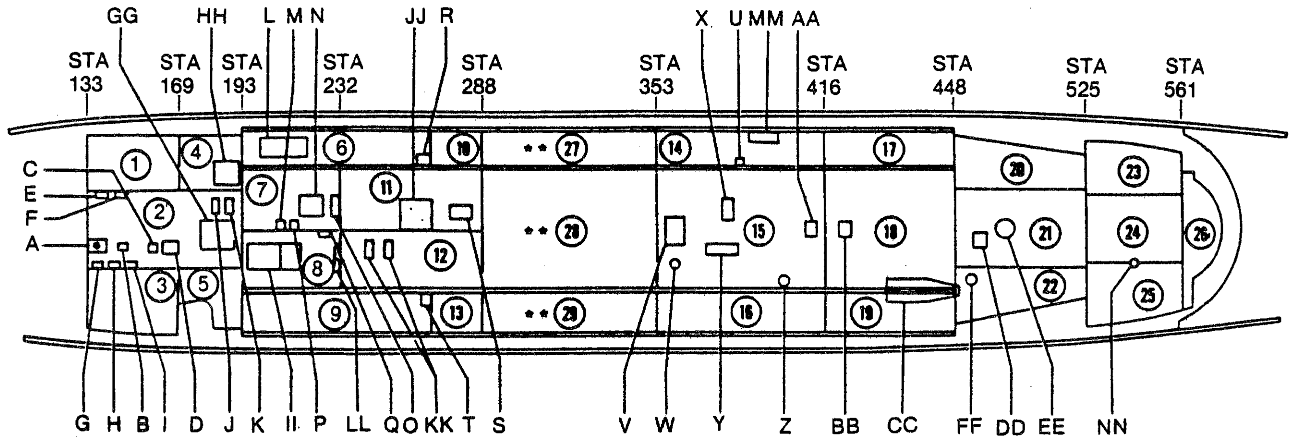
NOTE: Use the Line Coding Chart as an aid in identifying the different types of tubing in the aircraft. (See Figure 1.)



Line Coding Chart
Figure 1.

B. Components location (See Figure 2) for components which can be reached by removing floorboards).

C. Replenishing data (See Figure 3 thru Figure 6).



A. RIGHT FRONT WINDSHIELD POWER UNIT
B. LEFT DV WINDSHIELD POWER UNIT
C. RIGHT DV WINDSHIELD POWER UNIT
D. LEFT FRONT WINDSHIELD POWER UNIT
E. RIGHT DV WINDOW CONTROLLER
F. LEFT DV WINDOW CONTROLLER
G. DEICE SWITCH BOX
H. RIGHT FRONT WINDSHIELD CONTROLLER
I. LEFT FRONT WINDSHIELD CONTROLLER
J. LEFT DC GENERATOR OVERVOLTAGE RELAY
K. RIGHT DC GENERATOR OVERVOLTAGE RELAY
L. ANTI-CRASH RELAY/PITOT HEAT RELAY PANEL
M. LEFT TGT RESISTANCE SPOOL
N. MASTER WARNING LIGHTS CONTROL BOX
O. WING AND TAIL DEICER TIMER
P. RIGHT TGT RESISTANCE SPOOL
Q. 115V TO 26V AC INSTRUMENT TRANSFORMERS
R. RIGHT HYDRAULIC SHUTOFF VALVE
S. MAIN BUS TERMINAL BOX
T. LEFT HYDRAULIC SHUTOFF VALVE
U. BLOWER SHUTOFF VALVE
V. FLAP DRIVE MOTOR AND GEARBOX
W. COCKPIT DUCT ELEMENT
X. CABIN WINDOW DESICCATOR UNIT
Y. CABIN/COCKPIT TEMPERATURE CONTROL BOX
Z. CABIN ANTICIPATOR ELEMENT
AA. COCKPIT TEMPERATURE CONTROL VALVE
BB. CABIN TEMPERATURE CONTROL VALVE
CC. WATER SEPARATOR
DD. WATER SEPARATOR ANTI-ICE CONTROL BOX
EE. BOTTOM ANTI-COLLISION LIGHT WELL
FF. WATER SEPARATOR ANTI-ICE SENSOR ELEMENT
GG. WINDSHIELD HEAT RELAY BOX
HH. WINDSHIELD JUNCTION BOX
II. MID RELAY BOX
JJ. PROP JUNCTION BOX
KK. AC TRANSFORMER-RECTIFIERS
LL. FUEL HEAT TIMER BOX (AIRCRAFT 1-82 and 114, HAVING ASC 117 AND AIRCRAFT 83-200, 322 AND 323).
MM. SECOND AIR CONDITIONER SILENCER (AIRCRAFT 1-86, and 114, HAVING ASC 103A AND AIRCRAFT 87-200, 322 AND 323).
NN. WATER SEPARATOR ANTI-ICE VALVE (AIRCRAFT 1-80, and 114, HAVING ASC 128A AND AIRCRAFT 81-200, and 322 AND 323).

*AIRCRAFT 1-120, ONLY

**REMOVABLE ON AIRCRAFT 28-200, 322 AND 323, EXCLUDING 114.

Fuselage Floor Plan and Panel Numbers
Figure 2.

INDEX	UNIT	CONTENTS	SPECIFICATION AND GRADE	CAPACITY			REMARKS
Note (A) 1	(See Figure 4) Water/Methanol Tank (4)	Water/Methanol	Rolls Royce Spec AEP-1-W/M (Latest issue)	US Gal 42.0	Imp Gal 34.97	Metric Liters 158.97	
Note (B) 1	(See Figure 5) Water/Methanol Tank (2)	Water/Methanol	Rolls Royce Spec AEP-1-W/M (Latest issue)	US Gal 23 each tank	Imp Gal 19.15 each tank	Metric Liters 87.17 each tank	Filler Outboard of Fuel Tank Top of Wing
Note (C) 1	(See Figure 6) Water/Methanol Tank (2)	Water/Methanol	Rolls Royce Spec AEP-1-W/M (Latest issue)	US Gal 23.7 each tank	Imp Gal 18.81 each tank	Metric Liters 89.58 each tank	Aft Wing Fuselage Fillet
Note (D) 2	(See Figure 4 & 5) Fuel Tank (2)	Fuel		US Gal 775.0 each tank	Imp Gal 654.34 each tank	Metric Liters 2933.38 each tank	Top of Wing Inboard of Wing Gap Band (Wing Station 308)
Note (C) 2	(See Figure 6) Fuel Tank (16)	Fuel	See Note (G)	US Gal 1794 all tanks	Imp Gal 1427 all tanks	Metric Liters 6781.32 all tanks	Top of Wing Outboard of Wing Gap Band (Wing Station 308)
3	Accessory Gearbox (2)	Oil	(See Note F)	US Gal 0.75 each gearbox	Imp Gal 0.63 each gearbox	Metric Liter 2.84 each gearbox	Top of each Nacelle
4	Accessory Gearbox Drive Universal (2)	Grease	Andok 260 Specn. DTD 900/4802 Marfak 3 HD (Alt)	As required			Each Engine
5	Auxiliary Power Unit	Oil	(See Note F)	US Gal 1.69	Imp Gal 1.41	Metric Liters 6.4	Inside tail compartment
6	Main Landing Gear Shock Strut (2)	Hydraulic Fluid Dry Nitrogen	(See Note E)	US Gal each Strut #1-0-96 #2-0-99	Imp Gal each strut #1-0-8 #2-0-82	Metric Liters each strut #1-3.63 #2-3.75	A/C 1-162 not having ASC 169 or ASC 175 A/C 1-162 having ASC 169 or ASC 175 and A/C 163-200. See Instruction Plate on main gear doors
7	Main Wheel Tire (4)	Nitrogen or dry compressed air		110 psi for 33,600 TOGW 120 psi for 35,100 TOGW 123 psi for 36,000 TOGW			A/C 1-60 and 114 not having ASC 69 A/C 1-60 and 114 having ASC 69 and A/C 61-162 A/C 1-162, 322 and 323 having ASC 175 and A/C 133-200.
8	Battery (2)	Distilled Water					Aft lower outboard each engine Nacelle
9	Engine Oil Tank (2)	Oil	(See Note F)	US Gal 3.98 each tank	Imp Gal 3.31 each tank	Metric Liters 15.06	At top of each engine
10	Hydraulic Reservior	Hydraulic Fluid	(See Note E)	US Gal 4.9	Imp Gal 4.08	Metric Liters 18.55	Aft of Main Door
11	Nose Wheel Tire (2)	Air		50 psi for all TOGW			
12	Nose Landing Gear Shock Strut	Hydraulic fluid dry nitrogen	(See Note E)	US Gal 0.37	Imp Gal 0.31	Metric Liters 1.40	Top of strut - see instruction plate on nose wheel drag brace fairing
13	Brake Accumulator	Nitrogen or dry compressed air		800 psi ± 25 psi			Nose wheel well, left side
14	Oxygen Cylinder	Oxygen	Federal Specification BB-0-925 Grade A Type 1	1800 psi			In radome compartment
15	Emergency Air Bottle	Nitrogen or dry compressed air		1900-2000 psi			Nose wheel well, right side
16	Boot Strap Turbine	Oil	(See Note F)	524 CC			Inside tail compartment

NOTE:

A. Aircraft 1-106 and 114 not having ASC 165.
B. Aircraft 107-200, 322 and 323 and Aircraft 1-106 and 114 having ASC 165.
C. Aircraft 1-200, 322 and 323, having ASC 125.
D. Mixing of fuels is permissible subject to any limitation in the Flight Manual.
E. Refer to combined air and fluid servicing (this section).
F. Refer to lubricating oils table.
G. Refer to approved fuels list.(this section)

Servicing Diagram
Figure 3.

"Reference to manufacturers in this Tech. Pub. are for identification purposes only and are neither specified nor furnished as a source for obtaining such parts."

2. GI Lubricating Oils Tables

NOTE: This list will only contain those oils that are approved for GI oil serviced equipment and are common between all equipment

NOTE: YES = Approved Oils, NO = Not Approved Oils.

CAUTION: THE USE OF ANY UNAPPROVED OIL SHOULD BE USED ONLY WITH AUTHORIZATION OF THE EQUIPMENT MANUFACTURER. REFER TO VENDORS MANUAL FOR COMPLETE LIST OF APPROVED LUBRICATING OILS.

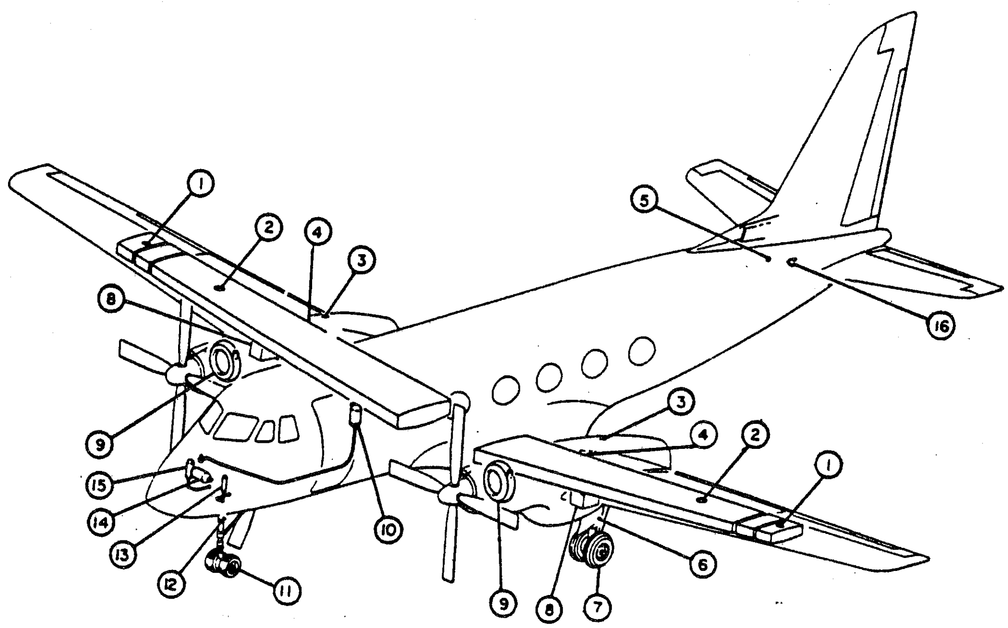
OILS ARE LISTED UNDER A MILITARY SPECIFICATION; HOWEVER, ONLY BRAND NAMES SPECIFICALLY AUTHORIZED SHOULD BE USED.

MIXING OF BRANDS AND/OR SPECIFICATIONS SHOULD BE DONE IN ACCORDANCE WITH MANUFACTURER'S RECOMMENDATIONS.

Table 1 MIL-L-7808 (3 Centistoke Oils).

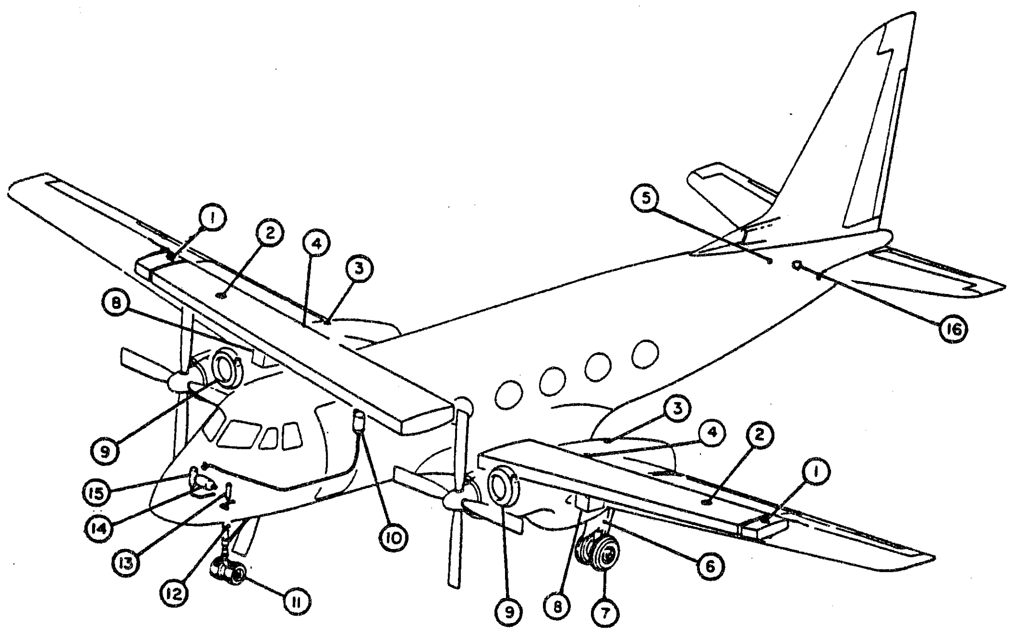
Manufacturer/Oil	Engine/Prop	Gearbox	APU	Bootstrap
Aeroshell Turbine Oil 750	YES	YES	YES	YES
Castrol 98	YES	YES	YES	YES

NOTE: For index number callout, see Figure 3.



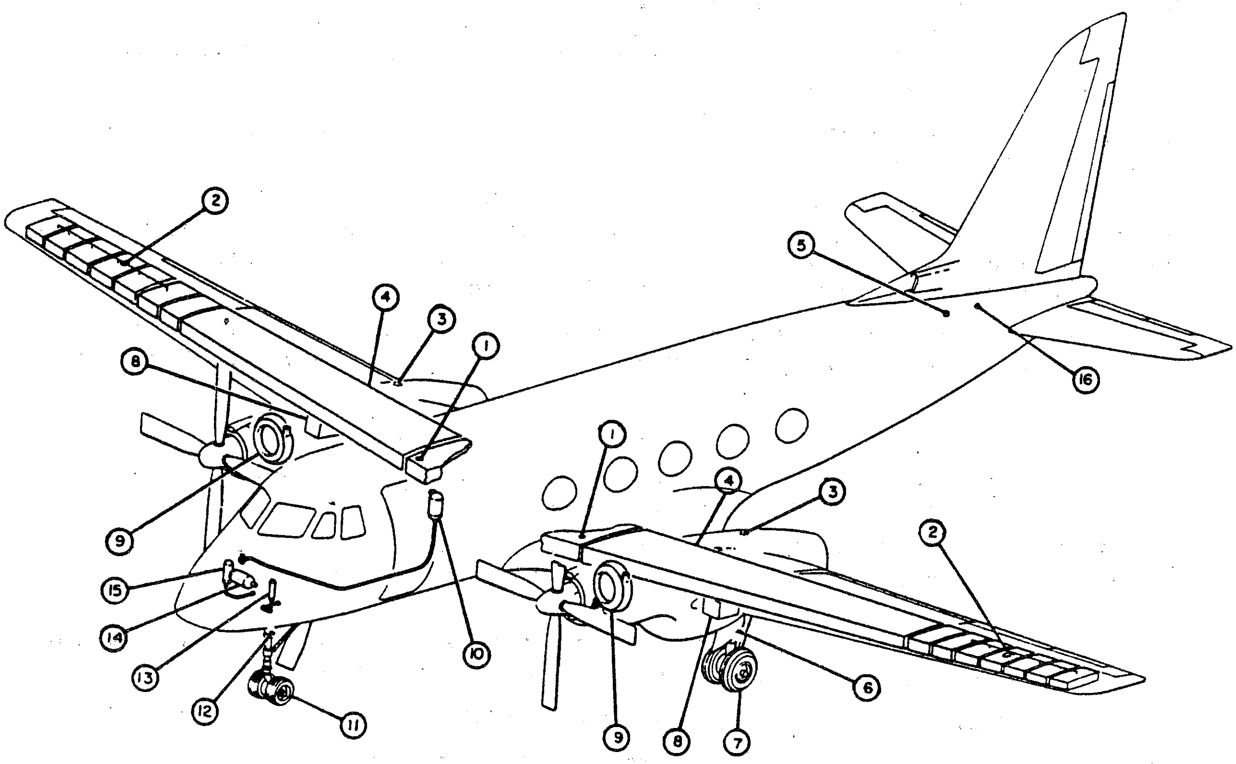
Aircraft 1-106 and 114 not having ASC 165
Figure 4.

NOTE: For index number callout, see Figure 3.



Aircraft 1-106 and 114 having ASC 165 and Aircraft 107-200.
Figure 5.

NOTE: For index number callout, see Figure 3.



Aircraft Having ASC 125.
Figure 6.

3. Lubrication Schedule — Special Service Notes on Lubrication

CAUTION: SYNTHETIC COMPOUNDS SUCH AS THOSE FOUND IN AIRCRAFT OILS AND GREASES CONTAIN ELEMENTS WHICH CAN SOFTEN PAINT, NATURAL RUBBER NEOPRENE, AND SOME ELECTRICAL INSULATORS. IF THIS TYPE LUBRICANT IS SPILLED ON ANY OF THESE MATERIALS, WIPE IT OFF THOROUGHLY WITH A CLEAN CLOTH. DO NOT MIX DIFFERENT TYPE GREASES. IF PREVIOUSLY USED GREASE IS NO LONGER AVAILABLE, COMPLETELY CLEAN AND RELUBE UNIT WITH ANOTHER AUTHORIZED GREASE.

1. All exposed lubrication points must be serviced immediately after the aircraft is washed.

   NOTE: MIL-G-81322 may be used in place of MIL-G-23827 (MIL-G-21164). If the aircraft is operated in extremely cold climates, we recommend that MIL-G-23827 (MIL-G-21164) Low/High temperature grease be used. This grease has lower temperature range (−100°F to +250°F) than the MIL-G-81322 (−65°F to +350°F).

2. Specialized lubrication requirements of instruments, electronic equipment, and engine accessories are not covered in the lubrication chart.

3. Remove all foreign matter from joints, fittings, or bearing surfaces immediately before application of lubricant. Use a clean cloth saturated with a cleaning solvent. Apply lubricant sparingly to prevent accumulation of contaminants. When applying lubricants through pressure type fittings, ensure that the lubricant has come out around bushings or bearing. Wipe off excess.

   NOTE: When performing area inspections, check for painted over fittings. Remove all paint from fitting and ensure fitting will accept the recommended lubricant. Replace fittings that will not accept lubricant.

4. Pressure gun with flush adapter must be held perpendicular to surface of the fitting.

5. PIANO TYPE HINGES - Following lubrication, care should be taken to remove all excess lubricant, since dirt and grit will accumulate on it and cause undue wear.

6. CONTROL SYSTEM BEARINGS - The control system bearings used in this aircraft are sealed and do not require lubrication except as noted in this lubrication schedule. Keep all control system bearings wiped clean.

7. LANDING GEAR SHOCK STRUTS / ACTUATOR CYLINDERS - Wipe all exposed hydraulic actuator piston surfaces daily with a clean cloth moistened with phosphate ester type IV fluid.

8. ELEVATOR TORQUE TUBES - A light coating of grease MIL-G-81322 should be applied every 300 hours.

9. CONTROL CABLE SEALS - Lubricate cable and seal with Molybdenum Disulfide MIL-G-21164

10. ENGINE CONTROLS - Apply grease MIL-G-23827 at two lubrication fittings on engine control box and one lubrication fitting on right side of engine controls crossover shaft.

11a. HYDRAULIC PUMP - DC GEN. - Remove from the drive pad every 150 hours and apply grease Aeroshell 17, Esso Andok 260 or Marfak 3HD to drive shaft spines. On gearbox part number PTG 14-30 and PTG 14-31, the gearbox end of the hydraulic pump spline is not to be lubricated with grease, as it has a wet spline.

11b. HYDRAULIC PUMP - On both types of hydraulic pumps, the hydraulic pump end of the splined coupling is to be lubricated every 600 hours with Plastilube #3 concurrently with the spline wear inspection.

12. ACCESSORY GEARBOX DRIVE SHAFT (OR) - Marfak 3HD, Aeroshell 5 or Andok 260 grease.

    CAUTION: DO NOT ALLOW OIL OR LUBRICANT TO RUN DOWN THE BALL SCREW INTO THE GEARBOX OF THE FLAP ACTUATOR.

13. WING FLAPS (L/R) - On aircraft having ASC 150, position flap actuator track rollers to coincide with 0.250 inch diameter holes located in track, by setting auxiliary hydraulic pump switch to ON and moving the emergency flap control handle to the down position.

14. SEAT TRACK - Clean all dirt and grit from tracks before lubrication.

15. WINDSHIELD WIPER SHAFTS (L/R) - Lubricate windshield wiper shafts every three months with anticorrosive solution conforming to Specification MIL-C-81309 (LPS3). Alternates: LPS1, LPS2, CRC2-26 or CRC3-36.

16. GEARBOX SUPPORT LINKS - Lubricate gearbox support links (two places per gearbox) with MIL-G-23827. If aircraft is equipped with support links per ASC 201, this lubrication is not required.

17. BOOTSTRAP SERVICING - Check bootstrap oil level every 50 hours, replenish as required. Drain and replace oil every 200 hours (more frequently in humid climates). Use only those oils specifically approved by AiResearch by brand name and number.

18. MAIN LANDING GEAR DRAG BRACE (L/R) - Lubricate specified drag brace surfaces with MIL-G-23827 in conjunction with inspection/lubrication per CB 262 and amendments.

19. SPILL VALVE - Lubricate spill valve shaft with anticorrosion solution conforming to specification MIL-C-81309 during area inspection.(See Chapter 5 for requirements.

20. NOSE LANDING GEAR LATCH SPRING CLEVIS AND BUNGEES - Lubricate sliding surfaces of latch spring clevis and compression link bungees with a light coating of Silicone oil SF8150 or SF96-50 during DOWN and LOCK check per CB 238B at each operational test of the landing gear system when aircraft is on jacks or 600 landings/12 months.

21. MAIN GEAR TIMER VALVE BUNGEE LINKAGE (OR) - Lubricate mating surfaces of bungee, link and bellcrank with a coating of MIL-G-23827 before installation of hardware after inspection. Lubrication is an annual requirement.

22. MAIN GEAR AND NOSE GEAR BEARINGS

    CAUTION: DO NOT SPIN BEARINGS WHEN DRYING WITH COMPRESSED AIR. WASH BEARING SEALS IN DENATURED ALCOHOL AND DRY WITH A CLEAN, SOFT CLOTH.

    DO NOT HANDLE BEARING COMPONENTS WITH BARE HANDS. USE A CLEAN, LINT-FREE CLOTH. DO NOT USE A DIFFERENT TYPE GREASE IN WHEELS ON THE SAME AIRCRAFT. WHEN GREASE TYPE IS CHANGED, REGREASE ALL WHEEL BEARINGS.

    A. Clean and repack wheel bearings at tire change. Clean wheel bearings with dry-cleaning solvent and dry thoroughly using dry, filtered compressed air.

    B. Immediately after drying, inspect the cone and bearings for wear or damage, and replace if necessary.

    C. Immediately after inspection, repack bearings with clean bearing grease, MIL-G-81322 and lubricate felt seals with MIL-L-7870 before installation.

23. NOSE AND MAIN GEAR DOOR HINGES, RODS AND MLG DOOR CONTROL SHAFT - Apply oil, MIL-L-7870 on all hinges and rods containing bushings that are not called out on illustration.

24. DOOR HINGES AND LATCHES - Lubricate every 150 hours with oil, MIL-L-7870, wipe off all excess.

25. ROD END BEARINGS — Use special fitting adapter to lubricate rod end bearings with flush lubricator fittings P/N MS24203 or equivalent.

26. Flaked or powdered graphite must not be used as a dry lubricant in this aircraft, because of inherent corrosion problem involved. Preferred lubricant is Molybdenum Disulfide grease, MIL-G-21164.

    NOTE: Molybdenum Disulfide (MO_2) comes as both a powder (MIL-M-7866) and a grease (MIL-G-21164). The powdered form should not be used in the field since it does not give adequate corrosion protection. Baked on dry film lubricants of Molybdenum Disulfide have proved satisfactory, but this requires special processing and results in the lubricant being "bonded" to the material. Mixtures of this powder with oils or greases should not be done in field applications where performance data have not been established. For field application, use Molybdenum Disulfide grease (MIL-G-21164) for a good corrosion protection.

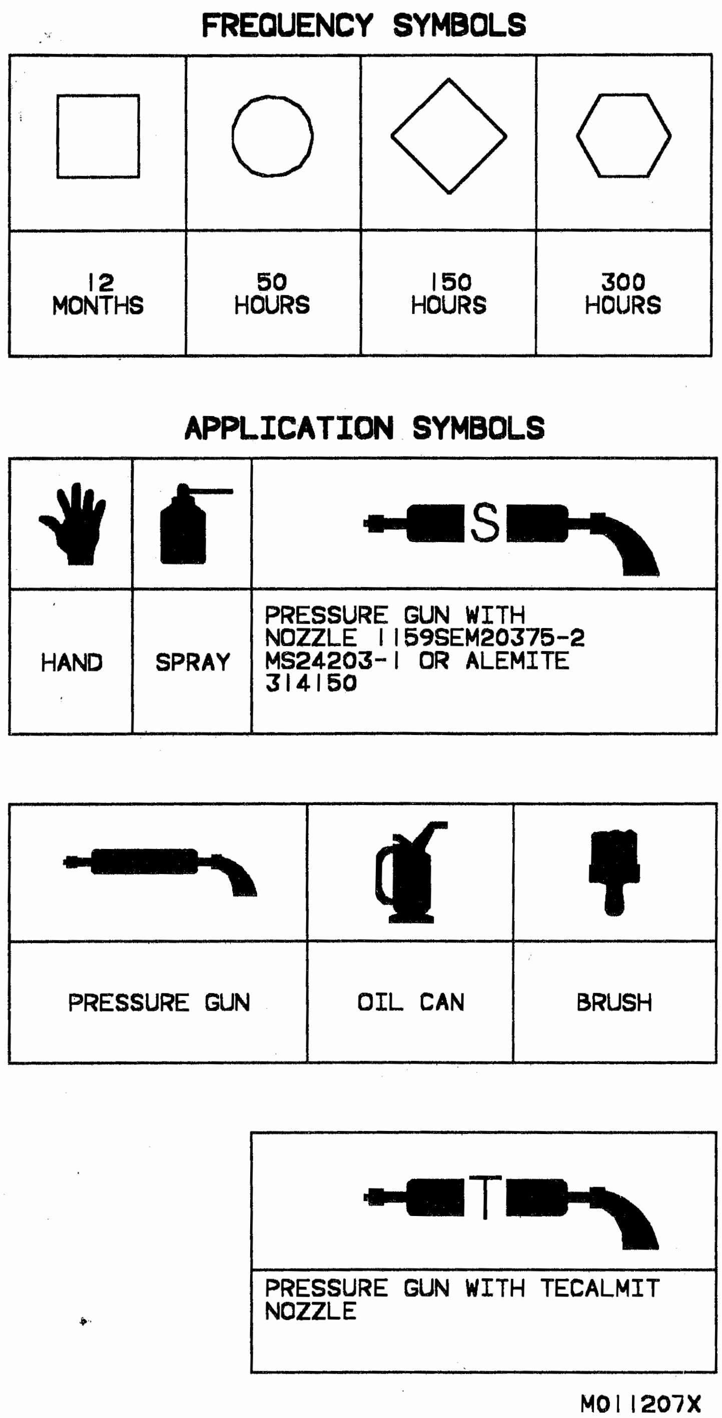

FREQUENCY SYMBOLS
12 MONTHS
50 HOURS
150 HOURS
300 HOURS
APPLICATION SYMBOLS
S
HAND
SPRAY
PRESSURE GUN WITH NOZZLE 1159SEM20375-2 MS24203-1 OR ALEMITE 314150
PRESSURE GUN
OIL CAN
BRUSH
T
PRESSURE GUN WITH TECALMIT NOZZLE
M011207X

Figure 7.

TABLE OF LUBRICANTS

IDENTIFICATION LETTER	BASE SPECIFICATION	TYPE OF LUBRICATION
GG	MIL-G-7178	GRAPHITE GREASE
GMD	MIL-G-21164	MOLYBDENUM DISULFIDE LOW/HIGH TEMPERATURE
GIA	MIL-G-23827	GREASE LOW/HIGH TEMPERATURE
GH	MIL-G-81322	GREASE HIGH TEMPERATURE, WATER RESISTANT
HO	SKYDROL LD-4 HYJET IV, HYJET IVA OR SKYDROL 500B-4	*OIL HYDRAULIC, ESTER BASE
MMD	MIL-M-7866	*MOLYKOTE MOLYBDENUM DISULFIDE
HILO	MS NO. 1	GREASE
OGP	MIL-L-7870	OIL GENERAL PURPOSE, LOW TEMPERATURE
OHA	MIL-H-5606 or MIL-H-83282B	OIL, HYDRAULIC, MINERAL BASE

*SEE SPECIAL SERVICE NOTES

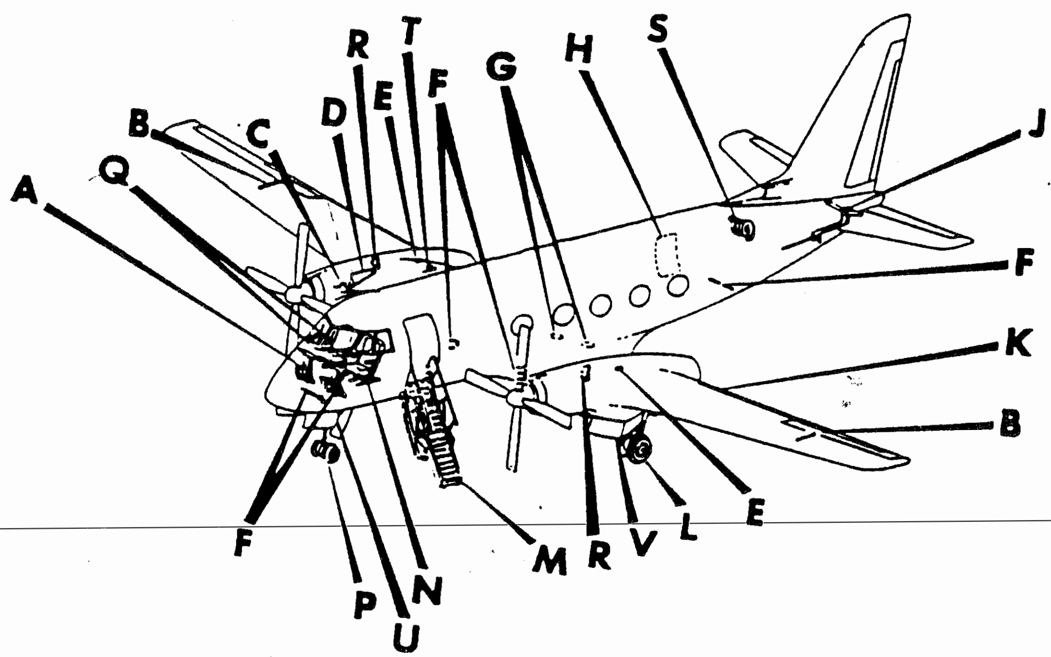
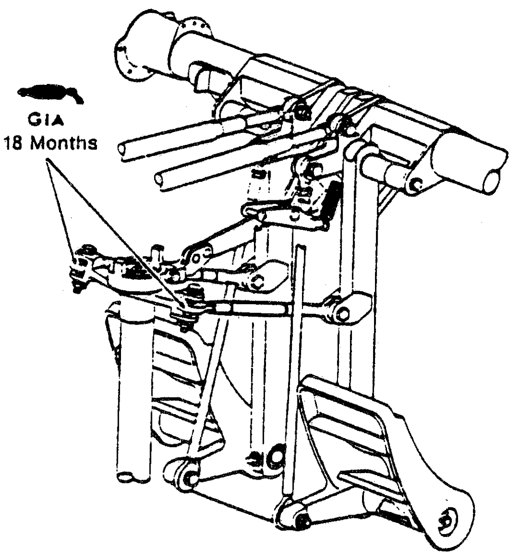
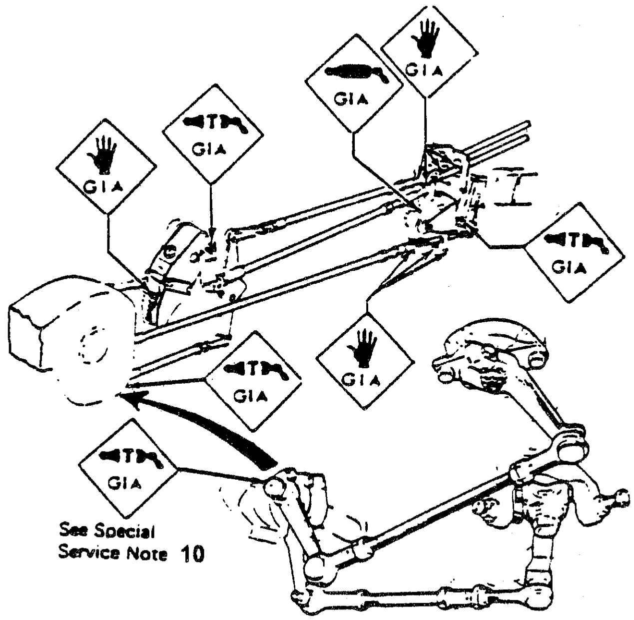


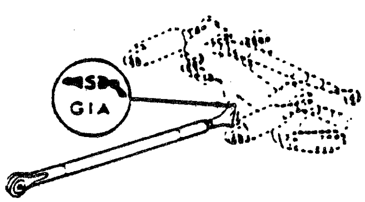
Figure 8.



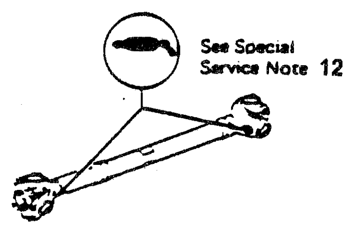
A RUDDER CONTROLS
(Left and Right)



C ENGINE CONTROLS
(Left and Right)

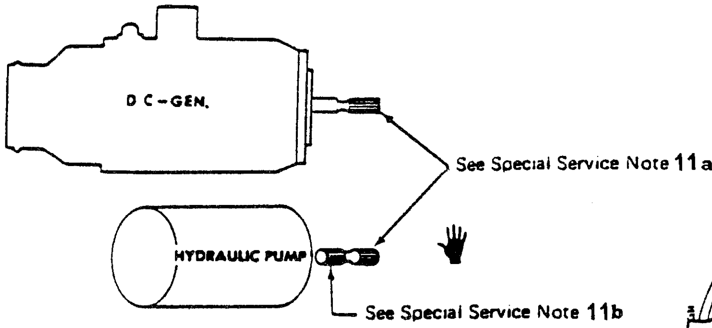


B AILERON PUSH RODS
(Left and Right)

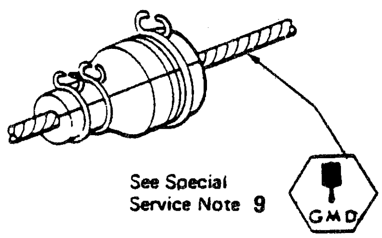


D ACCESSORY GEAR
BOX DRIVE SHAFT
(Left and Right)

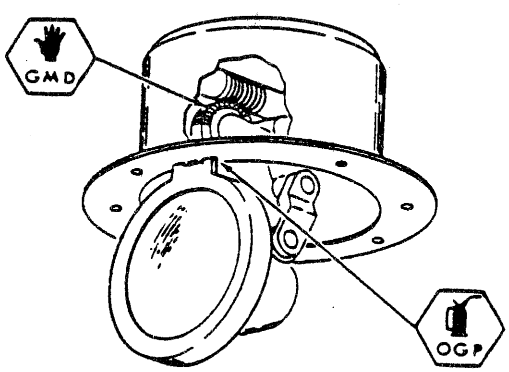
Figure 9.



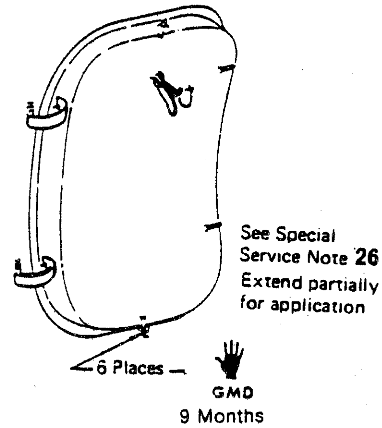
E ENGINE DRIVEN HYDRAULIC PUMP AND GENERATOR



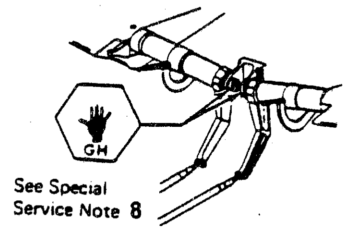
F PRESSURE SEALS



G RETRACTABLE LANDING LIGHT

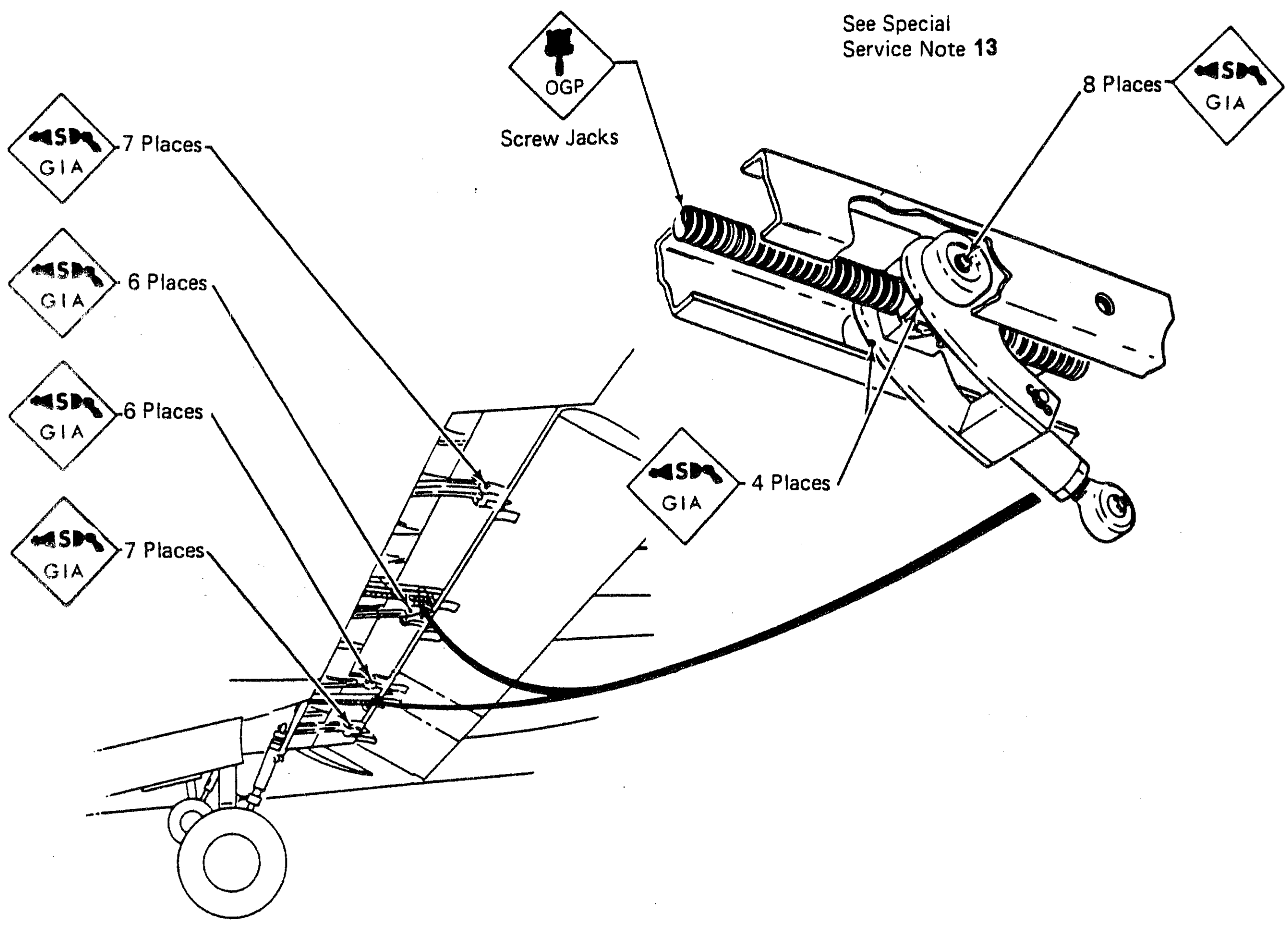


H BAGGAGE DOOR



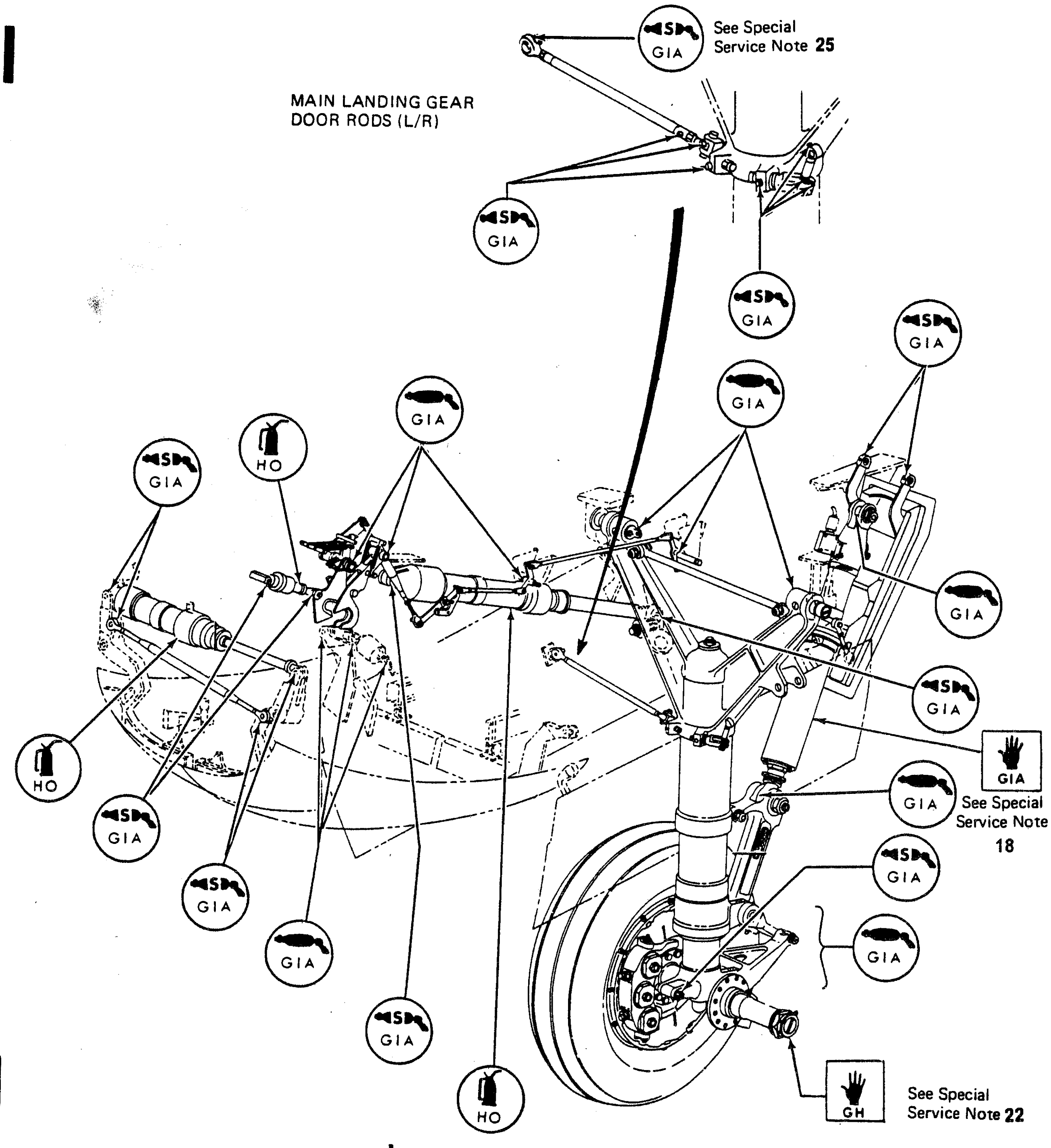
J ELEVATOR TORQUE TUBES

Figure 10.



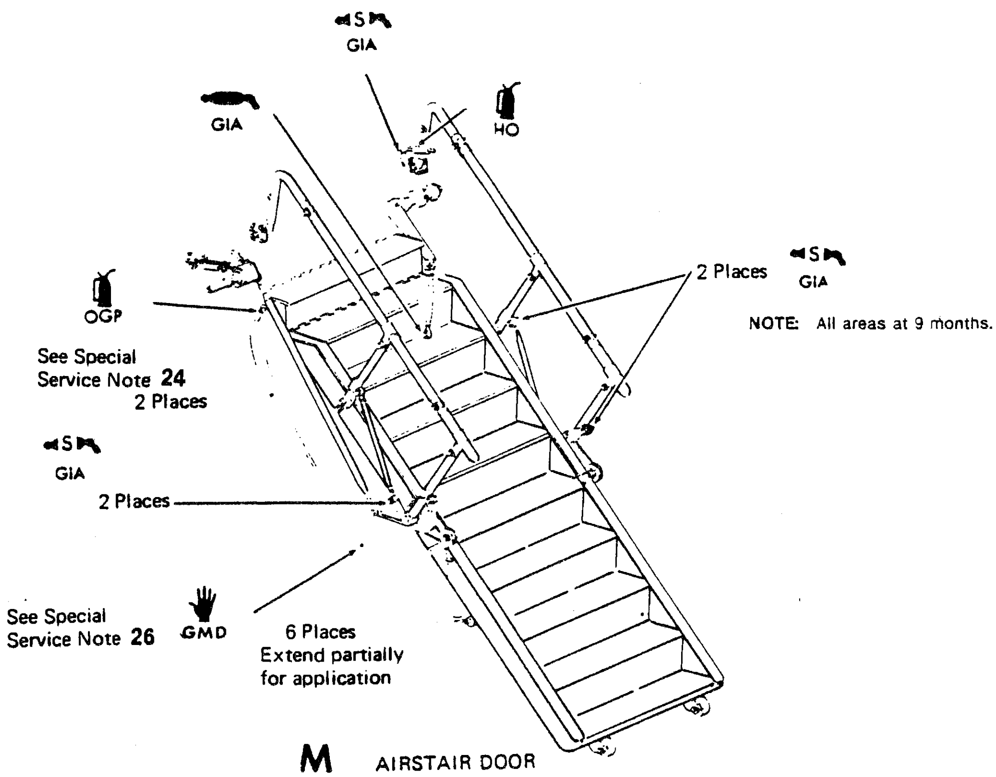
K FLAP ACTUATOR TRACK ROLLERS

Figure 11.

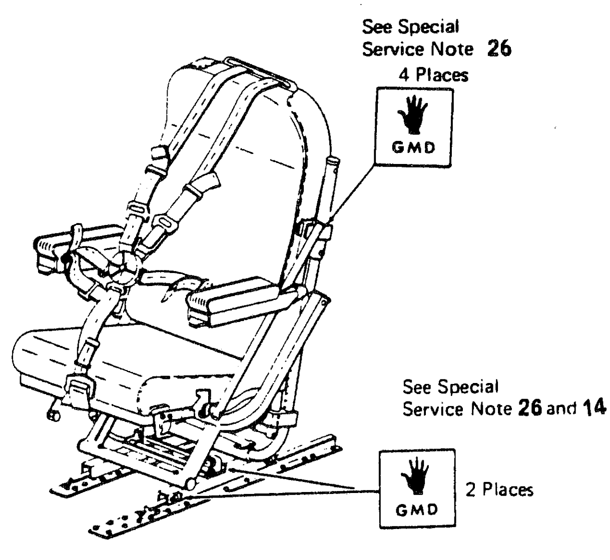


MAIN LANDING GEAR (L/R)
Left Wheel Omitted for Clarity

Figure 12.

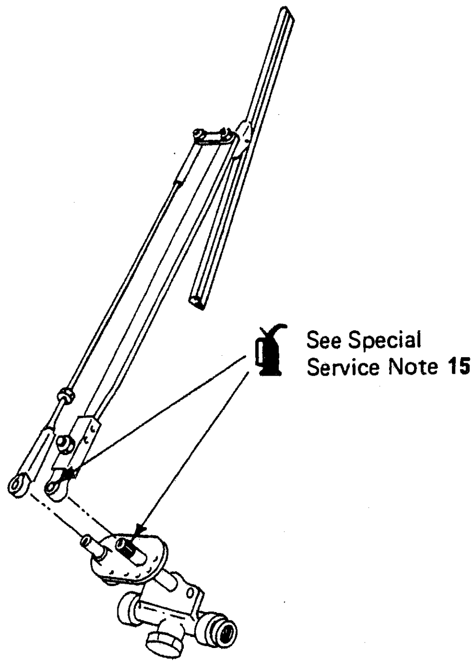


M AIRSTAIR DOOR

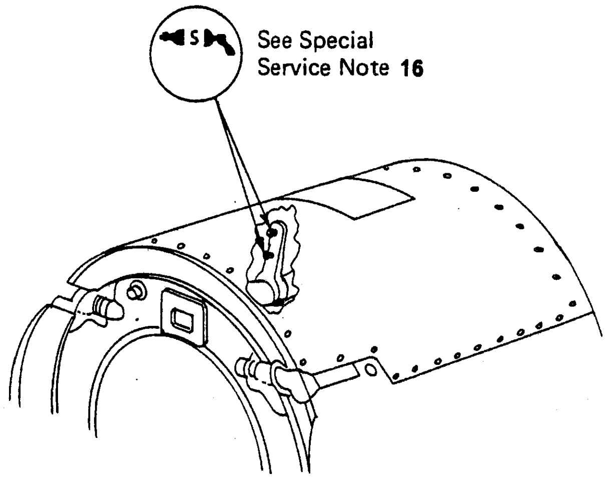


N PILOT'S and CO PILOT'S SEATS

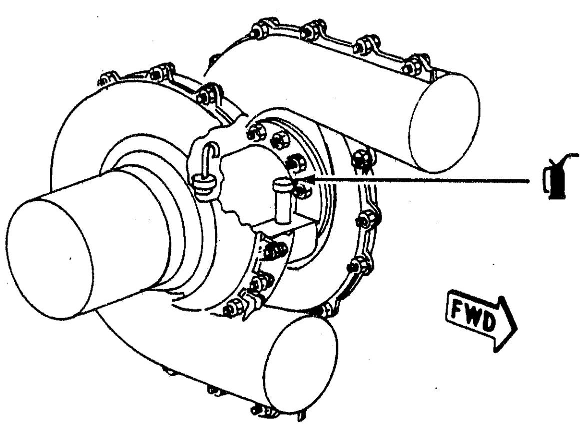
Figure 13.



Q WINDSHIELD WIPER ACTUATOR
(LEFT AND RIGHT)

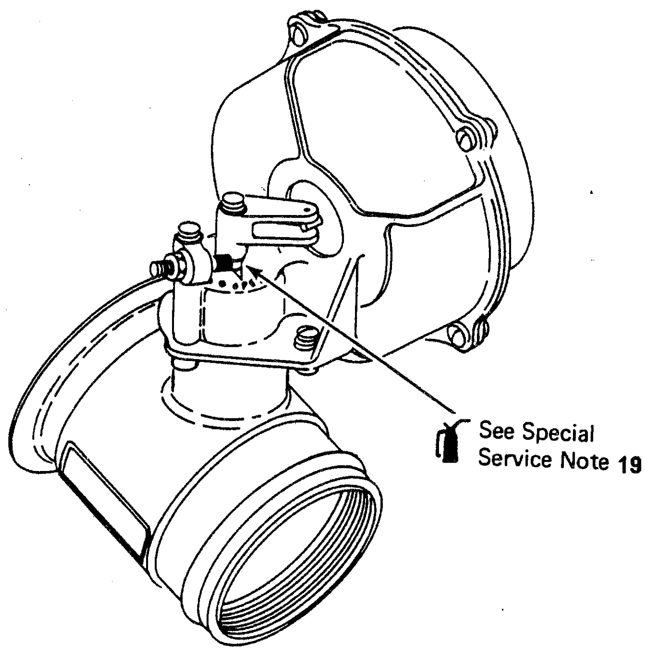


R GEARBOX SUPPORT LINKS
(LEFT AND RIGHT)



See Special Service Note 17

S COOLING TURBINE
(BOOTSTRAP UNIT)



T SPILL VALVE
(RH SIDE ONLY)

Figure 14.

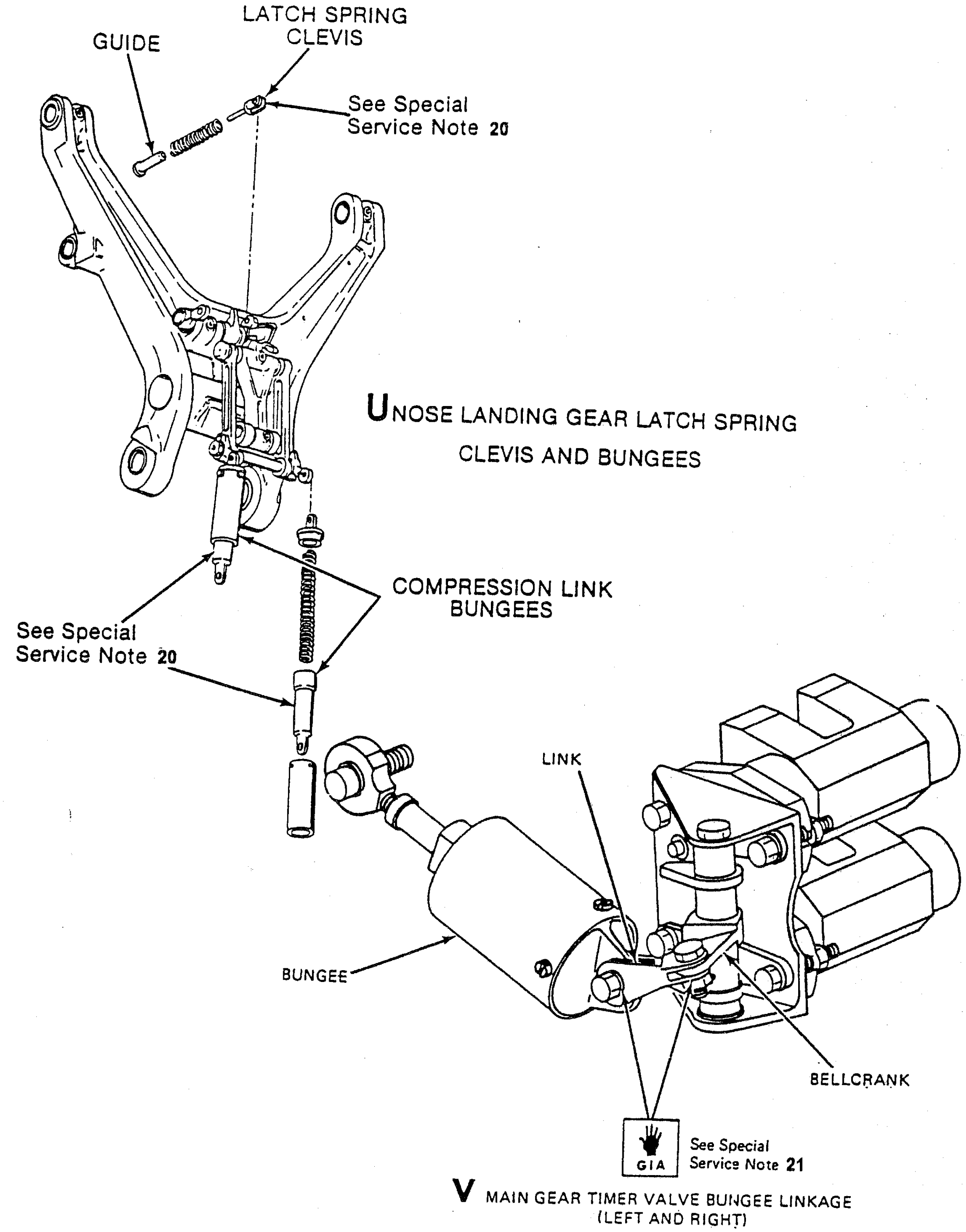
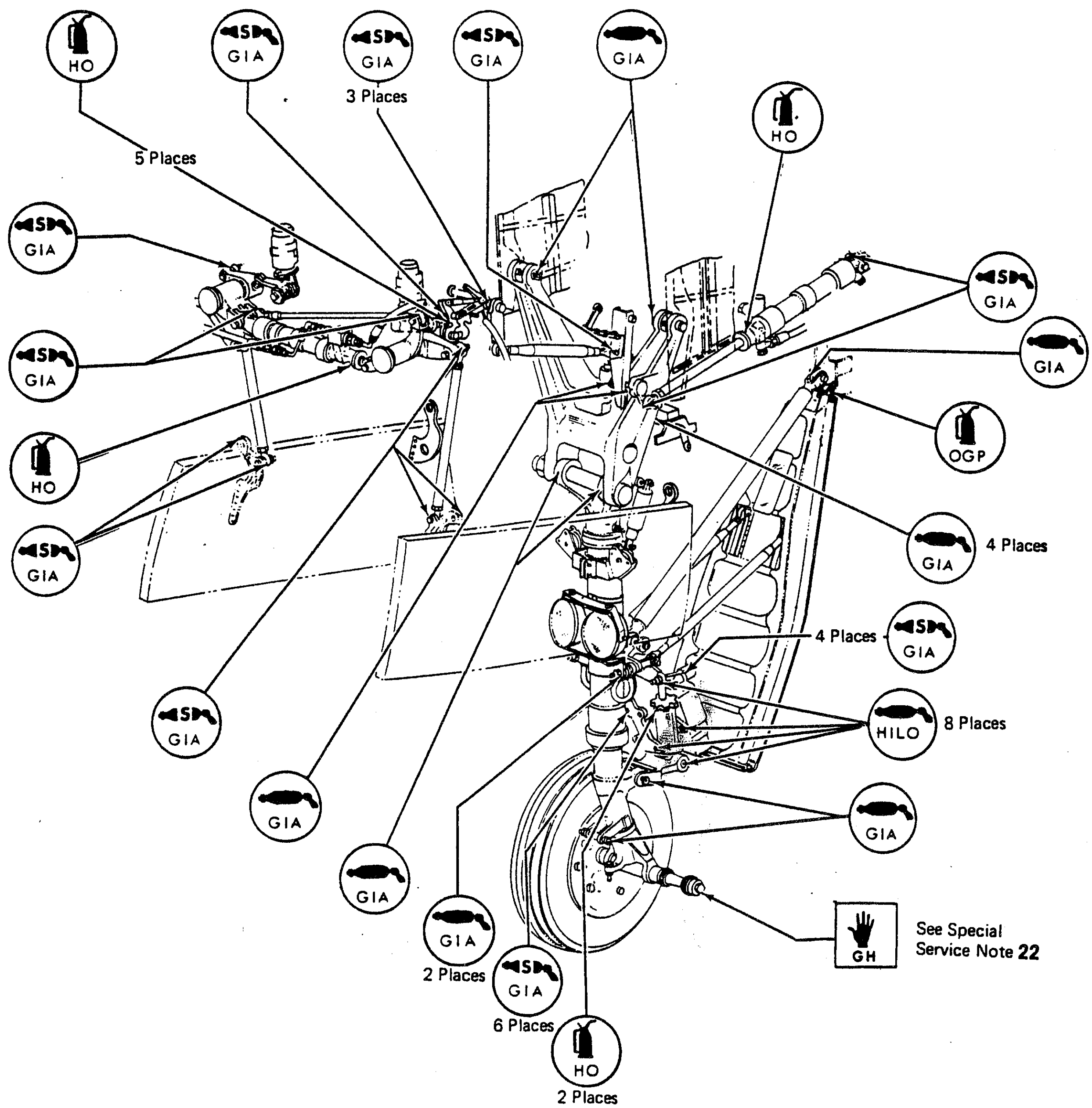

LATCH SPRING CLEVIS
GUIDE
See Special Service Note 20
U NOSE LANDING GEAR LATCH SPRING CLEVIS AND BUNGEES
COMPRESSION LINK BUNGEES
See Special Service Note 20
LINK
BUNGEE
BELLCRANK
GIA
See Special Service Note 21
V MAIN GEAR TIMER VALVE BUNGEE LINKAGE (LEFT AND RIGHT)

Figure 15.


HO
5 Places
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Service Note 22
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Figure 16.

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SERVICING — MAINTENANCE PRACTICES

1. **General**

NOTE: For certain servicing and pre-flight inspections, it is necessary to open the wheel well (clamshell) doors. The ground servicing control valve is located on the right side, aft end of the nosewheel well. The control lever has two positions, forward marked DOOR OPEN position and aft marked FLIGHT POSITION. The lever is spring loaded aft and locked in this position by a spring loaded knurled knobbed pin. To open the doors the knurled knob is pulled, unlocking the lever. The lever is then moved forward and is locked in this position with a separate ground lock pin with red streamer (part number 159GT1007 - Ground Safety Lock). Main dc bus power and the auxiliary hydraulic pump are utilized in this procedure.

2. **Clamshell Door Opening Procedure**

A. To open wheel well (clamshell) doors from outside aircraft utilizing the outside battery switch proceed as follows:

(1) Hook up left battery (if not connected).

(2) Open outside battery switch door (outboard side left nacelle).

(3) Place outside battery switch ON.

(4) In nosewheel, well, rotate ground servicing valve lever to door OPEN position and secure lever in place using ground safety lock pin and red streamer flag, Gulfstream Aerospace Tool part number 159GT1007.

(5) If airstair door is open, set outside door switch (on nosewheel junction box-nosewheel well) to OPEN position and hold until all clamshell doors are fully open. Auxiliary Hydraulic Pump will operate. Release switch to OFF position when doors are fully open.

(6) If airstair door is closed, unlatch and unlock airstair door using outside controls. (Do not open door). Set OUTSIDE DOOR SWITCH (Nosewheel well junction box - nosewheel well) to CLOSE position switch will stay in this position. Auxiliary Hydraulic Pump will energize and clamshell doors will open. When doors are fully open, return OUTSIDE DOOR SWITCH to OFF. Lock and latch airstair door.

(7) Install the following ground safety struts to prevent inadvertent closing of doors and possible injury to personnel: Part number 159GT1011 (1 required) nose gear doors and part number 159GT1014 (2 required) main gear doors.

(8) Turn off OUTSIDE BATTERY SWITCH and close access door.

(9) Disconnect left battery if desired.

3. **Clamshell Door Opening Procedure (Utilizing Auxiliary Hydraulic Pump Switches)**

NOTE: Auxiliary Hydraulic system pressure is required to open the doors for ground service. During this operation the normal system door control valve was not disturbed. Its last position with gear down ported the main system pressure to the close side of the wheel well door cylinder. Applying main hydraulic system pressure will, therefore, close the doors.

A. If power is available on aircraft and cockpit is accessible, the cockpit AUXILIARY HYDRAULIC PUMP SWITCHES (Pilots and copilots) may be used in lieu of the outside door switch as follows:

(1) Place GROUND SERVICING VALVE LEVER in nosewheel to DOOR OPEN position and lock lever with Gulfstream Aerospace Tool part number 159GT1007.

(2) Place pilots or copilots AUXILIARY HYDRAULIC PUMP SWITCH in cockpit to ON until doors are fully open.

(3) Set AUXILIARY HYDRAULIC PUMP SWITCH to OFF.

(4) Install nose and main gear door safety struts (part numbers 159GT1011 and 159GT1014).

4. Wheel Well Door Closing Procedure

A. To close wheel well doors, on aircraft not having ASC 109, proceed as follows:

(1) Remove wheel well door ground safety struts.

WARNING: CLEAR ALL PERSONNEL FROM WHEEL WELLS, WHEN VALVE IS RETURNED TO FLIGHT POSITION, BEFORE APPLYING MAIN SYSTEM HYDRAULIC PRESSURE.

(2) Remove lockpin P/N 159GT1007 from ground service control valve, and return valve to FLIGHT position.

(3) Apply main system pressure by either of the following:

- Rotate either left of right propeller by hand in direction of rotation until all wheel doors are fully closed.
- Cranking cycle of either left or right engine.
- Starting cycle of either left or right engine.
- Attaching an external hydraulic rig to the attachment fittings on the right engine hydraulic pump.

B. To close wheel well doors, on aircraft having ASC 109, proceed as follows:

(1) Rotate auxiliary to main selector valve from FLIGHT to MAIN ON position and lock in place using lockpin P/N 159GT1007.

(2) Energize auxiliary hydraulic pump by following the same proceeds as outlined in door opening procedure.

(3) Remove lockpin P/N 159GT1007 at completion of closing doors and ensure lever has returned to FLIGHT position.

5. Aircraft Washing

A. Pressure washing of aircraft is not recommended due to the adverse effects on aircraft components. The high velocity water and cleaning agents can force dirt, contamination and moisture into bearings, bushed joints, actuator seals, electrical components, faying surfaces and structural joints resulting in increase maintenance costs and unserviceability.

B. The use of manual washing methods (scrubbing with a brush) on the aircraft is recommended. Cleaning agent should be pH neutral (7) or slightly alkaline to avoid a corrosive potential (car wash type cleaners are suggested).

C. Water directed at close range by hoses with even modest pressure can remove joint lubricants and damage aircraft components. To avoid problems, water pressure should be minimized. Avoid direct rinsing of all bearings, bushings, electrical connectors and electrical components. Exercise care around landing gear joints and bushings so that the protective cover of old grease is not removed. All exposed lubrication points must be serviced immediately after completion of wash.

D. Avoid directing spray into openings or onto surfaces adjacent to openings. Keep water nozzle at least 3 feet away, where possible, from the surface being rinsed. Position water stream at a direct angle contact of 45° or less to minimize impact pressure.

AIR SERVICING — MAINTENANCE PRACTICES

1. **Brake Accumulator**

CAUTION: WHEN CHECKING OR ADDING AIR TO THE ACCUMULATOR, ALL HYDRAULIC PRESSURE MUST BE RELIEVED BY ACTUATING THE PARK/EMER BRAKE HANDLE. FOLLOW PROCEDURE ON PLACARD IN THE NOSE WHEEL WELL.

A. Brake accumulator is located in nose wheel well left side. Pressure gage is on copilots instrument panel. Gage records pressure of nitrogen introduced into accumulator through air filler valve. Gage should read 800 ± 25 psi (preload).

B. If accumulator pressure reading is below 1425 psi service accumulator as follows:

NOTE: On Aircraft 1 - 87 and 114 not having ASC 137, accumulator pressure gage should indicated 1900 psi with auxiliary hydraulic pump operating. On Aircraft 88 - 200, 322 and 323 and Aircraft 1 - 87 and 114 having ASC 137 accumulator pressure gage should indicate 1425 to 1500 psi with the auxiliary pump operating.

C. Actuate parking brakes selector valve (T-handle) repeatedly until observed air pressure stabilizers. (800 ± 25 psi)

WARNING: ENSURE AIR-NITROGEN TRAILER IS EQUIPPED WITH A PRESSURE REGULATOR

D. Connect hose from air-nitrogen (100-3000 psi) trailer to accumulator air filler valve.

E. Open valve on air-nitrogen trailer and service accumulator with dry nitrogen.

F. When accumulator gage reads 800 ± 25 psi, close valve and disconnect nitrogen from accumulator air filler valve.

G. Recharge brake accumulator with hydraulic fluid by turning on auxiliary hydraulic pump.

2. **Landing Gear Emergency Bottle**

A. Landing gear emergency bottle filler valve and gage is located in nose wheel (right side looking forward). A preflight check is required. Normal charge is 1950 ± 50 psi.

B. To fill air bottle, proceed as follows:

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Remove cap from air filler valve in nose wheel well.

WARNING: ENSURE AIR-NITROGEN TRAILER IS EQUIPPED WITH A PRESSURE REGULATOR.

(2) Connect hose from air-nitrogen (100-3000 psi) trailer to filler valve.

(3) Loosen swivel nut on filler valve slowly, 2 1/2 turns and allow bottle to fill with nitrogen or dry compressed air to 1900-2000 psi.

(4) Tighten swivel nut on filler valve to a torque of 50-70 inch pounds.

(5) Turn off nitrogen supply, vent supply line to atmosphere and remove supply line.

(6) Install valve cap.

3. **Tire Servicing**

NOTE: Gulfstream Aerospace recommends dry nitrogen only be used when servicing aircraft tires.

Inflate nose and main tires to following pressures:

A. Nose wheel tires - 50 psi

B. Main wheel tires:
- 110 psi for 33,600 pounds takeoff gross weight (Aircraft 1 - 60 and 114 not having ASC 69)
- 120 psi for 35,100 pounds takeoff gross weight (Aircraft 1 - 60 114 having ASC 69 and 61 - 162)

- 123 psi for 36,000 pounds takeoff gross weight (Aircraft 1 - 162, 322 and 323 having ASC 175 and Aircraft 133 - 200).

COMBINED AIR AND FLUID SERVICING — MAINTENANCE PRACTICES

1. **Main and Nose Landing Gear Shock Struts — Servicing**

A. Clean Landing Gear Shock Struts.

(1) Clean shock strut piston polished surface and landing gear actuating cylinders exposed piston rod surfaces by wiping with soft clean cloth saturated with hydraulic fluid. Use same type as used in aircraft hydraulic system.

(2) In order to avoid physical damage to the landing gear components, it is essential the following precautions be exercised in selection and application of cleaning solutions.

- Methyl Ethyl Ketone (MEK) has been found a satisfactory cleaning solution. A number of commercially available cleaners and other products used as cleaners, particularly Varsol, Shell 640, Esso Turbo Oil 35 and Methyl Isolbutyl Ketone (MIBK) have a powerful softening effect on the type of rubber compound used on this aircraft.
- Careless application of cleaner must be avoided. All cleaners, including MEK, have an undesirable effect on rubber compounds. Cleaner must be applied sparingly, brush application is recommended, and immediately wiped off with a clean cloth. Wheel bearings and other lubricated parts must be protected against contact by the cleaner to prevent reduction of efficiency or total loss of lubricant. Tires and decelostat brake units must be covered during cleaning procedure. It is also recommended tires be placed on blocks as a precautionary measure against accidental spillage.

B. Main Gear Strut Filling Procedure. (Aircraft 1 - 162, 322 and 323 not having ASC 169 or ASC 175.)

(1) Jack aircraft until tires clear ground.

WARNING: DO NOT LOOSEN VALVE BODY WHILE UNIT IS DISCHARGING NITROGEN. SERIOUS INJURY OR LOSS OF LIFE MAY RESULT IF VALVE BODY FLIES OUT.

(2) Loosen hex swivel nut and allow nitrogen to discharge.

(3) When all nitrogen is discharged, remove filler valve.

(4) Fully compress main strut and fill to overflow with hydraulic fluid. Fill slowly to avoid foaming. Use same type as used in the aircraft hydraulic system.

(5) Manually extend the main gear strut and permit to stand for a few minutes. Stroke to full up position and again slowly add hydraulic fluid in this position. Repeat if necessary.

(6) Install and secure filler valve.

WARNING: USE DRY NITROGEN ONLY. NEVER USE OXYGEN OR AIR.

(7) Using dry-nitrogen (100-3000 psi), inflate strut to 381 psi with strut fully extended.

NOTE: If aircraft is not on jacks, nitrogen pressure in strut can be checked in accordance with instruction plate on main gear door. Using a high pressure gage (0-2000 psi), increase or decrease pressure so that it corresponds to dimension X.

C. Main Gear Strut Filling Procedure (Aircraft 1 - 162, 322 and 323 having ASC 169)

(1) Jack aircraft until tires clear ground.

WARNING: DO NOT LOOSEN VALVE BODY WHILE UNIT IS DISCHARGING NITROGEN. SERIOUS INJURY OR LOSS OR LIFE MAY RESULT IF VALVE BODY FLIES OUT.

(2) ~~Loosen hex swivel nut and allow nitrogen to discharge.~~

(3) When all nitrogen is discharged, remove filler valve.

(4) Fully compress main strut and fill to overflow with hydraulic fluid, fill slowly to avoid foaming. Extend and compress strut to ensure all voids within strut are filled with fluid when strut is fully compressed. Use same type fluid as used in the aircraft hydraulic system.

(5) Extend strut and add 3 1/2 ounces of hydraulic fluid.

(6) Install and secure filler valve.

WARNING: USE DRY NITROGEN ONLY. NEVER USE OXYGEN OR AIR.

(7) Using dry nitrogen (100-3000 psi), inflate strut to 258 psi with strut fully extended.

NOTE: Nitrogen pressure may be checked with gear on the ground. Use high pressure air gage (0-2000 psi). This filling procedure should be maintained in service in accordance with new instruction plate Bendix No. 2572584.

D. Main Gear Strut Filling Procedure. (Aircraft 1 - 162, having ASC 175 and Aircraft 163 - 200, 322 and 323)

(1) Jack aircraft until tires clear ground.

WARNING: DO NOT LOOSEN VALVE BODY WHILE UNIT IS DISCHARGING NITROGEN. SERIOUS INJURY OR LOSS OR LIFE MAY RESULT IF VALVE BODY FLIES OUT.

(2) Loosen hex swivel nut and allow nitrogen to discharge.

(3) When all nitrogen is discharged, remove filler valve.

(4) Fully compress main strut and fill to overflow with hydraulic fluid. Fill slowly to avoid foaming. Extend and compress strut to ensure all voids within strut are filled with fluid when strut is fully compressed. Use same type fluid as used in aircraft hydraulic system.

(5) Extend strut and add an extra 3 1/2 ounces of hydraulic fluid.

(6) Install and secure filler valve.

WARNING: USE DRY NITROGEN ONLY. NEVER USE OXYGEN OR AIR.

(7) Using dry nitrogen (100 - 3000 psi), inflate strut to 280 psi with strut fully extended.

NOTE: Nitrogen pressure may be checked with gear on the ground. Use high pressure air gage (0 - 2000 psi). This filling procedure should be maintained in service in accordance with instruction plate Bendix No. 2573148.

E. Nose Gear Strut Filling Procedure

(1) Jack up the aircraft until tires are clear of the ground.

(2) Remove filler valve as follows:

WARNING: DO NOT LOOSEN THE VALVE BODY WHILE THE UNIT IS DISCHARGING NITROGEN. SERIOUS INJURY OR LOSS OR LIFE MAY RESULT, IF VALVE BODY FLIES OUT.

(a) With a wrench, check to ensure top 3/4 inch hex swivel nut is tight (turn clockwise), then remove the valve cap.

(b) Back off hex swivel nut and allow nitrogen to exhaust itself.

(c) The amount the 3/4 hex swivel nut is loosened (up to maximum, 3 1/2 turns), will govern rate of discharge.

(d) When pressure is completely exhausted, remove filler valve assembly by turning hex stem assembly counterclockwise with a wrench.

(3) Fully compress nose gear strut and fill to overflow with hydraulic fluid. Fill slowly to avoid foaming. Use same type fluid as used in aircraft hydraulic system.

(4) Manually extend nose gear strut and permit it to stand for a few minutes. Stroke to full up position and again slowly add hydraulic fluid in this position. Repeat if necessary.

(5) Replace and secure nitrogen-oil filler valve.

(6) Using dry nitrogen only, inflate the nose gear strut as per instructions on plate attached to nose wheel drag brace fairing.

F. To Discharge or Check Strut Pressure

(1) With a wrench, check to ensure top 3/4 inch hex swivel nut is tight (turn clockwise), then remove the valve cap.

(2) To dissipate strut pressure, loosen hex swivel nut by turning counterclockwise a maximum of 3 1/2 turns.

(3) To check strut pressure, attach pressure gage to valve housing and check.

(4) Tighten 3/4 inch hex swivel nut, remove pressure gage, replace and tighten valve cap.

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FIRE EXTINGUISHER AGENT SERVICING — MAINTENANCE PRACTICES

1. Fire Extinguisher System — Servicing

NOTE: Fire extinguisher containers cannot be recharged in aircraft. They must be recharged with special equipment at an authorized agency equipped to perform this work, therefore, the only servicing which can be performed is checking the container pressure.

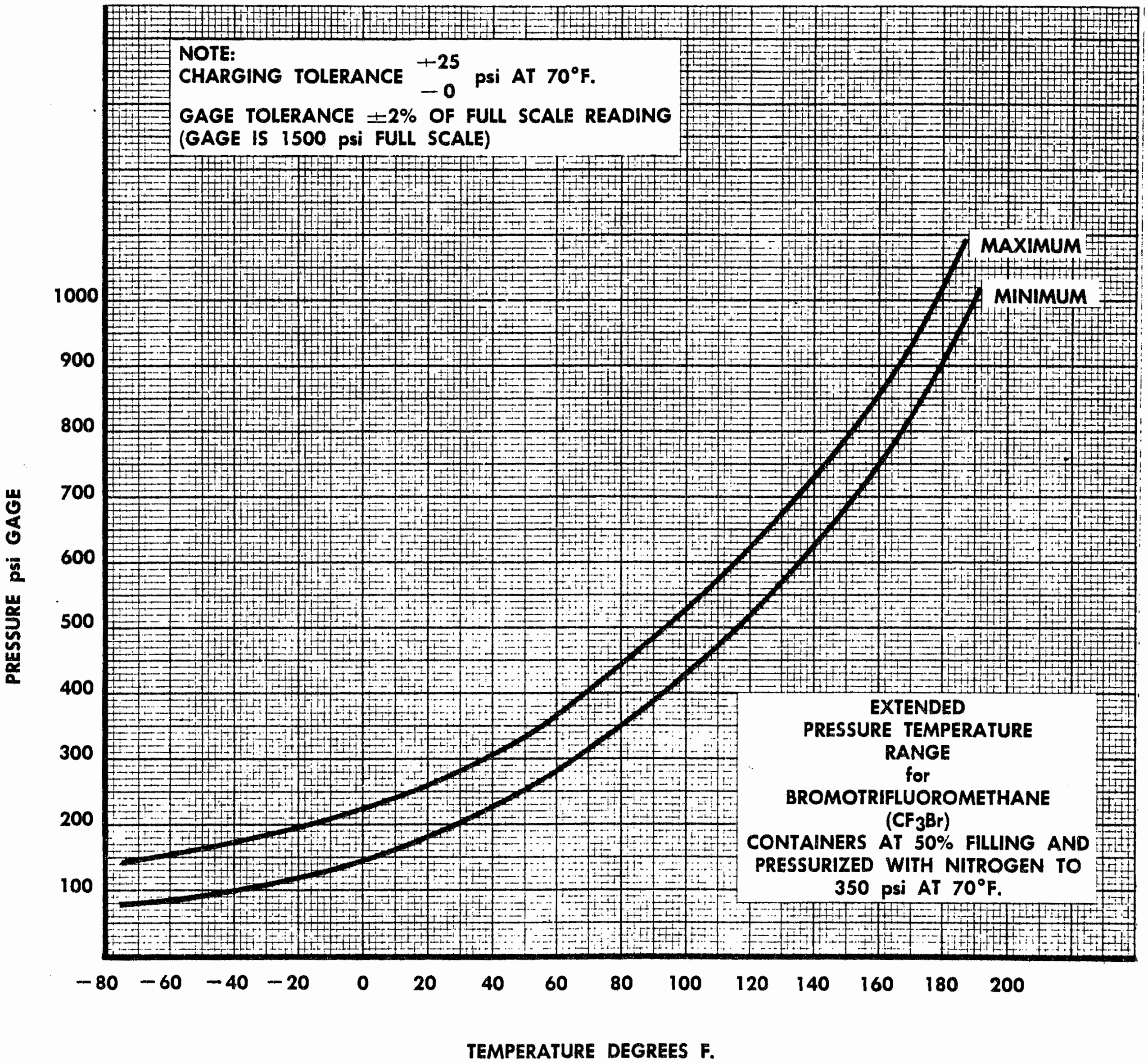
A. Check Container Pressure.

A fire extinguisher container pressure gage is located on fire extinguisher container in each engine nacelle and on fire extinguisher in tail compartment near the Auxiliary Power Unit (APU). On left engine, gage is visible on aft top inboard portion of nacelle and on right engine, gage is visible on aft top outboard portion of nacelle. Fire extinguisher container and gage for Auxiliary Power Unit (APU) is mounted on a bulkhead in tail section, right side forward of tail access door.

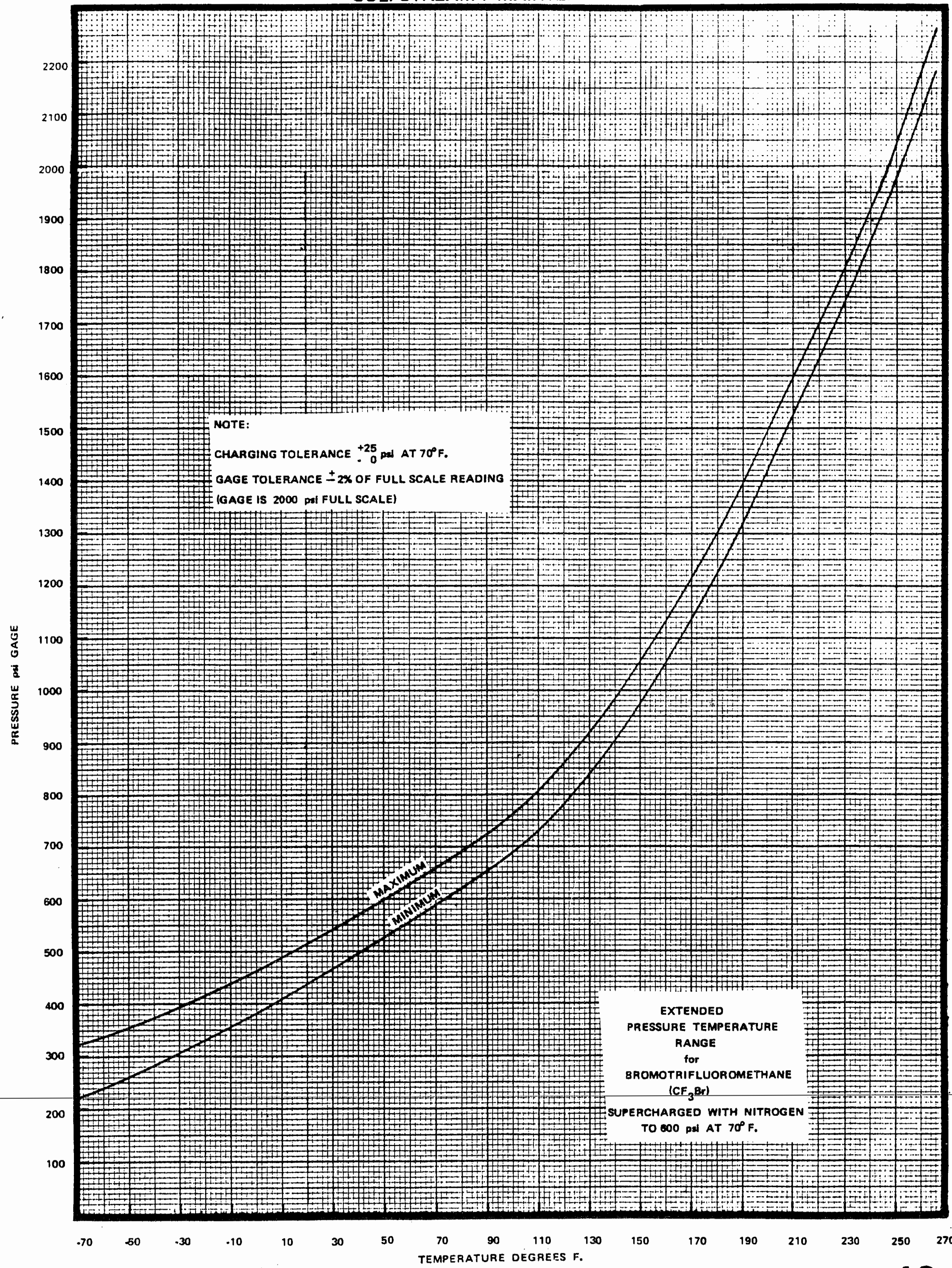
WARNING: BROMOTRIFLUOROMETHANE (CF_3BR) IS NONTOXIC AND IS CLASSED AS NON-POISONOUS. HOWEVER, IT CAN BE HARMFUL AND SHOULD BE HANDLED WITH CARE. DO NOT BREATHE THE VAPOR OR PERMIT THE LIQUID TO COME IN CONTACT WITH SKIN OR CLOTHING. IF LEAKAGE IN A CLOSED AREA IS SUSPECTED, THIS AREA SHOULD NOT BE ENTERED UNTIL ALL VAPOR HAS BEEN DISSIPATED AND THE AREA IS WELL VENTILATED. USE A HALIDE (PREST-O-LITE OR EQUIVALENT) TO ENSURE THAT AREA IS SAFE.

B. Check all fire extinguisher agent containers for proper pressure. See Figure 201 for APU fire extinguisher pressure and Figure 202 for engine fire extinguisher pressures. If pressure does not fall within prescribed limits, remove and replace container, see Chapter 26 for Fire Extinguisher — Maintenance Practices.

WARNING: FIRE EXTINGUISHERS ARE DISCHARGED BY ELECTRICALLY FIRED EXPLOSIVE CARTRIDGES. THESE CHARGES PROPEL A METAL SLUG THROUGH A DIAPHRAGM. FIRING OF THESE CARTRIDGES WITH BONNETS REMOVED IS HAZARDOUS. CARTRIDGES CAN BE FIRED BY VERY LOW VOLTAGES (1.5 VOLTS). USE EXTREME CAUTION. THERE ARE TWO CARTRIDGES IN EACH ENGINE FIRE EXTINGUISHER CONTAINER, AND ONE IN THE APU CONTAINER.



Engine Fire Extinguisher Extended Temperature Pressure Range
Figure 201.



APU Fire Extinguisher Extended Temperature Pressure Range
Figure 202.

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FUEL SERVICING — MAINTENANCE PRACTICES

1. Fuel Tanks — Servicing

(Refer to Section 12-0 and Rolls-Royce Maintenance Manual M-Da-7-G for approved fuels.)

A. Overwing Replenishing of Fuel Tanks (See Section 12-0).

CAUTION: BECAUSE OF LOCATION THE FUEL AND WATER/METHANOL FILLER CAPS, BE CAREFUL NOT TO INADVERTENTLY PUT FUEL INTO WATER/METHANOL TANKS. ENGINE WILL BE OVERTORQUED IF FUEL IS USED INSTEAD OF WATER/METHANOL.

B. Check quantity in tanks using cockpit gages.

CAUTION: IF TANKS NEED REFILLING, CARE SHOULD BE TAKEN NOT TO DRAG HOSE ACROSS WING DE-ICER BOOTS AS DAMAGE TO BOOTS COULD RESULT. IT IS RECOMMENDED HOSE BE BROUGHT OVER TRAILING EDGE OF WING RATHER THAN LEADING EDGE.

C. Ground fuel hose nozzle to aircraft structure through grounding jacks located outboard of filler caps.

D. Add sufficient fuel to fill tanks without overflowing.

E. Check and ensure filler caps are securely fastened.

WARNING: OVERFLOWING TANKS CREATE A SERIOUS FIRE HAZARD AND MUST BE AVOIDED. DAMAGE TO DE-ICER BOOTS COULD ALSO RESULT. SPILLED FUEL MUST BE WIPED UP IMMEDIATELY.

2. Fuel System — Servicing

NOTE: Jet fuels have a stronger affinity for water than Avgas. Settling time for water in jet fuel is five times as long as for water in Avgas. Following recommended ground procedures can reduce suspended or free water to 30 parts/million. Suspended water in a quantity of 40 parts/million is sufficient to ice a filter.

A. Corrective or Preventive Action - Ground Handling.

Acceptable water content for jet fuel is 30 parts/million. Because fuel with this water content is not available at all airports, use of a "GO-NO-GO" kit and the following procedures are recommended.

B. Water drain main tanks after each flight and prior to refueling. This clears tanks of any water condensed in flight.

C. Allow maximum possible time between refueling and takeoff. Water drain prior to takeoff.

D. Whenever possible, fuel at or below ambient temperature should be used.

E. Use a "GO-NO-GO" kit each time aircraft is refueled.

F. Drain engine fuel filter daily, and check for water content with a "GO-NO-GO" kit.

NOTE: Set tank boost pump switch to ON during draining, to prevent air from entering fuel system.

3. Defueling — Pumping thru Drain Valves in Main Landing Gear Wheel Well

WARNING: WHEN DEFUELING, HAVE FIRE FIGHTING EQUIPMENT AVAILABLE TO COMBAT PETROLEUM FIRES. DO NOT PERMIT SMOKING,OR USE OF ANY ELECTRICAL EQUIPMENT WHICH DOES NOT EMPLOY EXPLOSION PROOF FEATURES. ELECTRICALLY GROUND ALL EQUIPMENT AND AIRCRAFT. ENSURE THERE ARE NO DISCONNECTED ELECTRICAL LEADS WITHIN TANKS.

CAUTION: PRIOR TO DEFUELING FUEL TANK VENT SHOULD BE CHECKED TO ENSURE IT IS OPEN AND NO OBSTRUCTIONS EXIST IN OVERBOARD VENT LINE. A CLOSED OR OBSTRUCTED VENT COULD RESULT IN COLLAPSING WING.

WHEN DEFUELING AIRCRAFT 164 - 200, AND AIRCRAFT 1 - 163, 322 AND 323 HAVING ASC 178. THE DRAIN VALVES ARE RELOCATED DOWNSTREAM OF FLOWMETER TRANSMITTER. WHEN DEFUELING, FUEL FLOW MAY EXCEED FLOW LIMITATIONS OF FLOWMETER INDICATOR RESULTING IN DAMAGE TO INSTRUMENT. IN ORDER TO PREVENT THIS, THE AC INSTRUMENT BUS MUST BE DEACTIVATED BY ENSURING A, B AND E-INVERTERS ARE IN OFF POSITION.

NOTE: This procedure is written for right side only. It must be repeated on left side to completely defuel aircraft fuel tanks. It is recommended a slow acting shutoff valve be installed at drain valve outlet during defueling operations. Slow acting shutoff is closed first when draining under pressure in order to prevent shocking of altitude capsule in flow control unit.

A. Attach one end of drain hose to right hand drain valve located in main landing gear wheel well.

B. Check HP fuel cock and crossfeed closed.

C. Check FIRE PULL T-handle is in open position.

D. Supply 28V dc power to aircraft.

E. Select R Norm or R Aux fuel boost pump ON.

F. Open drain valve and open slow acting valve.

G. When defueling is completed position fuel boost pump switches to OFF.

H. Close slow acting valve and drain valve.

I. Remove drain hose.

J. Remove slow acting valve.

K. Fuel remaining in tanks at this point may be removed through two tanks drains located on lower inboard wing planks.

4. Defueling — Suction thru Drain Valves in Main Landing Gear Wheel Well.

WARNING: WHEN DEFUELING, HAVE FIRE FIGHTING EQUIPMENT AVAILABLE TO COMBAT PETROLEUM FIRES, DO NOT PERMIT SMOKING, OR USE OF ANY ELECTRICAL EQUIPMENT WHICH DOES NOT EMPLOY EXPLOSION PROOF FEATURES. ELECTRICALLY GROUND ALL EQUIPMENT AND AIRCRAFT. ENSURE THERE ARE NO DISCONNECTED ELECTRICAL LEADS WITHIN THE TANKS.

CAUTION: PRIOR TO DEFUELING, FUEL TANK VENT SHOULD BE CHECKED TO ENSURE IT IS OPEN AND NO OBSTRUCTIONS EXIST IN OVERBOARD VENT LINE. A CLOSED OR OBSTRUCTED VENT COULD RESULT IN COLLAPSING THE WING.

WHEN DEFUELING, AIRCRAFT 164 - 200, AND AIRCRAFT 1 - 163, 322 AND 323 HAVING ASC 178. DRAIN VALVES ARE RELOCATED DOWNSTREAM OF FLOWMETER TRANSMITTER. WHEN DEFUELING OR DRAINING FUEL, THE FUEL FLOW MAY EXCEED FLOW LIMITATIONS OF FLOWMETER INDICATOR RESULTING IN DAMAGE TO THE INSTRUMENT. IN ORDER TO PREVENT THIS, THE AC INSTRUMENT BUS MUST BE DEACTIVATED BY ENSURING A, B AND E-INVERTERS ARE IN OFF POSITION.

NOTE: This procedure is written for right side only. It must be repeated on left side to completely defuel the aircraft fuel tanks. It is recommended a slow acting shutoff valve be installed at drain valve outlet during defueling operations. Slow acting shutoff valve is closed first when draining under pressure in order to prevent shocking of altitude capsule in the flow control unit.

A. Attach one end of defueling hose to right hand drain valve located in main landing gear wheel well.

B. Check HP fuel cock and crossfeed closed.

C. Check FIRE PULL T-handle is in OPEN position.

D. Supply 28V DC power to aircraft.

E. Select R Norm or R Aux fuel boost pump ON.

F. Open drain valve and open slow acting valve.

G. Under truck suction, commence defueling.

H. Select fuel boost pump OFF.

I. When defueling is completed close slow acting valve and drain valve.

J. Remove defueling hose.

K. Remove slow acting valve.

L. Fuel remaining in tanks at this point may be removed through tank drains located on lower inboard wing planks.

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WATER METHANOL SERVICING — MAINTENANCE PRACTICES

1. Water Methanol Tanks — Servicing

WARNING: WATER METHANOL IS AN EXTREMELY VIOLENT POISON. IF IT IS INHALED OR SWALLOWED, OR ENTERS THE BODY BY ANY OTHER MEANS, IT WILL CAUSE BLINDNESS AND GENERAL PHYSICAL DECAY, WITH DEATH RESULTING EITHER SOON OR ULTIMATELY. IT CANNOT BE APPLIED EXTERNALLY WITHOUT CAUSING SERIOUS INJURY. AVOID INHALATION OF ITS VAPORS OR PROLONGED CONTACT WITH SKIN, AND PROMPTLY WASH OFF ANY SPILLED ON THE SKIN.

ANTIDOTE: IF ACCIDENTALLY TAKEN INTERNALLY:

A. GIVE FIRST AID AS FOLLOWS: IMMEDIATELY GIVE EMETIC OF MUSTARD IN MILK TO INDUCE VOMITTING. GIVE MILK, WHITE OF EGG, OR FLOUR IN WATER AND PURGATIVE OF EPSOM SALT.

B. CALL PHYSICAN AT ONCE.

CAUTION: OVERFLOWING TANKS CREATE A SERIOUS FIRE HAZARD AND MUST BE AVOIDED. DAMAGE TO DE-ICER BOOTS COULD ALSO RESULT. SPILLED WATER METHANOL MUST BE WIPED UP IMMEDIATELY.

A. Checking water methanol mixture prior to replenishing tanks.

Water methanol mixture remaining in aircraft tanks may have become contaminated and contamination may be of a large enough quantity to affect specific gravity of water methanol mixture therefore, a specific gravity check of mixture should be made (see Water/Methanol Mixture — Test (Field Test, this section) before servicing.

B. Overwing Replenishing of Water Methanol Tanks (See Section 12-0 Figure 1).

CAUTION: BECAUSE OF LOCATION OF FUEL AND WATER METHANOL FILLERCAPS, BE CAREFUL NOT TO INADVERTENTLY PUT FUEL INTO WATER METHANOL TANKS. ENGINE WILL BE OVERTORQUED IF FUEL IS USED INSTEAD OF WATER METHANOL.

(1) Check quantity in tanks using cockpit gages.

CAUTION: IF TANKS NEED REFILLING, CARE SHOULD BE TAKEN NOT TO DRAG HOSE ACROSS WING AS DAMAGE TO DE-ICER BOOTS COULD RESULT. IT IS RECOMMENDED HOSE BE BROUGHT OVER TRAILING EDGE OF WING RATHER THAN LEADING EDGE.

(2) Ground water/methanol hose nozzle to aircraft structure through grounding jacks located inboard of filler caps.

(3) Add sufficient water methanol to fill tanks without overflowing.

NOTE: Pumps which are alternately exposed to water methanol, then air, are subject to corrosion. This corrosion may start after 12 hours exposure to air. It is recommended that pumps be covered with water methanol at all times. In terms of quantity required to cover the pumps, following is recommended.

- Standard System (Dual Bag) 12 gallons (1/4 tank) (Aircraft 1 - 106 and 114 not having ASC 165.)
- Standard System (Single Bag) 12 gallons (1/2 tank)
- (Aircraft 107 - 200, 322, and 323, and Aircraft 1 - 106, and 114, having ASC 165.)
- Modified System (Single Tank) 12 gallons (1/2 tank) (Aircraft, having ASC 125.)

(4) Check and ensure filler caps are securely fastened.

2. Water/Methanol Mixture — Testing (Field Test)

A. Drain a small quantity of water/methanol mixture into a clean glass container. Mixture must meet the following requirements:

(1) Mixture shall be clear and free of sediment or suspended matter.

(2) Specific gravity of water/methanol mixture at 60°F (15.6°C) shall be not less than 0.9412 (18.8° API) and not more than 0.9445(18.3° API). At mixture temperatures other than 60°F (15.6°C) specific gravity must be within the limits given on the curves in Chapter 89 of the Maintenance Manual for the Rolls-Royce Dart 529 Engine.

NOTE: For information on specifications, preparation and storage, determination of non-volatile solids content of water, filtration, determination of silica content of water refer to Chapter 89 of Rolls Royce Maintenance Manual M-Da 7-G.

WATER-METHANOL SERVICING — INSPECTION/CHECK

Testing Water-Methanol Mixture (Field Test)

A. Drain a small quantity of the water-methanol mixture into a clean glass container. The mixture must meet the following requirements:

(1) The mixture shall be clear and free of sediment or suspended matter.

(2) The specific gravity of the water-methanol mixture at 60° F (15.6° C) shall be not less than 0.9412 (18.8° A.P.I.) and not more than 0.9445 (18.3° A.P.I.). At mixture temperatures other than 60° F (15.6° C) the specific gravity must be within the limits given on the curves in Chapter 89 of the Maintenance Manual for the Rolls-Royce Dart 529 Engine.

NOTE: For information on specifications, preparation and storage, determination of non-volatile solids content of water, filtration, determination of silica content of water refer to Chapter 89 of Rolls Royce Maintenance Manual M-Da 7-G.

HYDRAULIC SERVICING — MAINTENANCE PRACTICES

1. **Equipment and Approved Fluids**

A. Approved Hydraulic Fluids

Type IV, phosphate ester fluids listed below are preferred for the hydraulic system. Type IV fluids have all the beneficial characteristics of the Type II fluids plus they provide erosion resistance to hydraulic components and improved thermal stability. HyJet IV and Skydrol LD-4 are lower density, resulting in a significant weight saving.

All approved phosphate ester fluids are compatible and can be safely mixed. However, up to 15% HyJet IV in Skydrol 500B may cause an increase in valve erosion. There is no restriction on HyJet IV-A mix.

All other concentrations of mixture of the approved phosphate ester fluids have complete compatibility.

The approved Type IV fluids are listed below.

IDENTIFICATION	COLOR	VENDOR
HyJet IV	Purple	Chevron Chemical Company
HyJet IV-A	Purple	Chevron Chemical Company
Skydrol LD-4	Purple	Monsanto Chemical Company
Skydrol 500B-4	Purple	Monsanto Chemical Company

The approved Type II fluids are listed below. These fluids are no longer produced.

IDENTIFICATION	COLOR	VENDOR
Skydrol 500A	Purple	Monsanto Chemical Company
Skydrol 500B	Purple	Monsanto Chemical Company
Oronite HyJet W	Purple	Chevron Chemical Company
Aero Safe 2300W	Blue	Stauffer Chemical Company
Skydrol 7000	Green	Monsanto Chemical Company

CAUTION: ALL RECOMMENDED SAFETY AND HANDLING PRECAUTIONS SHOULD BE ADHERED TO WHEN SERVICING AIRCRAFT OR HANDLING HYDRAULIC FLUIDS.

NEW AND IN SERVICE FLUID LIMITS	*NEW	**IN-SERVICE
APPEARANCE	NO CLOUDINESS, PHASE SEPARATION OR PRECIPITATION	
SPECIFIC GRAVITY @ 25°C/77°F	0.999	0.995 - 1.066
MOISTURE WT. %	0.20	0.1 - 0.8
NEUTRALIZATION NO. mg KOH/gm	0.07	1.5 MAX.
KINEMATIC VISC. @ 100°F, CS	10.58	6.0 - 12.5

NEW AND IN SERVICE FLUID LIMITS	*NEW	**IN-SERVICE
ELEMENTAL CONTAMINATION: (PPM)		
CALCIUM	10	50 MAX.
POTASSIUM	10	50 MAX.
SODIUM	10	50 MAX.
CHLORINE	20	200 MAX.
SULFUR	20	500 MAX.
PARTICLE CONTAMINATION (PPM)	SHALL NOT EXCEED NAS 1638CL.7	SHALL NOT EXCEED NAS 1638CL.9
SIZE RANGE (MICRONS)		
5 TO 15	32,000	128,000
15 TO 25	5700	22,800
25 TO 50	1012	4050
50 TO 100	180	720
100	32	128

*CHEVRON HYJET AND THE BOEING COMPANY REQUIREMENTS.
**FROM THE BOEING COMPANY, BAE AND FOKKER.

B. Replenish Hydraulic Reservoir.

WARNING: ABOVE APPROVED HYDRAULIC FLUIDS ARE ACID BASE WHICH WILL CAUSE TEMPORARY BURNING AND IRRITATION IF IT COMES IN CONTACT WITH EYES. IF THIS OCCURS, IMMEDIATELY WASH EYES OUT WITH WATER. SHOULD ANY OF THE ABOVE HYDRAULIC FLUIDS BE SPILLED, WIPE UP IMMEDIATELY AS IT CAN DAMAGE MATERIALS SUCH AS RUBBER, PAINT, ETC.

NOTE: Before filling reservoir, check landing gear is in down and locked position, airstair door open and brake accumulator is charged.

(1) Check sight gage at reservoir located in hydraulic service center.

(2) Normally, main system and auxiliary system reservoir is filled from outside of the aircraft as follows:

(a) Remove reservoir filler cap access cover located just aft of airstair door. (See Section 12-0).

(b) Unscrew reservoir filler cap and fill to level of filler neck opening.

(c) Secure filler cap and access cover.

(3) In an emergency, auxiliary system side of the reservoir can be filled during flight from inside of aircraft as follows:

(a) Unscrew filter cap of reservoir located in hydraulic compartment. Air pressure trapped in reservoir will bleed off when approximately 1/2 turn of filler cap has been turned.

CAUTION: DO NOT OVERFILL.

(b) Fill reservoir to full capacity as indicated on sight gage. Replace and secure filler cap.

C. Ground Service Pressure Provisions

~~External self sealing quick disconnects are provided for connecting an external hydraulic power source.~~ Disconnects are located at right engine hydraulic pump.

Internal jumper connections are provided so main system can be pressurized with auxiliary hydraulic pump. These connections are located in nose wheel well, right side, just aft of gear. Two capped T-fittings are provided, one in each hydraulic gage line. To use, uncap fittings and attach a number four hydraulic flex line at least 15 inches long.

WARNING: JUMPER MUST BE REMOVED BEFORE FLIGHT.

CAUTION: THIS VALVE IS INTENDED FOR GROUND OPERATION ONLY. WHEN IN USE, GROUND SAFETY LOCK 159GT1007 MUST BE PROPERLY INSERTED AND RED FLAG DISPLAYED IN FULL VIEW. AFTER USE, SAFETY LOCK MUST BE REMOVED AND VALVE HANDLE RETURNED TO FLIGHT POSITION BEFORE TAKEOFF.

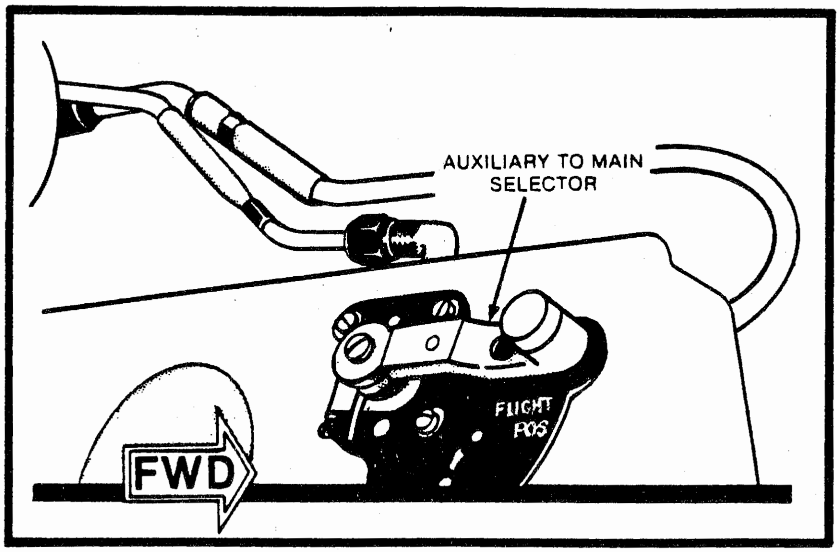
On Aircraft having ASC 109A manually operated valve enables maintenance personnel to pressurize main hydraulic system operating auxiliary hydraulic pump in lieu of main engine driven hydraulic pumps or a hydraulic rig.

Installation involves linking two previously mentioned T-fittings in nose wheel well with a permanently mounted, two position, manually operated selector valve, with associated plumbing (See Figure 201)

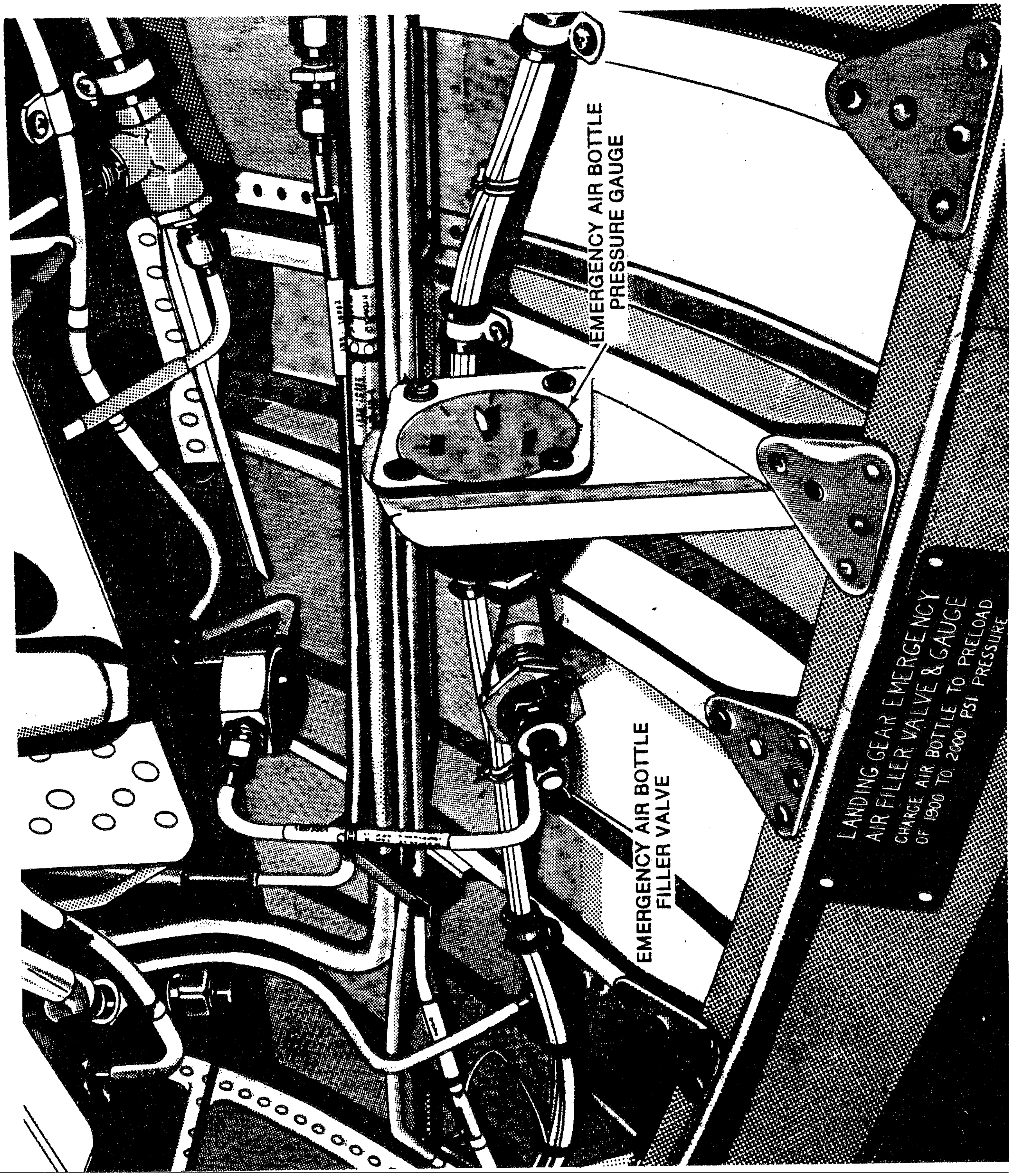
Valve is placarded FLIGHT POSITION which is closed, and AUX-TO-MAIN ON which is open.

Pneumatic Bottle (See Figure 202)

A pneumatic bottle is installed in nose wheel well to provide air pressure for emergency extension of the landing gear. A pressure gage for this bottle is located below and aft of the bottle. A preflight check is required and limitations are 1950 + 50 psi.



Auxiliary to Main Selector Valve Location
Figure 201.



Nose Wheel Well — Location of Landing Gear Emergency Air Filler Valve and Gage
Figure 202.

OIL SERVICING — MAINTENANCE PRACTICES

WARNING: SYNTHETIC LUBRICATING OIL USED IN ENGINES AND ACCESSORY GEARBOXES CONTAIN ADDITIVES WHICH, IF ALLOWED TO CONTACT SKIN FOR PROLONGED PERIODS, CAN BE TOXIC THROUGH SKIN ABSORPTION.

CAUTION: OIL SPECIFIED FTR USE IN OIL SYSTEM (ENGINE AND GEARBOX) IS A SYNTHETIC PRODUCT AND MUST NOT BE MIXED WITH ANY OTHER OIL. REFER TO ROLLS-ROYCE MAINTENANCE MANUAL FOR APPROVED OILS. IT IS INJURIOUS TO PAINT AND CERTAIN TYPES OF RUBBER. IT MUST NOT BE ALLOWED TO CONTAMINATE THOSE PARTS OF AIRCRAFT NOT NORMALLY IN CONTACT WITH IT. ANY OIL SPILLED DURING SERVICING MUST BE WIPED UP IMMEDIATELY.

1. Engine Oil System — Servicing

(See Section 12-0, Figure 3 and Figure 4)

CAUTION: USE ONLY SCRUPULOUSLY CLEAN CONTAINERS AND EQUIPMENT WHEN SERVICING ENGINES. CONTAMINATION OF OIL BY FOREIGN MATTER SUCH AS CLEANING FLUIDS, ETC., CAN DESTROY THE LUBRICATING PROPERTIES OF OIL AND FOUL UP FILTERS.

A. Check engine oil quantity, Replenish as necessary.

(1) Check quantity of oil with dipstick. To ensure stabilized conditions and consistent results, check oil level in the tank and any topping off required should be carried out between 10 and 30 minutes after shutdown. This procedure will obviate over or under-filling and will facilitate accurate recording of oil consumption. Refer to Chapter 12 or Rolls-Royce Maintenance Manual M-Da7-G for approved lubricants.

(2) For checking oil at times other than above, (i.e. for engine out of service standing idle for a period of time) refer to Rolls-Royce Maintenance Manual M-Da7-G.

2. Accessory Gearbox Oil — Servicing

(See Section 12-0 Figure 3 and Figure 4)

A. Accessory gearbox oil level check.

(1) Open access door above gearbox, top of nacelle.

(2) Check oil level with dipstick on top of unit. Replenish if necessary using procedure in step 2. B.

(3) Replace dipstick, ensuring it is locked in place (twist lock type) or retained in place (double spring leaf type).

(4) Inspect area for presents of foreign objects, security of all attachments leaks.

(5) Secure access door.

B. Accessory Gearbox Oil Service

Refer to Chapter 12 or Dowty Rotol Maintenance Manual 856/1 for approved lubricants.

Oil level in gearbox must be checked at times specified in Dowty Rotol Maintenance Manual oil servicing procedure. To add oil proceed as follows:

(1) Loosen wing nut and swing back fillercap.

(2) Release dipstick by either pushing it down and rotating 1/4 turn or lifting and positioning double leaf spring retainer to one side so it can be removed.

(3) Wipe dipstick clean and add oil until FULL mark on dipstick is reached.

~~(4) After a reasonable time, dipstick should again be removed and oil level checked, to ensure it~~ reads FULL.

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OXYGEN SERVICING — MAINTENANCE PRACTICES

1. Oxygen System — Servicing

A. Service Oxygen Cylinder

Oxygen cylinder is located in nose section on left side.

(1) Maximum pressure for oxygen cylinder is 1800 psi at 15°C (59°F), 1700 psi at −1°C (30°F), or 1970 psi at 43°C) (110°F.).

NOTE: Minimum amount of oxygen required for flight crew is 1000 PSI at 15°C (59°F). (Standard Day)

(2) Replace oxygen cylinder if pressure becomes less than 50 psi.

(3) Recharge cylinder through filler valve located on left side of nosewheel well.

NOTE: It is recommended oxygen cylinder pressure not be permitted to get below 100 psi.

B. To recharge oxygen cylinder using oxygen filler valve on left side, forward in nose wheel well, proceed as follows:

WARNING: USE ONLY AVIATOR GASEOUS BREATHING OXYGEN, MILITARY SPECIFICATION MIL-O-27210 TO SERVICE OXYGEN CYLINDER. KEEP ALL LUBRICANTS CLEAR OF OXYGEN.

Aircraft not having ASC 200A.

(1) In cockpit, ensure pilots and copilots regulator red emergency knob is fully closed and CABIN SHUTOFF VALVE (copilots side console-aft) is CLOSED.

(2) Open oxygen shutoff valve on pilots outboard skirt panel. Oxygen cylinder pressure gage on same panel should indicate cylinder pressure.

(3) Gain access to oxygen filler valve in forward, left side of nose wheel well by opening clamshell doors, using procedure outlined in Section 12-0.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(4) In nose wheel well, remove cap from oxygen filler valve. Clean any dirt and contamination from valve.

(5) In external oxygen filler equipment (must be capable of supplying 1800 psi at 15°C at (59°F) with shutoff valve on unit) crack external filler rig shutoff valve to blow rig line clear of dirt and contamination before connection to aircraft. Close valve.

(6) In nose wheel well, connect external oxygen filler equipment to oxygen filler valve. With a man in cockpit to observe cylinder pressure gage, slowly open external equipment rig shutoff valve and fill cylinder until gage reads the proper level as predetermined by Table A.

(7) Close external oxygen filler equipment shutoff valve when correct pressure is reached.

(8) In cockpit, close oxygen supply shutoff valve.

(9) Disconnect external equipment supply line from filler valve. Test filler valve opening for leaks with MIL-L-25576 Leak-Tec formula OX16 or solution conforming to MIL-L-25576. Wipe off solution after test is completed.

TEMP °F	TEMP °C	FILL TO PSIG
0	−17.8	1595
+ 10	−12.2	1630
+ 20	−6.7	1665
+ 30	−1.1	1700
+ 40	+ 4.4	1730
+ 50	+10.0	1765
+ 60	+15.0	1800
+ 70	+21.1	1835
+ 80	+26.7	1865
+ 90	+32.2	1900
+100	+37.8	1935
+110	+43.3	1970
+120	+48.9	2000

Oxygen Cylinder Replenishing — Pressure vs Temperature Relationship
Figure 201.

NOTE: Initial temperature refers to ambient temperature before filling. Filling pressure refers to pressure to which cylinders must be filled.

(10) Replace cap on filler valve and tighten to a low torque. Cap keeps connection clean,and ensures against leaks. Therefore, it must be replaced.

NOTE: Filter valve has an allowable leak rate of 0.5 liter per minute. It is therefore necessary to replace the cap on the valve to prevent leakage.

Aircraft having ASC 200A

(1) In cockpit, ensure pilots and copilots regulator red emergency knob is fully closed and the CABIN SHUTOFF VALVE (copilots side console aft) and the OXYGEN SUPPLY SHUTOFF VALVE on the pilots outboard panel are CLOSED.

WARNING: COMPARTMENT INTERIOR, FILLER VALVE AND CONNECTION MUST BE COMPLETELY FREE OF CONTAMINATION.

(2) Open oxygen servicing compartment access door and remove oxygen filler valve cap from valve.

(3) On external oxygen filler equipment (must be capable of supplying 1800 psi at 15°C (59°F) with shutoff valve on unit) crack external filler rig shutoff valve to blow rig line clear of dirt and contamination before connection to aircraft. Close valve.

(4) Connect external rig to oxygen filler valve.

(5) While observing OXYGEN CYLINDER PRESSURE GAGE, adjacent to oxygen filler valve, slowly open external rig shutoff valve and fill cylinder until CYLINDER GAGE reads proper level as determined from Table A of this page.

(6) Close external rig shutoff valve when correct pressure is reached.

(7) Disconnect external rig supply line from filler valve.

(8) Check filler valve opening for leaks with Leak-Tec formula OX16 or solution conforming to MIL-L-25576. Wipe off solution after test for leak is completed.

(9) Replace cap on filler valve and tighten to a low torque. Cap keeps connection clean, and ensures against leaks, therefore, it must be replaced.

NOTE: Filler valve has an allowable leak rate of 0.5 liter per minute.It is therefore necessary to replace cap on the valve to prevent leakage.

(10) Inspect area for presents of foreign objects, security of all attachments.

(11) Close access door.

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BATTERY SERVICING — MAINTENANCE PRACTICES

1. Servicing Batteries

- See Chapter 5 for time periods.
- See Chapter 24 for maintenance practices.

2. Determining State of Charge of a Battery (Flight Line)

A. Connect batteries.

B. Pull either L or R BATT CONTROL circuit breaker.

C. Set battery switch to NORM.

D. Energize one fuel pump and E inverter. This provides a current drain of approximately 35 amperes. (L or R AUX OR NORMAL BOOST PUMP SW-ON, ESS. AC BUS SEL. SW-NORM)

E. Note dc bus voltage on upper overhead panel. It should be above 22 volts. If it reads below 22 volts, battery may be considered discharged. Do not prolong time load is switched on because this needlessly drains battery.

F. Determine state of charge of other battery by same procedure as above.

3. Battery Charging

Battery Charging with External Power Source

A. Check switches in aircraft are set to OFF.

(1) Plug dc external power unit into power receptacle. Turn unit switch ON.

(2) Set EXT PWR control switch in cockpit to ON.

CAUTION: DO NOT USE VOLTAGES ABOVE 32 VOLTS DC AS EQUIPMENT MAY BE DAMAGED AND BATTERIES MAY OVERHEAT AND SPEW.

(3) Remove all unnecessary loads from main and essential dc bus.

(4) Note ammeter reading on external power unit. This reading is current required by aircraft main dc bus.

(5) Place aircraft battery switch in NORM position and note rise in current on external power cart ammeter. (If batteries are completely discharged this rise in value may be as high as 400 amperes.)

(6) Note ESS DC VOLTS reading on meter in overhead panel. It should indicate external power voltage (recommended 28 to 29 volts).

(7) Continue charging until ammeter has dropped to about same reading recorded before aircraft battery switch was closed. Time required is 10-15 minutes.)

(8) When current has dropped, batteries are charged and ready for service. Set aircraft battery switch to OFF. Turn off external power switch. Turn external power unit off and disconnect if from aircraft.

(9) If battery power is to low to energize the battery relay, perform the following:

(a) Ensure that EXT. POWER SW. is ON and that BATT SW is in NORM.

(b) Locate normal battery relay for associated battery. (Aft of battery in battery compartment).

WARNING: WHEN PERFORMING THE FOLLOWING STEPS ENSURE AIRCRAFT BATTERY SWITCH IS IN NORM POSITION WHEN JUMPER WIRE IS PLACED ON RELAY. OTHERWISE, BATTERY WILL ATTEMPT TO TAKE A CHARGE THROUGH THE SMALL JUMPER WIRE.

(c) Place jumper wire (momentarily) across GEN. and BATT. terminals of NORM. BATT. relay. Relay should close allowing battery to charge through heavy duty contacts of relay.

(d) Check ammeter on external power unit for rise in current as battery relay closes. When operation is complete, switch both external power and battery switch to OFF. Disconnect external power source from aircraft.

B. Charging Batteries with APU Generator.

CAUTION: AN ENGINE START SHALL NEVER BE ATTEMPTED WHEN APU GENERATOR IS ON LINE WITH, OR WITHOUT, AIRCRAFT BATTERIES. A START MADE IN THIS FASHION WILL CAUSE APU GENERATOR TO DELIVER CURRENT IN EXCESS OF ITS RATING. THIS CAN CAUSE PERMANENT DAMAGE TO APU GENERATOR.

MONITOR APU GENERATOR CURRENT.DO NOT EXCEED THE GENERATOR RATING. IT MAY BE NECESSARY TO CHARGE ONE BATTERY AT A TIME TO REMAIN WITHIN LIMITS.

(1) Remove all unnecessary loads from main and essential dc buses.

(2) Set battery switch to EMER.

(3) Start APU gas turbine unit and connect APU generator output to essential bus by placing APU GEN switch to ON.

NOTE: Batteries are now being charged from APU generator. A charge of one hour will provide sufficient battery capacity for 1 engine start. (One start is defined as three crank-overs for a maximum of 30 seconds duration each.)

(4) Upon completion of 1 hour charge, APU generator should be removed from bus by placing the APU GEN switch to OFF.

(5) Check bus voltage reading on dc bus voltmeter. Reading indicates battery voltage. Charged batteries should indicate 24 volts.

(6) Shut down APU and place APU MAST switch OFF.

ACCESS DOORS AND PANELS — DESCRIPTION / OPERATION

1. General

(See Figure 1 thru Figure 12)

Removable panels and access openings provide access for inspection and maintenance of the aircraft.

For the purpose of locating the numerous panels and access openings, the aircraft has been divided into areas identified by numbers. (See Figure 1) Each area located to the left and right of the fuselage is identified by odd and even, two-digit numbers, respectively, and each area about the fuselage centerline is identified by a three-digit number.

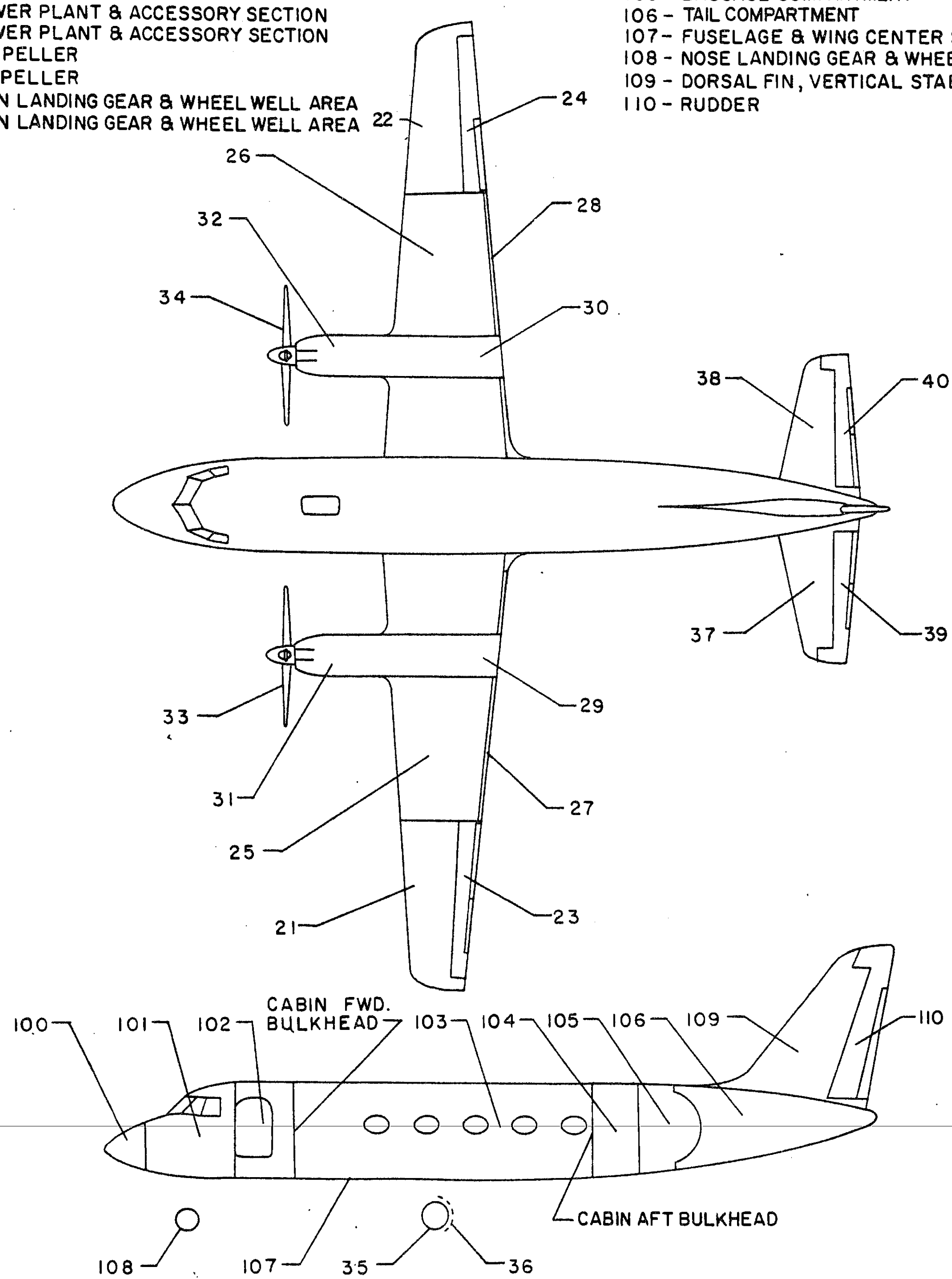
All access openings and covers are identified by a code number. The code number consists of three parts, for example 21-FT UPR-1. First, there is a two or three digit number which designates the general area where the access cover is located, in this case it is the left wing (21). Second, there are letters which further describe the purpose of the cover, again in this case (21-FT-UPR-1) it is the FUEL TANK, UPPER (Surface). Finally, there is one or two-digit number representing a numbering sequence of the access covers which, in this case, is cover number one.

A list of abbreviated letter symbols are given on each of the illustrations in Figure 2 thru Figure 10

The appropriate access covers or panels to use when performing maintenance are referred to in the pertinent chapter of this manual.

LEGEND

21 - LH OUTER WING PANEL
22 - RH OUTER WING PANEL
23 - LH AILERON
24 - RH AILERON
25 - LH INNER WING PANEL
26 - RH INNER WING PANEL
27 - LH FLAP
28 - RH FLAP
29 - LH NACELLE
30 - RH NACELLE
31 - LH POWER PLANT & ACCESSORY SECTION
32 - RH POWER PLANT & ACCESSORY SECTION
33 - LH PROPELLER
34 - RH PROPELLER
35 - LH MAIN LANDING GEAR & WHEEL WELL AREA
36 - RH MAIN LANDING GEAR & WHEEL WELL AREA
37 - LH HORIZONTAL STABILIZER
38 - RH HORIZONTAL STABILIZER
39 - LH ELEVATOR
40 - RH ELEVATOR
100 - NOSE COMPARTMENT
101 - COCKPIT
102 - ENTRANCE COMPARTMENT
103 - CABIN COMPARTMENT
104 - LAVATORY COMPARTMENT
105 - BAGGAGE COMPARTMENT
106 - TAIL COMPARTMENT
107 - FUSELAGE & WING CENTER SECTION
108 - NOSE LANDING GEAR & WHEEL WELL AREA
109 - DORSAL FIN, VERTICAL STABILIZER
110 - RUDDER

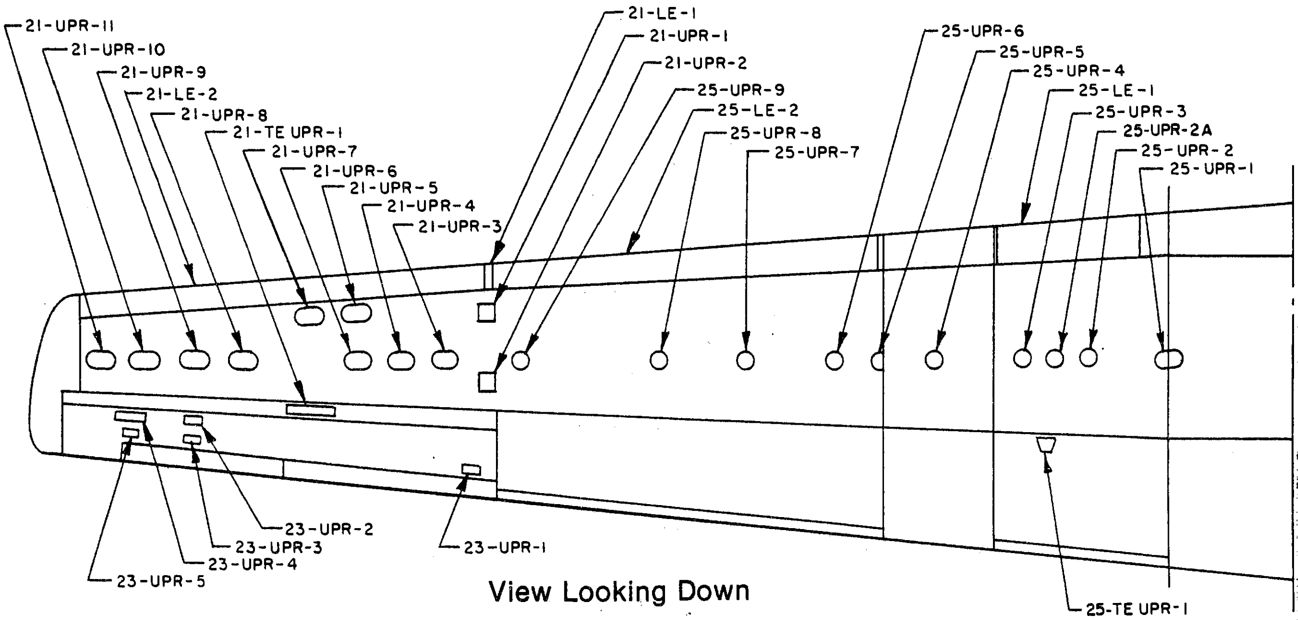


Area Locations
Figure 1.

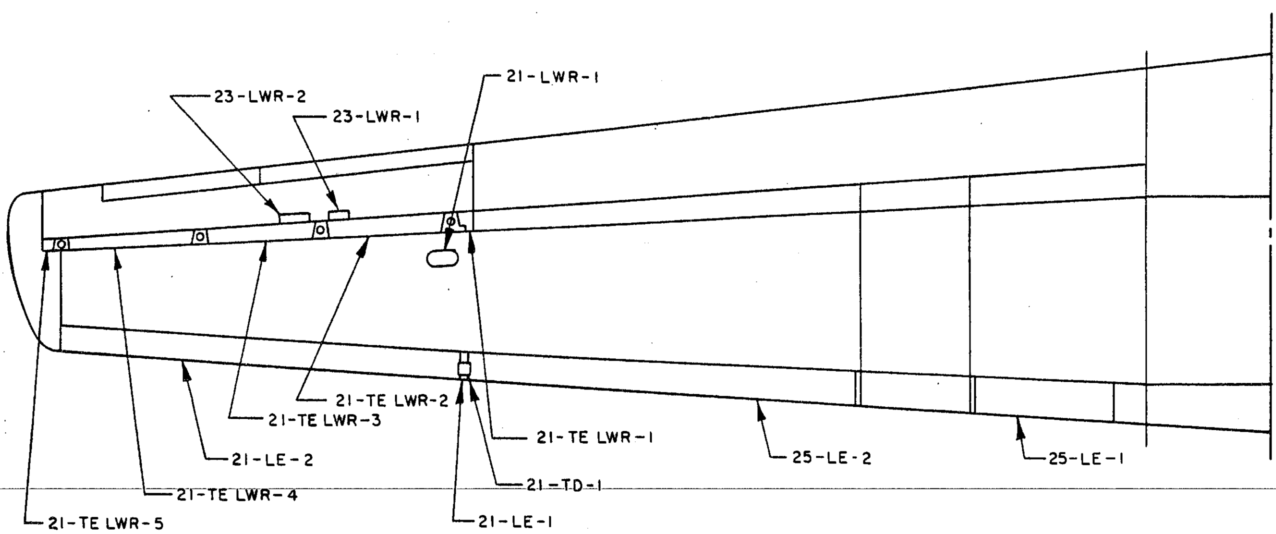
ACCESS OPENINGS

LEGEND

21 – LH OUTER WING PANEL	LWR – LOWER
23 – LH AILERON	TD – TRANSDUCER
25 – LH INNER WING PANEL	TE – TRAILING EDGE
LE – LEADING EDGE	UPR – UPPER



View Looking Down

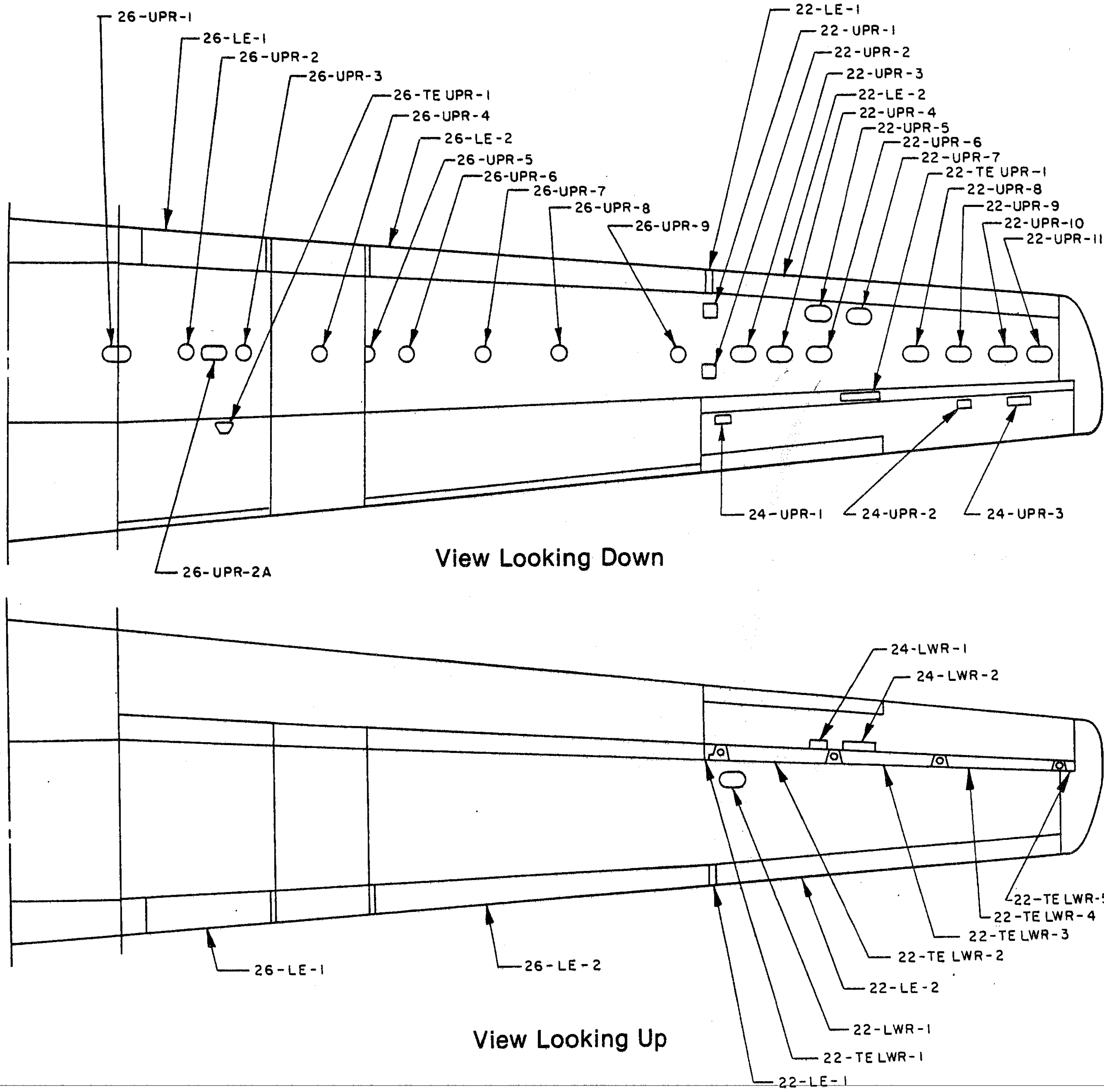


View Looking Up

ACCESS OPENINGS

LEGEND

22 - RH OUTER WING PANEL
24 - RH AILERON
26 - RH INNER WING PANEL
LE - LEADING EDGE
LWR - LOWER
TE - TRAILING EDGE
UPR - UPPER

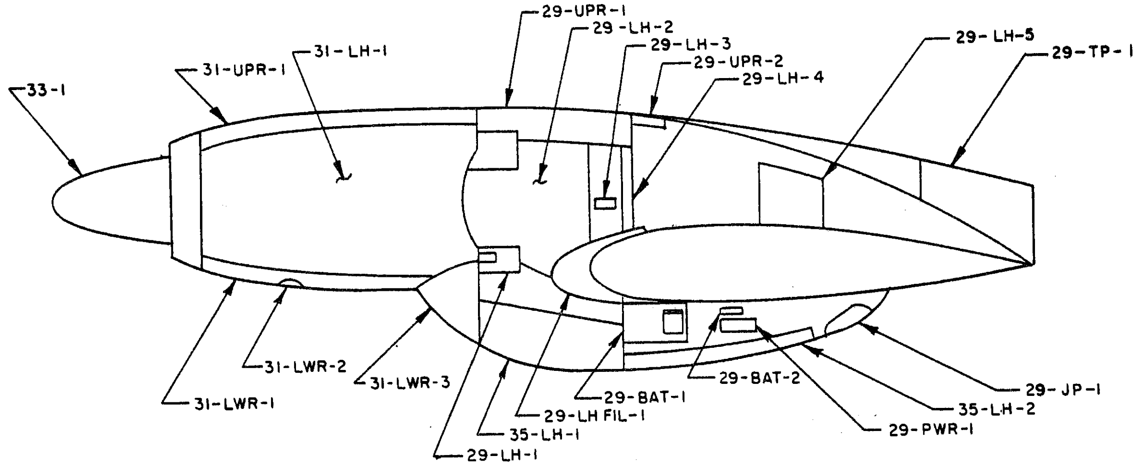


Right Wing/Aileron
Figure 3.

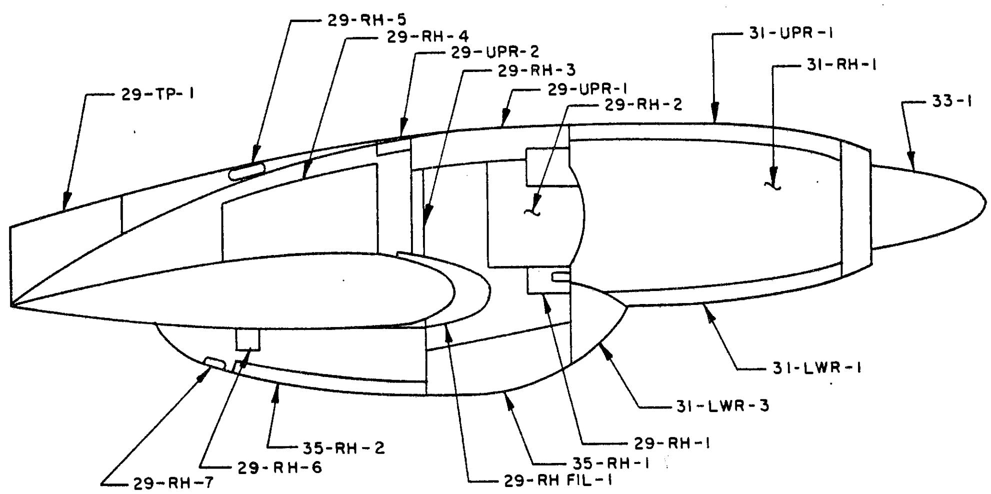
ACCESS OPENINGS

LEGEND

29 - LH NACELLE & ACCESSORY SECTION
31 - LH POWER PLANT
33 - LH PROPELLER
35 - LH MAIN LANDING GEAR & WHEEL WELL AREA
BAT - BATTERY
FIL - FILLET, WING TO NACELLE
JP - JACK POINT
LH - LEFT HAND
LWR - LOWER
PWR - POWER, EXTERNAL CONNECTION
RH - RIGHT HAND
TP - TAILPIPE
UPR - UPPER



View Looking Inbd



View Looking Outbd

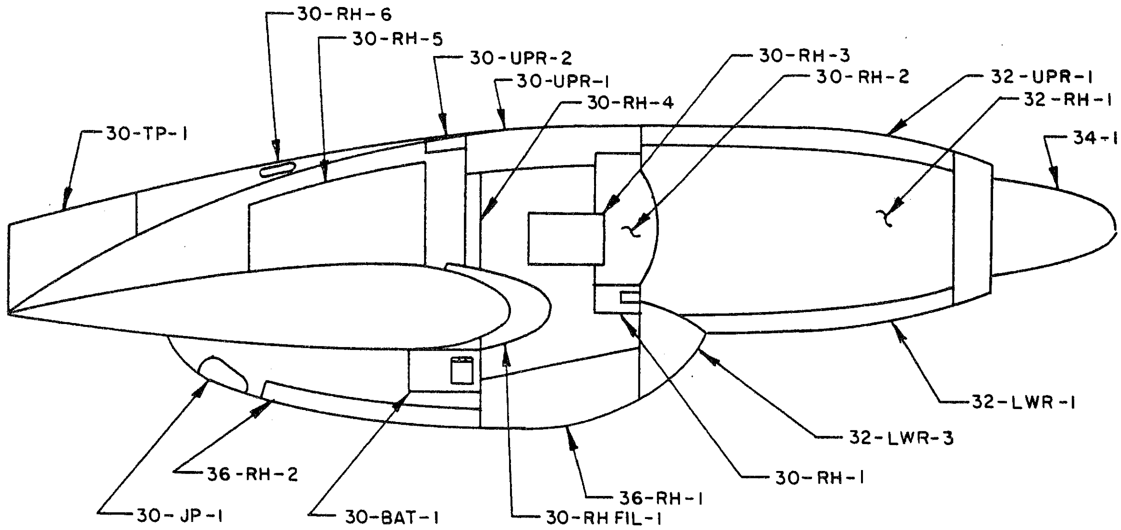
Left Engine Nacelle
Figure 4.

LEGEND

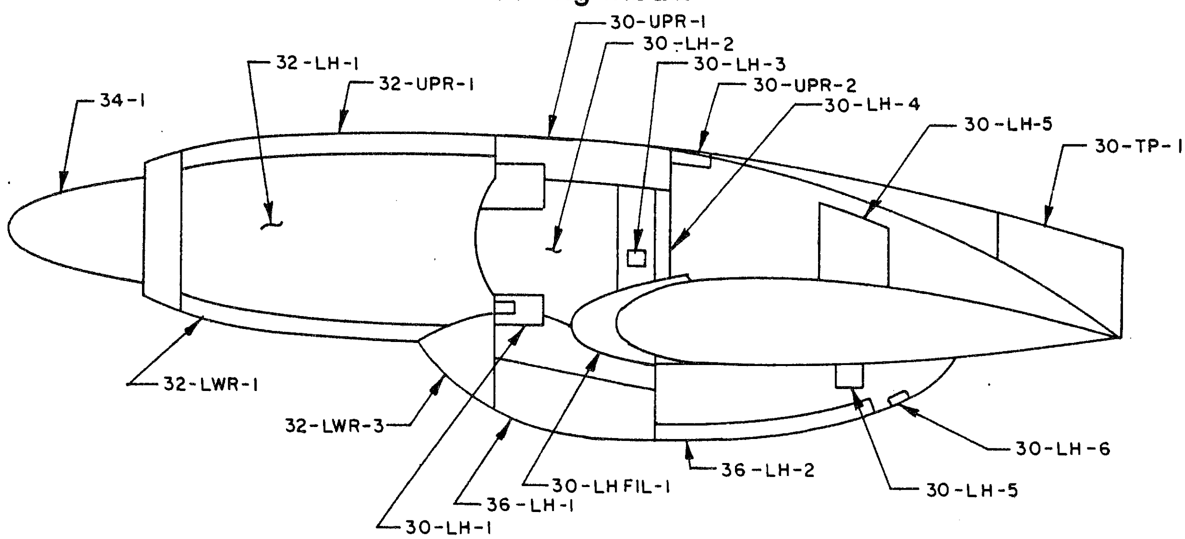
30 - RH NACELLE & ACCESSORY SECTION
32 - RH POWER PLANT
34 - RH PROPELLER
36 - RH MAIN LANDING GEAR & WHEEL WELL AREA
BAT - BATTERY
FIL - FILLET, WING TO NACELLE

JP - JACK POINT
LH - LEFT HAND
LWR - LOWER
RH - RIGHT HAND
TP - TAILPIPE
UPR - UPPER

ACCESS OPENINGS



View Looking Inbd



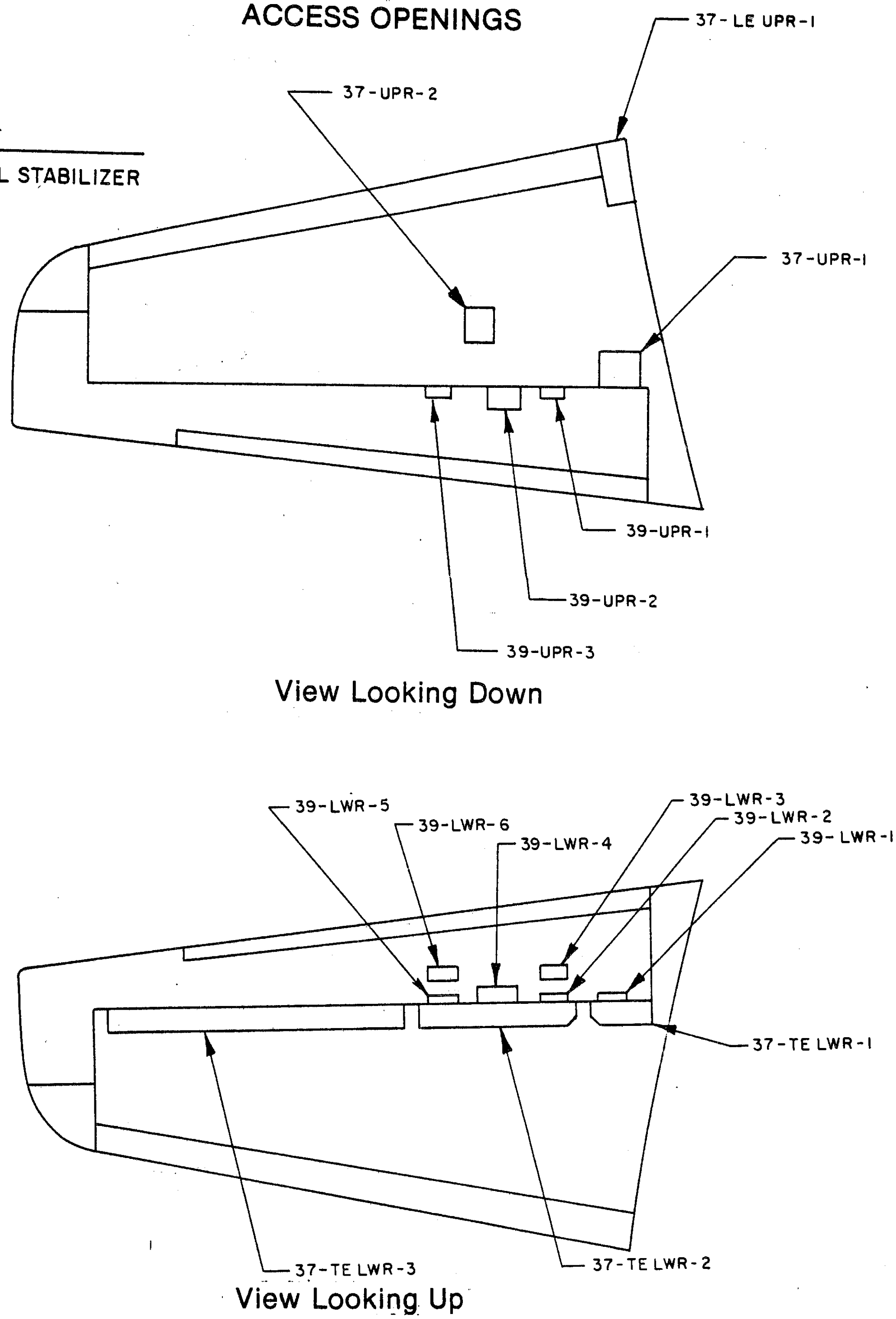
View Looking Outbd

Right Engine Nacelle
Figure 5.

ACCESS OPENINGS

LEGEND

37 – LH HORIZONTAL STABILIZER
39 – LH ELEVATOR
LE – LEADING EDGE
LWR – LOWER
TE – TRAILING EDGE
UPR – UPPER

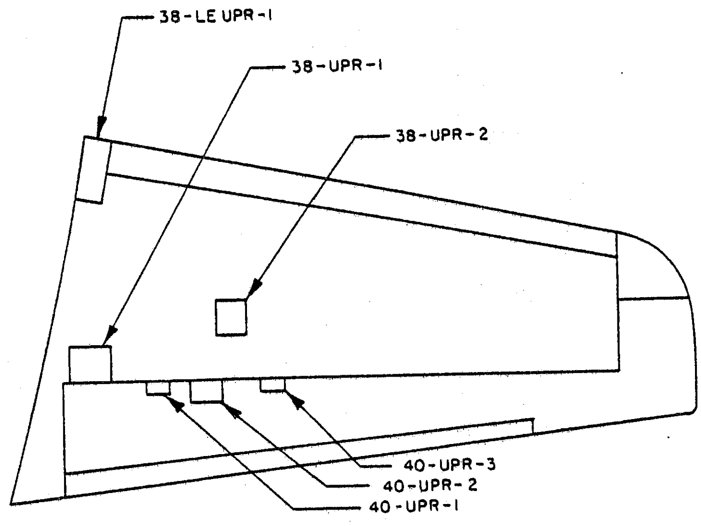


Left Horizontal Stabilizer/Elevator
Figure 6.

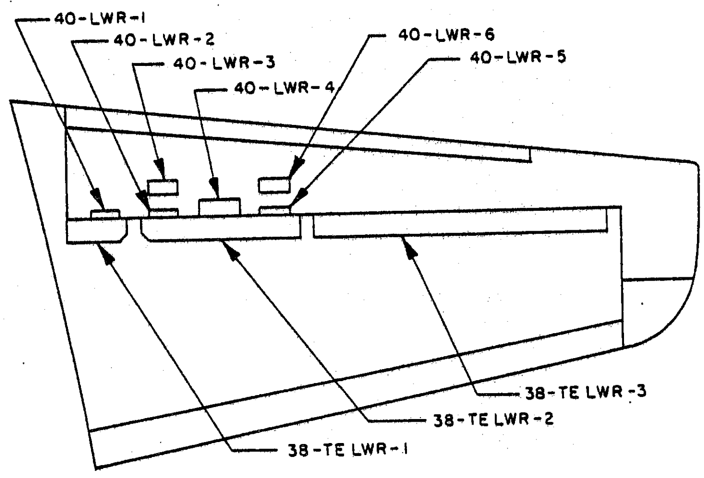
LEGEND

38 – RH HORIZONTAL STABILIZER
40 – RH ELEVATOR
LE – LEADING EDGE
LWR – LOWER
TE – TRAILING EDGE
UPR – UPPER

ACCESS OPENINGS



View Looking Down



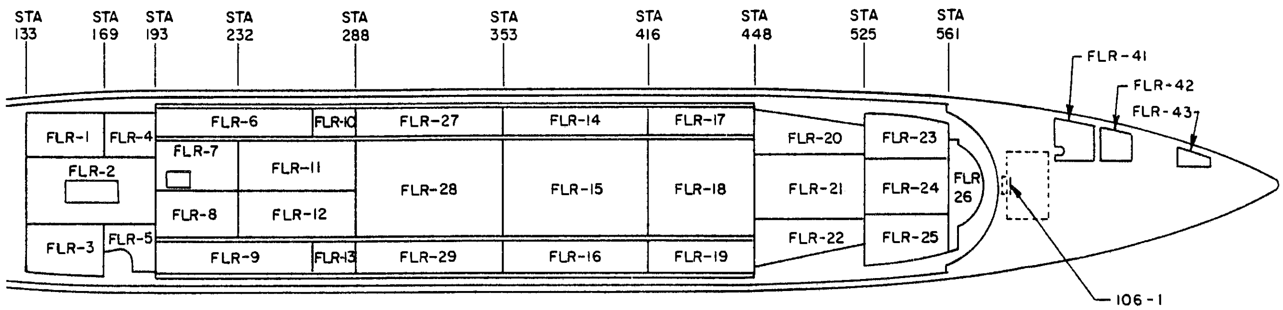
View Looking Up

Right Horizontal Stabilizer/Elevator
Figure 7.

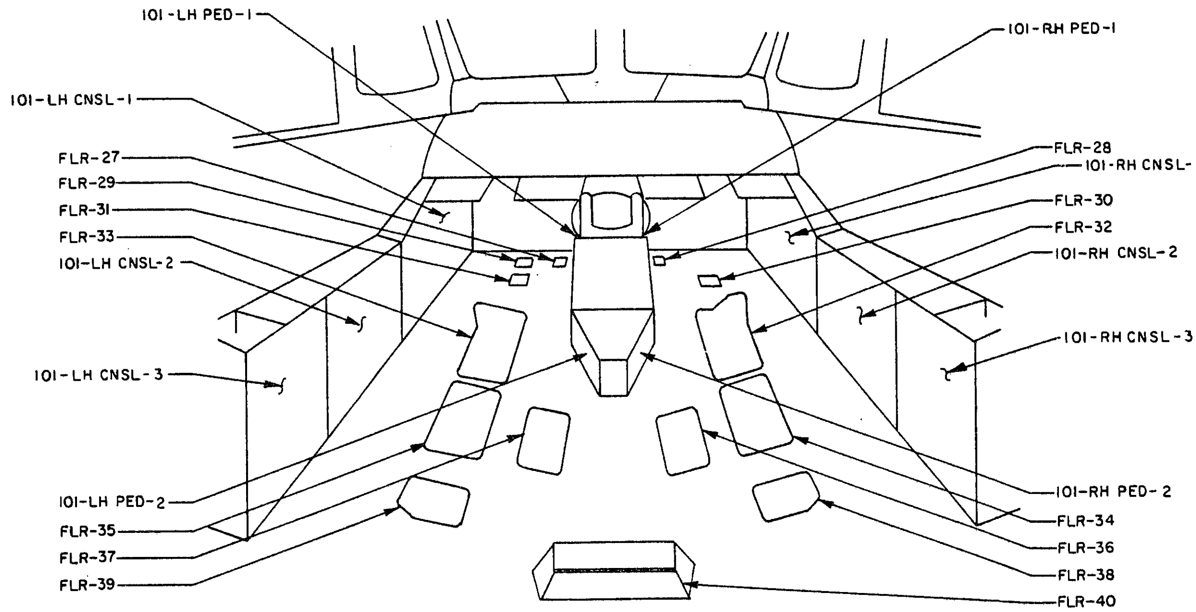
ACCESS OPENINGS

LEGEND

101 - COCKPIT COMPARTMENT
106 - INTERIOR-FUSELAGE TAIL SECTION
CNSL - CONSOLE
FLR - FLOORBOARD
LH - LEFT HAND
PED - PEDESTAL
RH - RIGHT HAND
STA - STATION



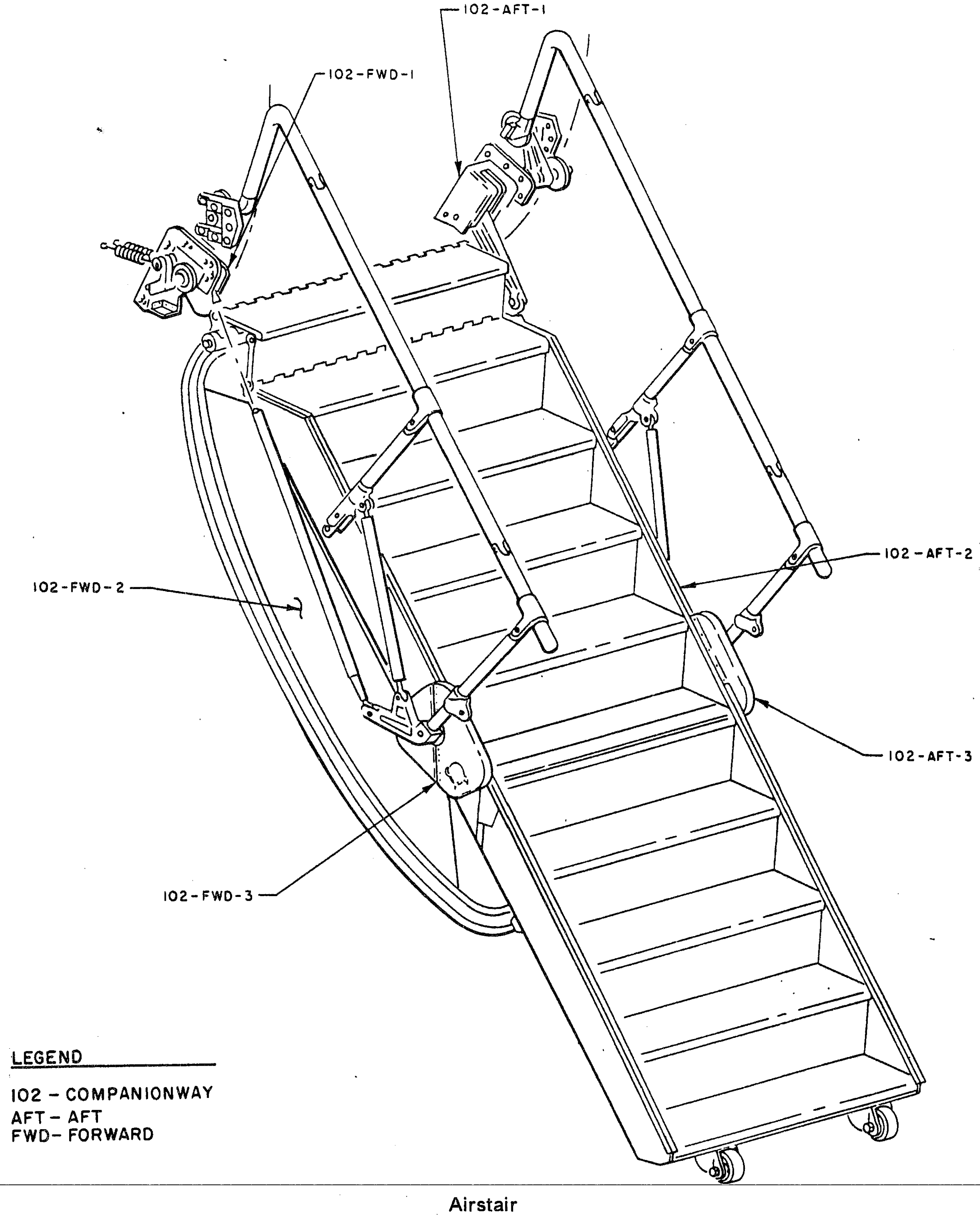
Floorboards
View Looking Down



Cockpit Access
View Looking Forward

Cockpit/Floorboard Arrangement
Figure 8.

ACCESS OPENINGS

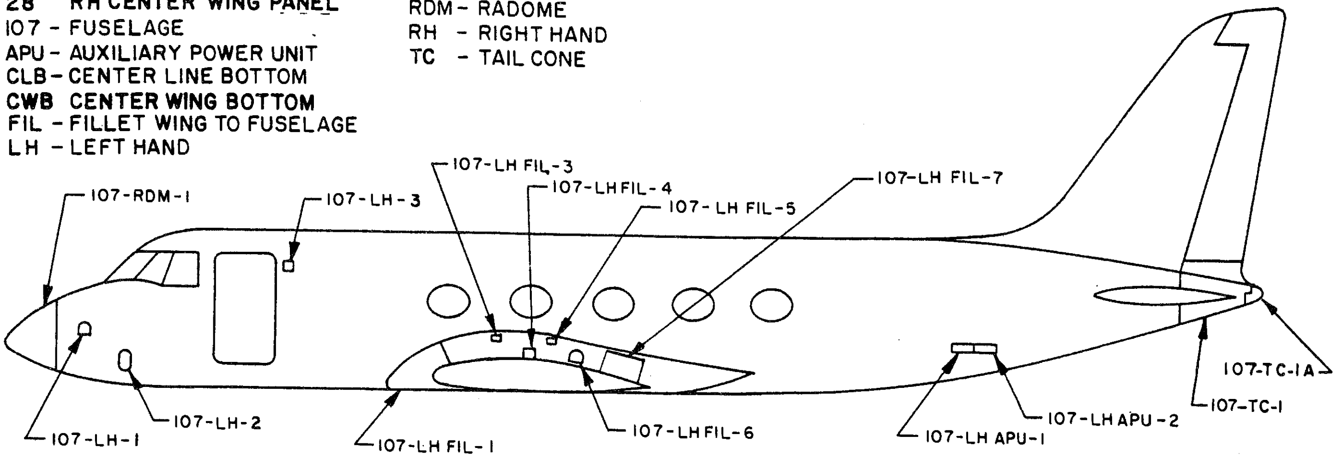


Airstair
Figure 9.

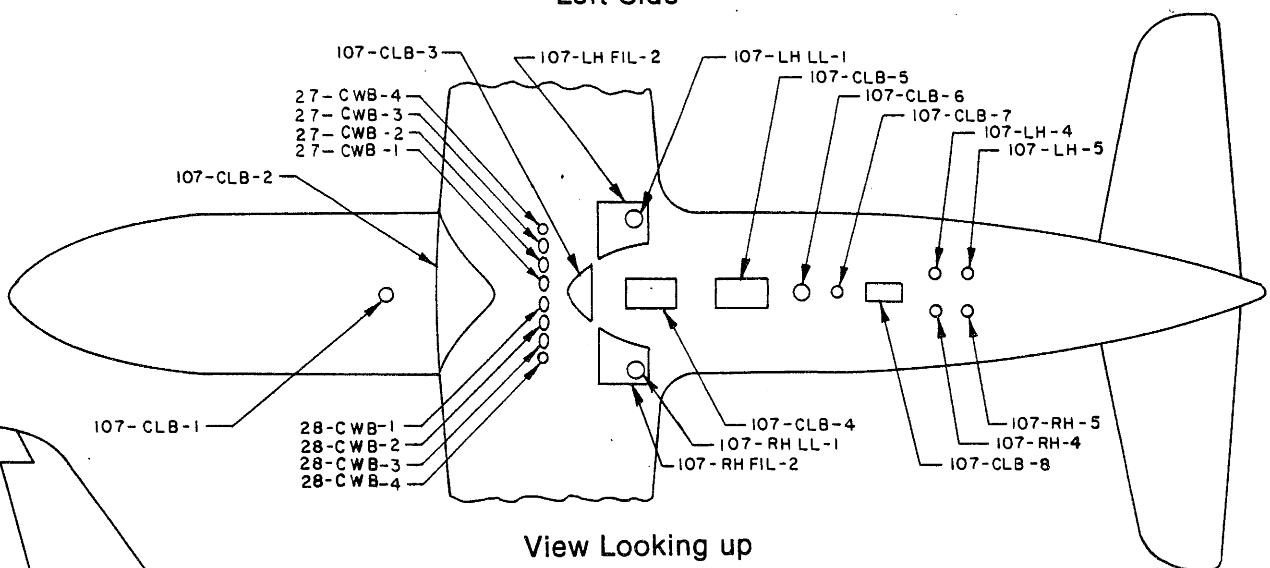
ACCESS OPENINGS

LEGEND

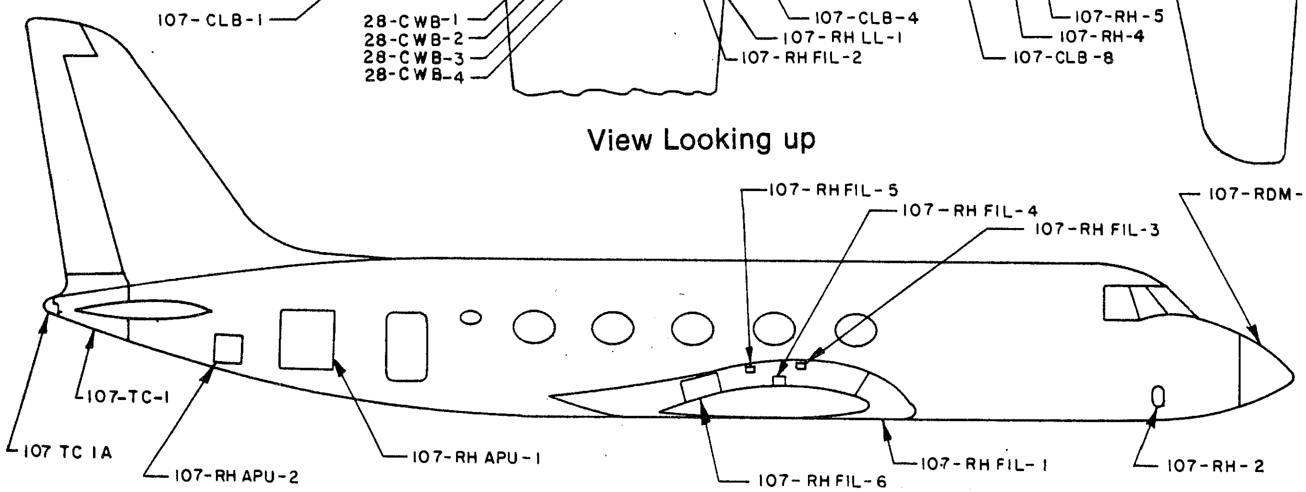
27 LH CENTER WING PANEL
28 RH CENTER WING PANEL
107 - FUSELAGE
APU - AUXILIARY POWER UNIT
CLB - CENTER LINE BOTTOM
CWB CENTER WING BOTTOM
FIL - FILLET WING TO FUSELAGE
LH - LEFT HAND
LL - LANDING LIGHT
RDM - RADOME
RH - RIGHT HAND
TC - TAIL CONE



Left Side



View Looking up



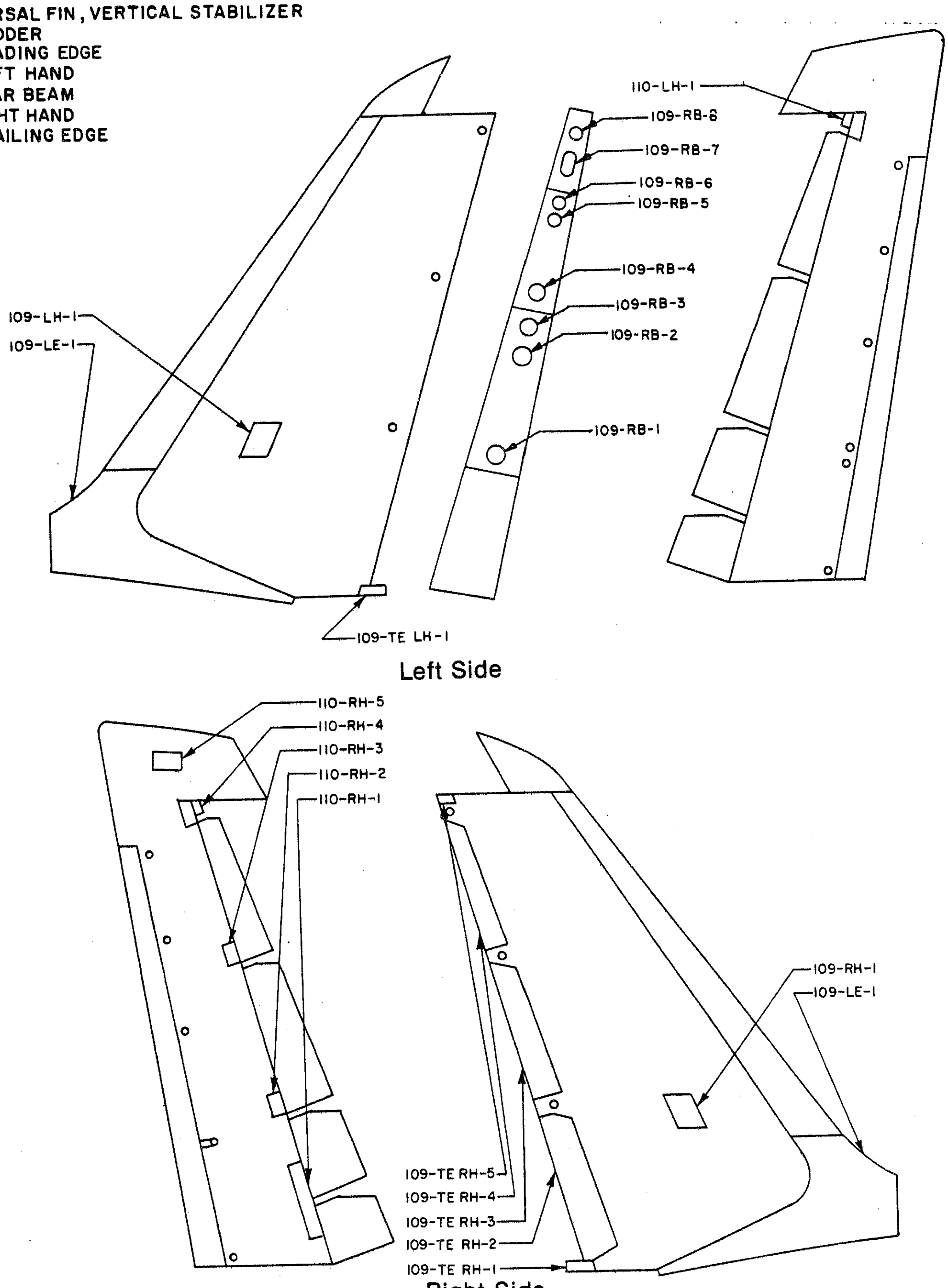
Right Side

Fuselage, Wing to Fuselage Fillet
Figure 10.

ACCESS OPENGINS

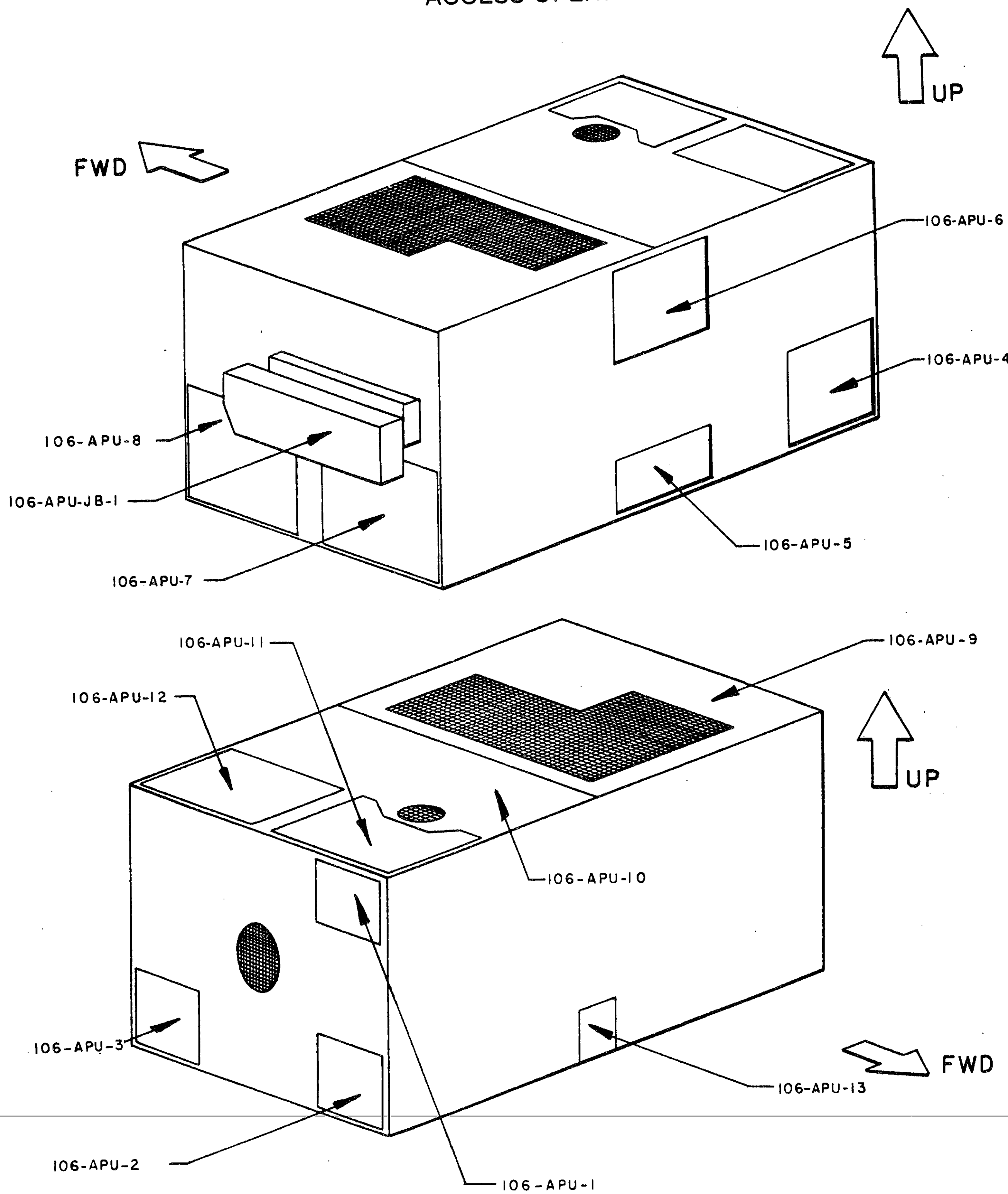
LEGEND

109 – DORSAL FIN, VERTICAL STABILIZER
110 – RUDDER
LE – LEADING EDGE
LH – LEFT HAND
RB – REAR BEAM
RH – RIGHT HAND
TE – TRAILING EDGE



Vertical Stabilizer/Rudder
Figure 11.

ACCESS OPENINGS



APU Enclosure
Figure 12

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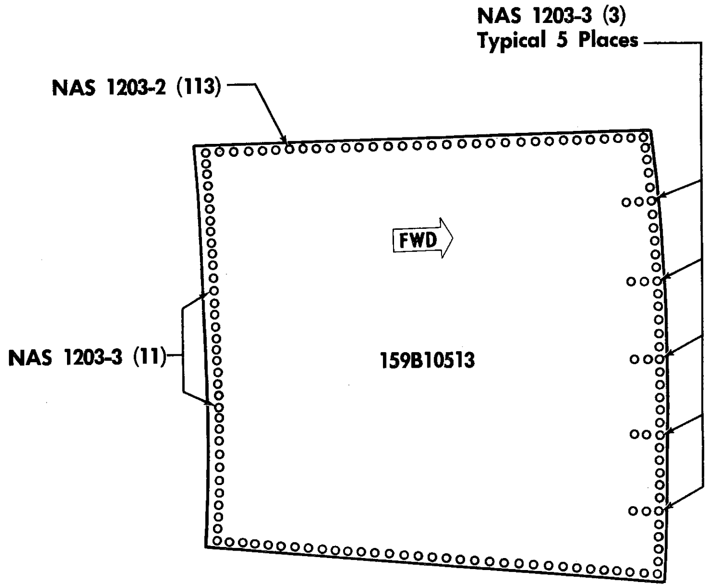
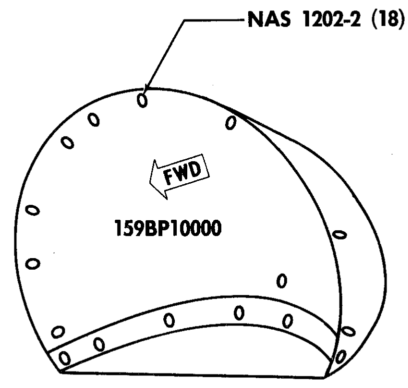
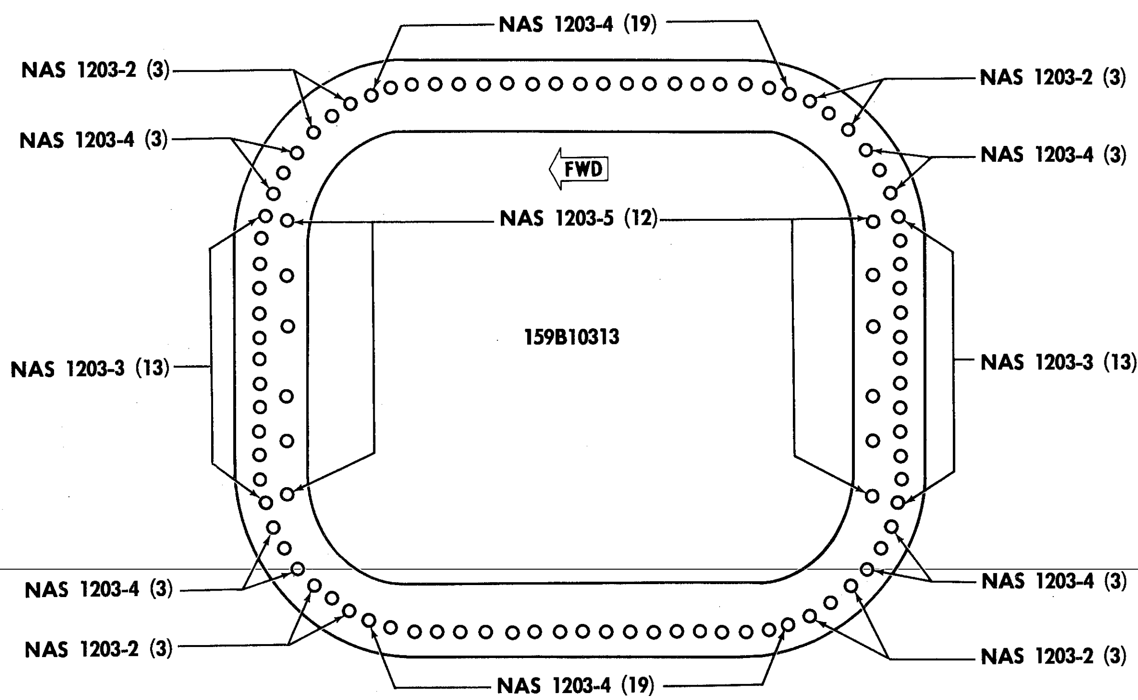
FUSELAGE ACCESS FASTENERS



GAS TURBINE COMPRESSOR
(RH-Only)



TAIL CAP COVER
Aft Section–Sta 753-759



ANTENNA COVERS
Bottom Sta 380-403
Bottom Sta 428-450

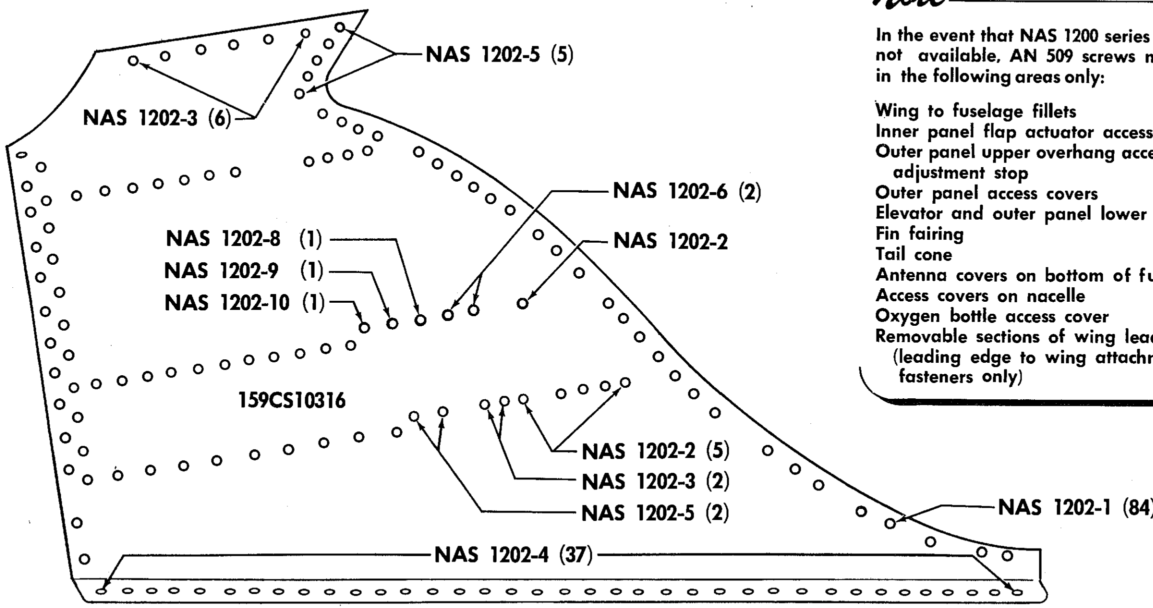
Shape of Access and Inspection Doors and Panels
Figure 3 (Sheet 9)

FIN TO FUSELAGE ACCESS FASTENERS

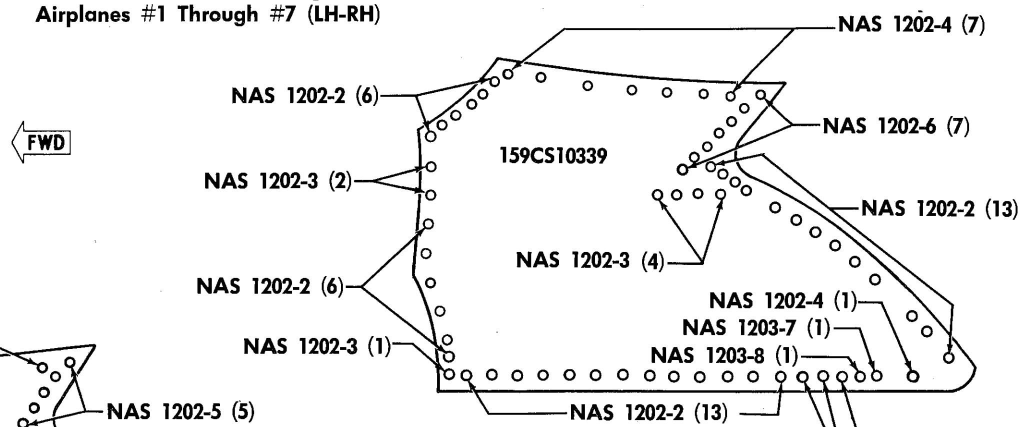
Note

In the event that NAS 1200 series screws are not available, AN 509 screws may be used in the following areas only:

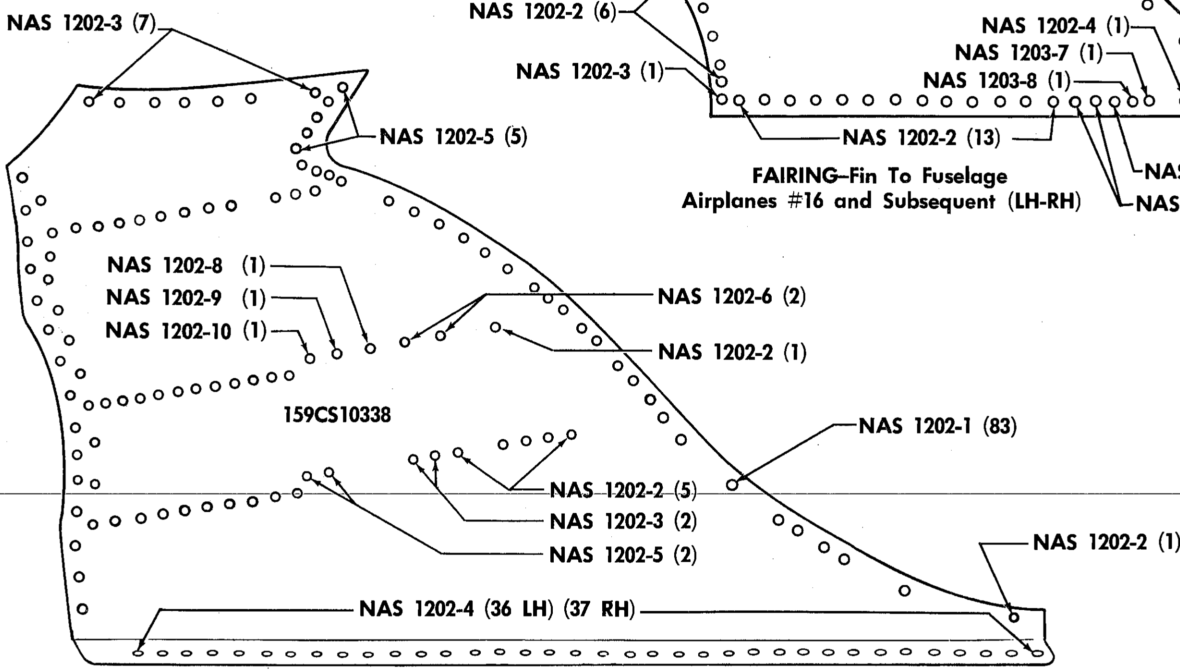
Wing to fuselage fillets
Inner panel flap actuator access
Outer panel upper overhang access to aileron adjustment stop
Outer panel access covers
Elevator and outer panel lower overhang
Fin fairing
Tail cone
Antenna covers on bottom of fuselage
Access covers on nacelle
Oxygen bottle access cover
Removable sections of wing leading edge (leading edge to wing attachment fasteners only)



FAIRING–Fin To Fuselage
Airplanes #1 Through #7 (LH-RH)



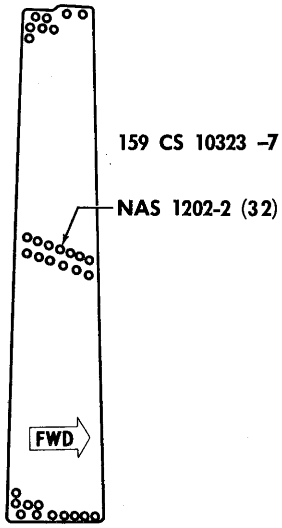
FAIRING–Fin To Fuselage
Airplanes #16 and Subsequent (LH-RH)



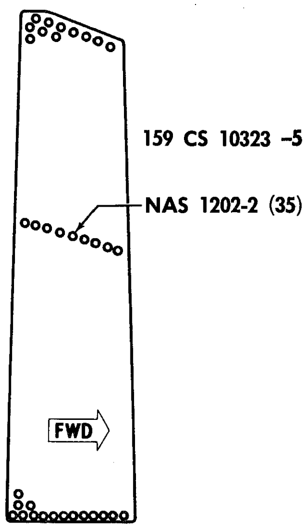
FAIRING–Fin To Fuselage
Airplanes #8 Through #15 (LH-RH)

Shape of Access and Inspection Doors and Panels
Figure 3 (Sheet 10)

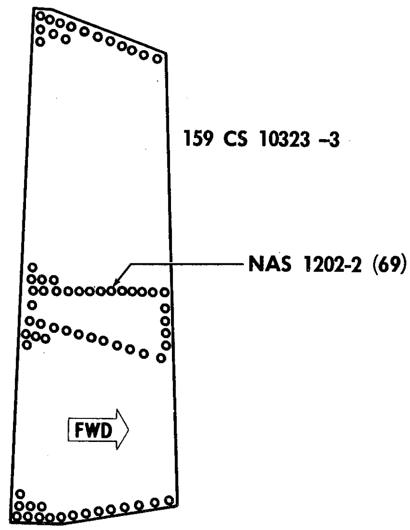
VERTICAL FIN ACCESS FASTENERS



RUDDER–TOP SEAL DOOR
(RH Only)



RUDDER–MID SEAL DOOR
(RH Only)

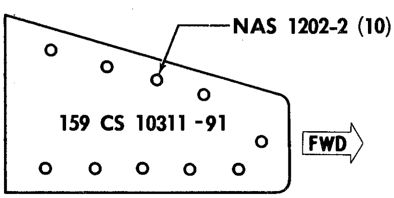


RUDDER–LOWER SEAL DOOR
(RH Only)

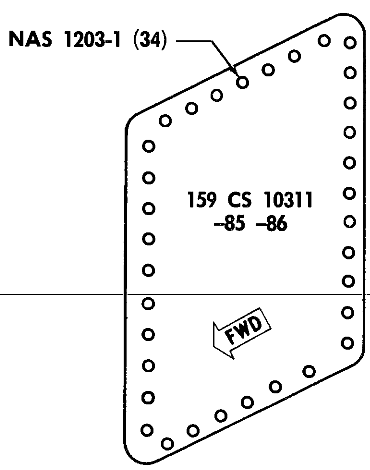
Note

In the event that NAS 1200 series screws are not available, AN 509 screws may be used in the following areas only:

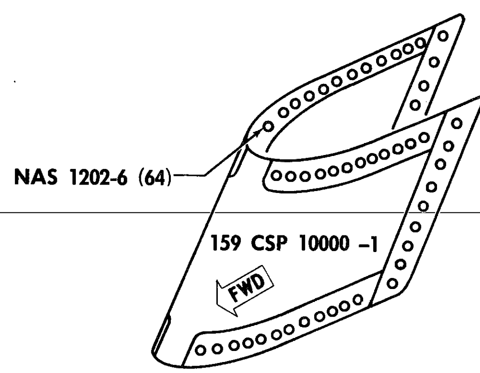
Wing to fuselage fillets
Inner panel flap actuator access
Outer panel upper overhang access to aileron adjustment stop
Outer panel access covers
Elevator and outer panel lower overhang
Fin fairing
Tail cone
Antenna covers on bottom of fuselage
Access covers on nacelle
Oxygen bottle access cover
Removable sections of wing leading edge (leading edge to wing attachment fasteners only)



NAVIGATION LIGHT
(RH Only)



TRIM TAB CABLE
(LH–RH)



FIN CAP
ANTI COLLISION LIGHT WINDOW

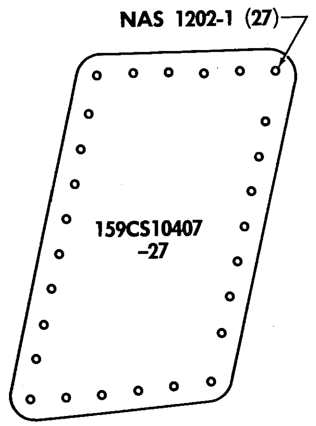
Shape of Access and Inspection Doors and Panels
Figure 3 (Sheet 11)

RUDDER ACCESS FASTENERS

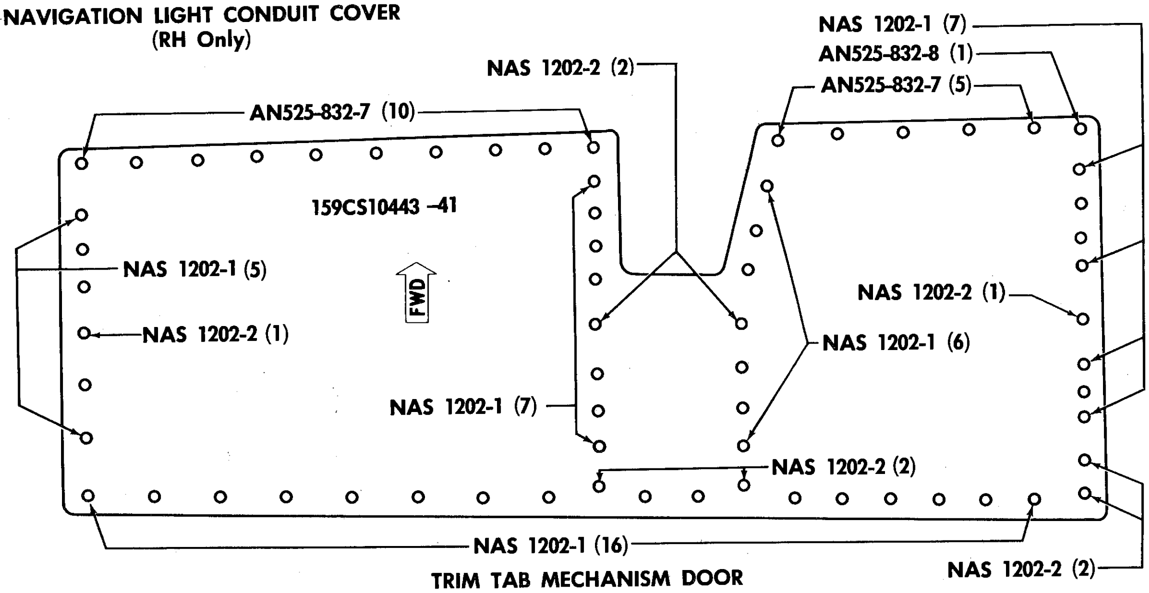
Note

In the event that NAS 1200 series screws are not available, AN 509 screws may be used in the following areas only:

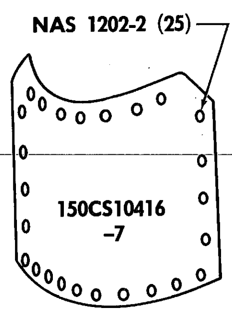
Wing to fuselage fillets
Inner panel flap actuator access
Outer panel upper overha ıg access to aileron adjustment stop
Outer panel access covers
Elevator and outer panel lower overhang
Fin fairing
Tail cone
Antenna covers on bottom of fuselage
Access covers on nacelle
Oxygen bottle access cover
Removable sections of wing leading edge (leading edge to wing attachment fasteners only)



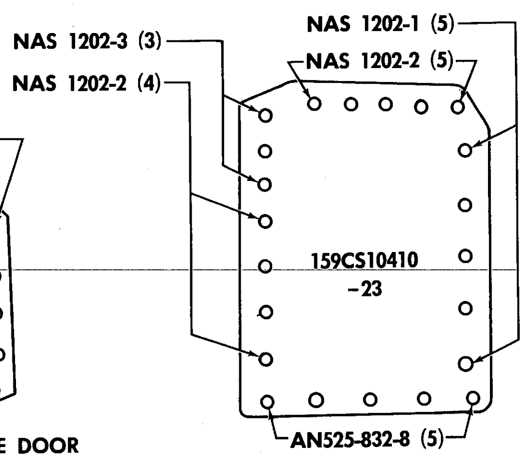
NAVIGATION LIGHT CONDUIT COVER
(RH Only)



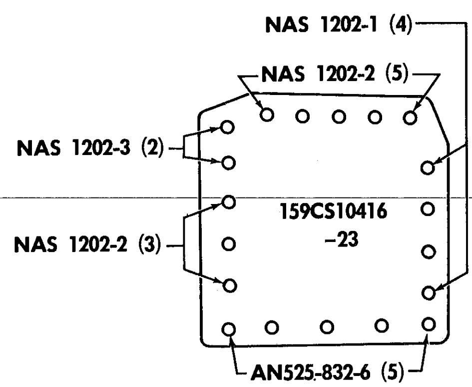
TRIM TAB MECHANISM DOOR
(RH Only)



UPPER RUDDER HINGE DOOR



RUDDER HINGE DOOR
Rudder Sta 68-74 (RH Only)



RUDDER HINGE DOOR
Rudder Sta 111-117 (RH Only)

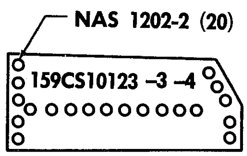
Shape of Access and Inspection Doors and Panels
Figure 3 (Sheet 12)

HORIZONTAL STABILIZER ACCESS FASTENERS

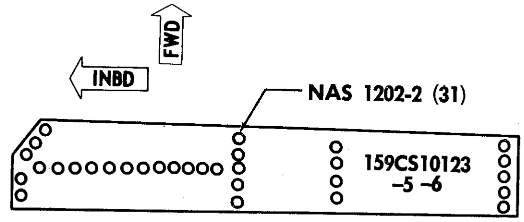
Note

In the event that NAS 1200 series screws are not available, AN 509 screws may be used in the following areas only:

Wing to fuselage fillets
Inner panel flap actuator access
Outer panel upper overhang access to aileron adjustment stop
Outer panel access covers
Elevator and outer panel lower overhang
Fin fairing
Tail cone
Antenna covers on bottom of fuselage
Access covers on nacelle
Oxygen bottle access cover
Removable sections of wing leading edge (leading edge to wing attachment fasteners only)

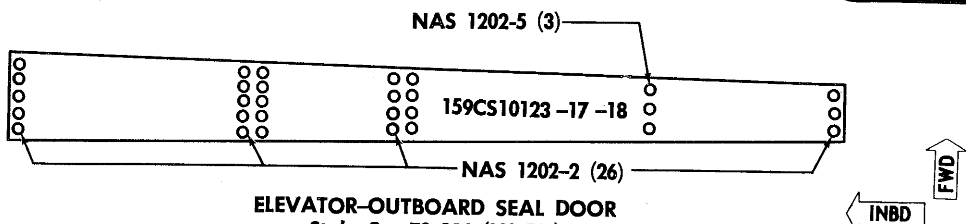


ELEVATOR–INBOARD SEAL DOOR
Stab. Sta 20–34 (LH–RH)

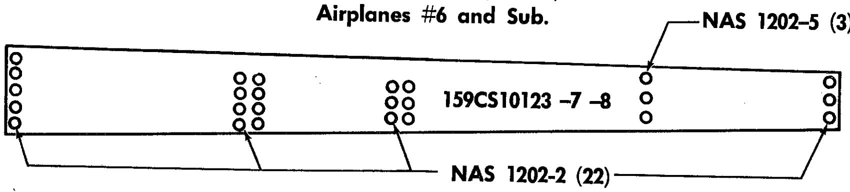


ELEVATOR–MID SEAL DOOR
Stab. Sta 34-70 (LH–RH)

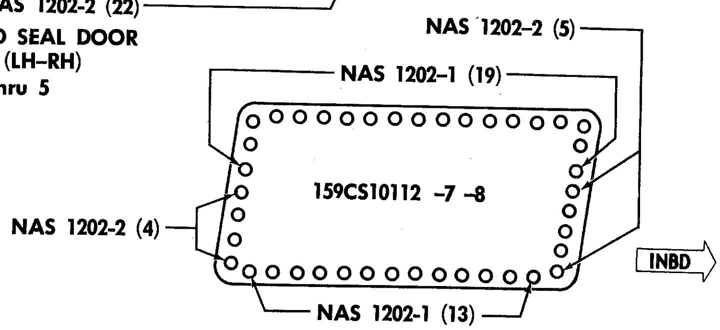
LH Looking Up



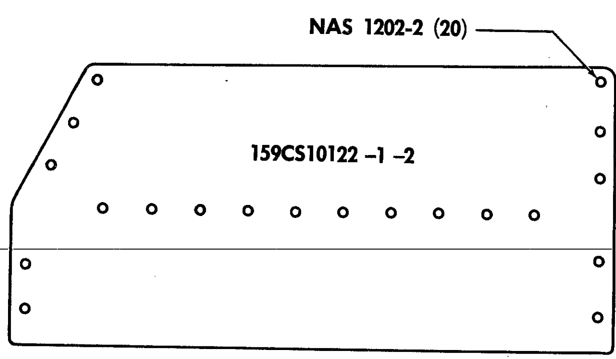
ELEVATOR–OUTBOARD SEAL DOOR
Stab. Sta 70–136 (LH–RH)
Airplanes #6 and Sub.



ELEVATOR–OUTBOARD SEAL DOOR
Stab. Sta 70–136 (LH–RH)
Airplanes #1 Thru 5

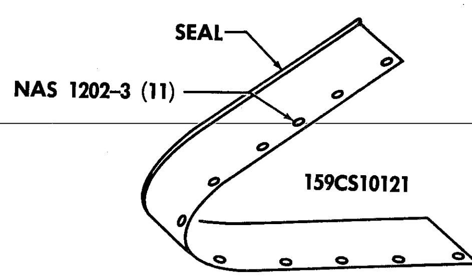


TRIM TAB CABLES BELLCRANK DOOR
Stab. Sta 46–58 (LH–RH)



LH Looking Down

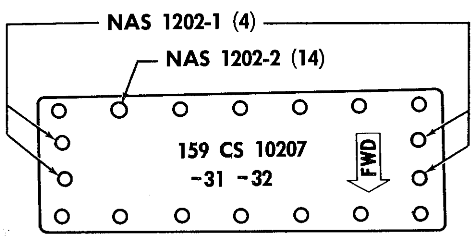
ELEVATOR–BOLT ADJUSTMENT DOOR
Stab. Sta 20–33 (LH–RH)



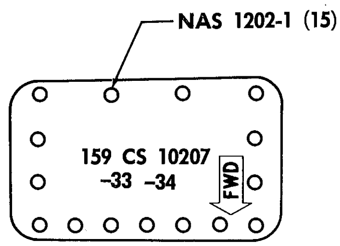
INBOARD NOSE SKIN
(LH–RH)

Shape of Access and Inspection Doors and Panels
Figure 3 (Sheet 13)

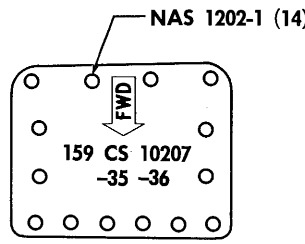
ELEVATOR ACCESS FASTENERS



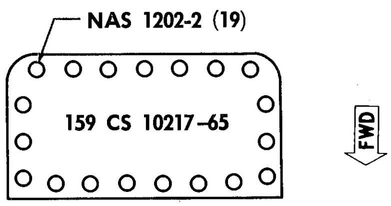
TAB ACTUATOR DOOR
Upper Surface (LH–RH)



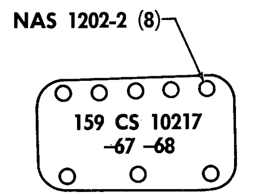
TAB INBOARD PUSH ROD DOOR
Upper Surface (LH–RH)



TAB OUTBOARD PUSH ROD DOOR
Upper Surface (LH–RH)



HINGE BOLT TRIM TAB DOOR
Lower Surface (2L–2R)

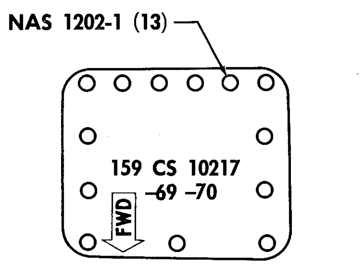


OUTBOARD HINGE DOOR
Lower Surface (LH–RH)

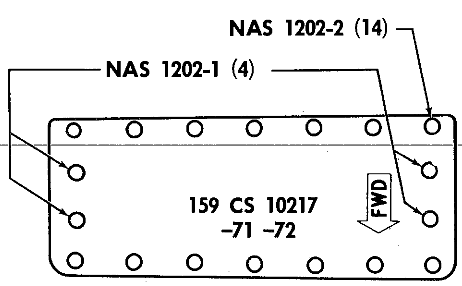
Note

In the event that NAS 1200 series screws are not available, AN 509 screws may be used in the following areas only:

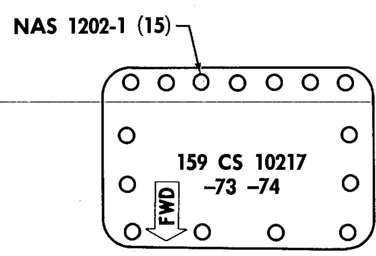
Wing to fuselage fillets
Inner panel flap actuator access
Outer panel upper overhang access to aileron adjustment stop
Outer panel access covers
Elevator and outer panel lower overhang
Fin fairing
Tail cone
Antenna covers on bottom of fuselage
Access covers on nacelle
Oxygen bottle access cover
Removable sections of wing leading edge (leading edge to wing attachment fasteners only)



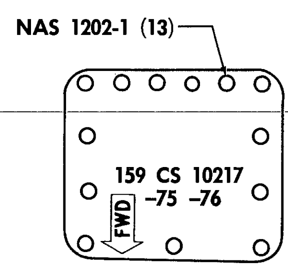
TAB OUTBOARD PUSH ROD DOOR
Lower Surface (LH–RH)



TAB ACTUATOR DOOR
Lower Surface (LH–RH)



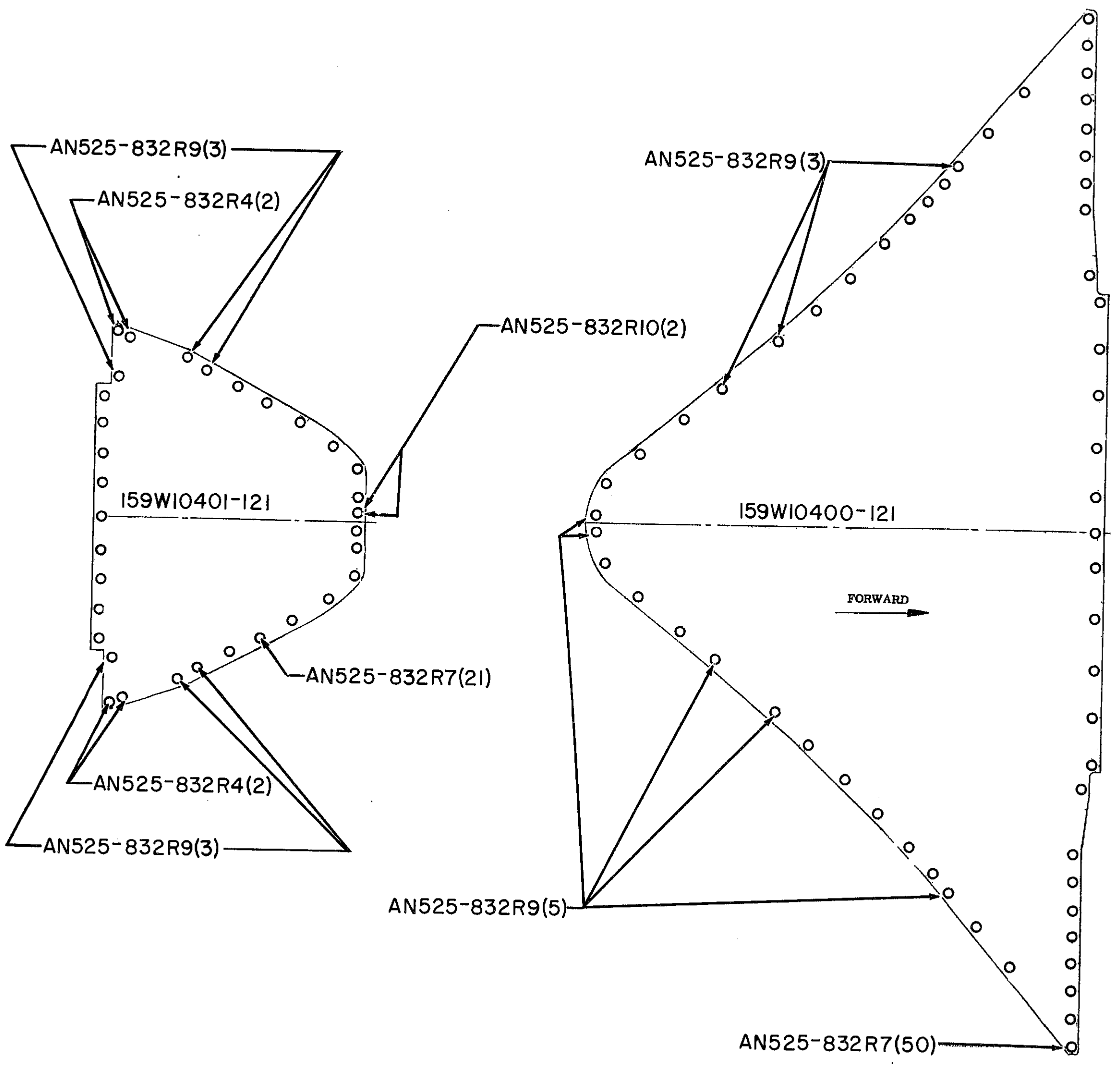
TAB INBOARD PUSH ROD DOOR
Lower Surface (LH–RH)



INBOARD HINGE DOOR
Lower Surface (LH–RH)

Shape of Access and Inspection Doors and Panels
Figure 3 (Sheet 14)

WING TO FUSELAGE ACCESS FASTENERS



WING TO FUSELAGE CLOSURE FAIRINGS
LOWER SURFACE

Shape of Access and Inspection Doors and Panels FAA #177 & Subs.
(Except 322 & 323) and FAA #1-176, 322 & 323 Modified by ASC 183
Figure 3 (Sheet 15)

TOOLS AND EQUIPMENT — SPECIAL SUPPORT EQUIPMENT

PART NO.	NOMENCLATURE	APPLICATION
GT7	Wrench Cabin Hood Spacer Nut G372	Used to install or remove G372 Nut.
GT452	Puller Piano Hinge Wire	Required for piano hinge wire for various doors and control surfaces.
98GT1024	Driver (5/32 inch diameter hole) Piano Hinge Wire	Required for piano hinge wire on various doors and control surfaces.
98GT1030	Ground Safety Lock Main Landing Gear	Attach to special fitting on upper end, outboard side of each drag brace, to lock gear down. flow required.)
159GT1002	Jack Pad Fuselage Forward Section (Sta 120.10)	Attach to two pickup holes in forward fuselage section at sta 120.10.
159GT1003	Adapter Wing / Fuel Cell / Sealant Gun	Used in conjuction with Model 223 Grover Smith sealant gun and SEMCO sealant gun to apply wing fuel cell sealant compound.
159GT1004	Work Stand Rolls Royce Dart RDa 7/2 - Engine	Procure only 159GT1004-T1. Used as work stand and also to transport engine at a maximum speed of 5 mph.
159GT1005	Throw Board Elevator (Stab. Sta 82)	Used to check throws on stabilizer at Stab. Sta 82.
159GT1006	Throw Board Aileron (Wing Sta. 307)	Used to check throws on aileron wing Sta 307.
159GT1007	Ground Safety Lock Hydraulics - Manual Selector Valve Nose Wheel Door	Used to lock the manual door selector valve (Whittaker 102463) in door open position. A/C having ASC 109 is, use to lock manual AUX. TO MAIN selector valve (Adel 55963) in ON position.
159GT1009	Wrench - Combination Vickers Engine Mount, Spanner and Torquing	Used to start, tighten and torque Vickers engine mount bolts. (Limited to the engine assemblies built up by Vickers - A/P No. 1 only.)
159GT1011	Ground Safety Strut Nose Wheel Door	Used to keep nose wheel doors in open position and to prevent inadvertent closure of doors during ground handling of aircraft.
159GT1013	Duct Plug Rolls-Royce Dart RDa 7/2 - Engine Oil Cooler Intake	Used for "shop use only", to protect intake to oil cooler duct from foreign matter (use Rolls-Royce full inlet duct cover RK20538 for field use).

PART NO.	NOMENCLATURE	APPLICATION
159GT1014	Door Strut Main Landing Gear Doors	Used to keep main landing gear doors in open position and to prevent inadvertent closure of doors during ground handling of aircraft.
159GT1015	Ground Safety Lock Nose Landing Gear	Used to prevent nose landing Gear from collapsing.
159GT1016	Hosting Sling Accessory Gearbox - Installation and Removal	Used to install and remove accessory gearbox Rotol SKA3407.
159GT1018	Hoisting Sling Stabilizer and Stabilizer Elevator Assembly.	Used to hoist either half of stabilizer and stabilizer elevator assembly.
159GT1019	Socket Wrench Main Landing Gear Shock Strut Metering Pin	Used on main landing gear metering pin.
159GT1020	Socket Wrench Nose Landing Gear Shock Strut Metering	Used on nose landing gear metering pin.
159GT1021	Puller MLG - Shock Strut Metering Pin-Diaphragm	Used to remove Metering Pin Diaphragm from inner cylinder on Main Landing Gear.
159GT1022	Puller NLG - Shock Strut Metering Pin - Diaphragm	Used to remove metering pin diaphragm from inner cylinder on nose landing gear.
159GT1025	Cover Tail Pipe Duct	Used to cover tail pipe duct exit. Two required.
159GT1026	Cover Engine	Used on Rolls-Royce Engine to cover air intake area. P/N-RK20538 may be used in its place. Two required.
159GT1027	Spring Compressor Nose Wheel Strut	Used to compress C101-849 and G101-850 spring for installation or removal from 117C 10098 bungee installation.
Item		
GT181	Puller Fuel Cell Access Cover	Used to hold water methanol cover PIN 159PM10029-1 in left and right wings, while securing screw.
159GT1034	Locking Plate Rolls-Royce Engine Power Lever Control System - Rigging	Used to lock Rotol SNC/4B correction motor in datum position for rigging Rolls-Royce engine power lever control system right engine.
159GT1037	Propeller Stand Rotol Prop RA59544	Handles one propeller, both in hangar and in the field.
159GT1038	Strap Tail Pipe Blanket Installation	Used to install tail pipe blanket.
159GT1039	Throwboard Rudder	Used at Vertical Fin W.L. 171 to check rudder throw.

PART NO.	NOMENCLATURE	APPLICATION
159GT1040	Screw Driver (Spec.) Cabin Side Window Roller	Used to install or remove side cabin window roller, Torrington, No. 3CF-11 or 14. (Six rollers per cabin.)
P71911	Adapter Assembly Forward Section - Lifting	To raise forward section of aircraft to allow tail to be lowered for hangar clearance.
P72786	Support Fixture Fuselage Station 561	To support tail section at Fuselage Station 561 while personnel are working aft in fuselage.
159GT1042	Cover Air Conditioning - Ram Air	Used to cover ram air outlet louver on left side to protect from foreign matter. (Plastic)
159GT1043	Cover Air Conditioning - Cabin Outflow Louvers	Used to cover twin outflow louvers on right side to protect from foreign matter. (Plastic)
159GT1044	Plug Turbine Gas Compressor - Exhaust	(Aircraft with compressor only) Used to plug turbine gas compressor exhaust outlet in rear right door to protect from foreign matter. (Plastic)
159GT1046	Plug Air Conditioning - Ram Air Scoop Inlet	Used to plug ram air, inlet scoop to protect from foreign matter. (Left and right required.)
159GT1047	Cover Pitot Tube	Used to cover pitot tube to protect from foreign matter. (Two required)
159GT1049	Tow Bar Nose Wheel	Attaches to nose wheel lower strut.
159GT1050	Rigging Template Flap Drive Link - Inboard and Outboard	Used to adjust the flap drive links, inboard and outboard respectively. Two of each (-3 and -5) are required for rigging.
159GT1051	Contour Board Fin and Rudder	Used with 159GT1039 throw board to check rudder at zero.
159GT1052	Centering Fixture Nose Wheel - Steering	Used for setting up the centering torque link steering and nose wheel alignment.
159GT1053	Adapter Stewart Warner - Lubrication Fitting	Used on standard alemite gun when lubricating various wing hinge and landing gear fittings.
159GT1054	Jury Cable Air Stair - Support	Used to support airstair when towing aircraft.
159GT1055	Kit Nose Landing Gear Bushing - Installation and Removal	Used to remove the nose landing gear yoke bushing.
159GT1056	Plug Engine	Used to protect the ram air scoop inlet from foreign matter. (Inlet on right side of right engine only.)

PART NO.	NOMENCLATURE	APPLICATION
159GT1057	Cover Propeller Blade	Used to protect the propeller blades from damage and foreign matter. (Four covers required for each propeller.)
159GT1058	Hoisting Sling-Engine	Used to hoist Rolls-Royce Dart 529 type engine with or without propeller.
159GT1059	Adapter Oxygen System to Supply	Used to adapt aircraft oxygen system to the ground oxygen filling system.
159GT1060	Spanner Wrench Fuel Tank Intercon-nector Retaining Nut	Used on 159PM10305-11 interconnector nut at Wing Station 348 1/2 and 364. (Aircraft having ASC 125)
159GT1061	Spanner Wrench Fuel Tank Intercon-nector Retaining Nut	Used on 159PM10306-11 interconnector nut at Wing Station 348 1/2 and 364. (Aircraft having ASC 125)
159FT1062	Spanner Wrench Fuel Tank Intercon-nector Retaining Nut	Used on 159PM10302-11 interconnector nut at Wing Station 331, 402 5/16, and 382 1/2. (Aircraft having ASC 125)
159GT1063	Spanner Wrench Fuel Tank Intercon-nector Retaining Nut	Used on 159PM10311-11 interconnector nut at Wing Sta 440 9/16 left and right, 414 9/16, 421 3/4 right only. Aircraft having ASC 125.
159GT1064	Spanner Wrench Fuel Tank Intercon-nector Retaining Nut	Used on 159PM10315-11 interconnector nut at Wing Sta 414 9/16 (left wing only). (Air-craft having ASC 125.
159GT1065	Spanner Wrench Fuel Tank Intercon-nector Retaining Nut	Used on 159PM10332 interconnector nut at Wing Sta 307. (Aircraft having ASC 125)
159GT1066	Spanner Wrench Fuel Tank Intercon-nector Retaining Nut	Used on 159PM10315-11 interconnector nut at Wing Sta 421 3/4 (left wing only.(Aircraft having ASC 125).
159GT1067	Spanner Wrench Fuel Tank Intercon-nector Retaining Nut	Used on 159PM10302-11 interconnector nut at Wing Sta 313 1/4. (Aircraft having ASC 125.)
159GT1068	Hoisting Sling Rudder Assembly	Used to hoist rudder assembly.
159SEM10002	Compressor Assembly (Decelostat)	Used to replace tires on decelostat fly-wheel assemblies.

Item Number	Part Number	Name	Remarks
(25)	GT181	Puller Fuel Cell Access Cover	Used to hold water methanol cover P/N 159PM10029-1 in left and right wings, while securing screw.
(26)	159GT1034	Locking Plate Rolls-Royce Engine Power Lever Control System-Rigging	Used to lock Rotol SNC/4B correction motor in datum position for rigging Rolls-Royce Engine Power Lever Control System Right Engine.
(27)	159GT1037	Propeller Stand Rotol Prop RA59544	Handles one propeller, both in hangar and in the field.
(28)	159GT1038	Strap Tail Pipe Blanket Installation	Used to install Tail Pipe Blanket.
(29)	159GT1039	Throwboard Rudder	Used at Vertical Fin W.L. 171 to check rudder throw.
(30)	159GT1040	Screw Driver (Spec.) Cabin Side Window Roller	Used to install or remove Side Cabin Window Roller, Torrington, No. 3CF-11 or 14. (Six rollers per cabin.)
(31)	P71911	Adapter Assembly Forward Section - Lifting	To raise forward section of aircraft to allow tail to be lowered for hangar clearance.
(32)	P72786	Support Fixture Fuselage Station 561	To support tail section at Fuselage Station 561 while personnel are working aft in fuselage.
(33)	159GT1042	Cover Air Conditioning - Ram Air	Used to cover ram air outlet louver on left side to protect from foreign matter. (Plastic)
(34)	159GT1043	Cover Air Conditioning - Cabin Outflow Louvers	Used to cover twin outflow louvers on right side to protect from foreign matter. (Plastic)
(35)	159GT1044	Plug Turbine Gas Compressor - Exhaust	(Aircraft with compressor only) Used to plug turbine gas compressor exhaust outlet in rear right door to protect from foreign matter. (Plastic)
(36)	159GT1046	Plug Air Conditioning - Ram Air Scoop Inlet	Used to plug ram air, inlet scoop to protect from foreign matter. (Left and right required.)
(37)	159GT1047	Cover Pitot Tube	Used to cover pitot tube to protect from foreign matter. (Two required)

Item Number	Part Number	Name	Remarks
(38)	159GT1049	Tow Bar Nose Wheel	Attaches to nose wheel lower strut.
(39)	159GT1050	Rigging Template Flap Drive Link - Inboard and Outboard	Used to adjust the flap drive links, inboard and outboard respectively. Two of each (-3 and -5) are required for rigging.
(40)	159GT1051	Contour Board Fin and Rudder	Used with 159GT1039 throw board to check rudder at zero.
(41)	159GT1052	Centering Fixture Nose Wheel - Steering	Used for setting up the centering torque link steering and nose wheel alignment.
(42)	159GT1053	Adapter Stewart Warner - Lubrication Fitting	Used on standard alemite gun when lubricating various wing hinge and landing gear fittings.
(43)	159GT1054	Jury Cable Air Stair - Support	Used to support airstair when towing aircraft.
(44)	159GT1055	Kit Nose Landing Gear Bushing - Installation and Removal	Used to remove the nose landing gear yoke bushing.
(45)	159GT1056	Plug Engine	Used to protect the ram air scoop inlet from foreign matter. (Inlet on right side of right engine only.)
(46)	159GT1057	Cover Propeller Blade	Used to protect the propeller blades from damage and foreign matter. (Four covers required for each propeller.)
(47)	159GT1058	Hoisting Sling-Engine	Used to hoist Rolls-Royce Dart 529 type engine with or without propeller.
(48)	159GT1059	Adapter Oxygen System to Supply	Used to adapt aircraft oxygen system to the ground oxygen filling system.
(49)	159GT1060	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10305-11 interconnector nut at Wing Station 348-1/2 and 364. (Aircraft 1–200, including 322 and 323 having ASC 125)
(50)	159GT1061	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10306-11 interconnector nut at Wing Station 348-1/2 and 364. (Aircraft 1–200, including 322 and 323 having ASC 125)
(51)	159FT1062	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10302-11 interconnector nut at Wing Station 331, 402-5/16, and 382-1/2. (Aircraft 1–200, including 322 and 323 having ASC 125)

Item Number	Part Number	Name	Remarks
(52)	159GT1063	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10311-11 interconnector nut at Wing Sta 440-9/16 left and right, 414-9/16, 421-3/4 right only. Aircraft 1 through 200, including 322 and 323, having ASC 125 incorporated.
(53)	159GT1064	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10315-11 interconnector nut at Wing Sta 414-9/16 (left wing only). (Aircraft 1 through 200 including 322 and 323 having ASC 125 incorporated
(54)	159GT1065	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10332 interconnector nut at Wing Sta 307. (Aircraft 1 through 200 including 322 and 323 having ASC 125 incorporated.)
(55)	159GT1066	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10315-11 interconnector nut at Wing Sta 421-3/4 (left wing only.(Aircraft 1 through 200 including 322 and 323, having ASC 125 incorporated.)
(56)	159GT1067	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10302-11 interconnector nut at Wing Sta 313-1/4. (Aircraft 1 through 200 including 322 and having ASC 125 incorporated.)
(57)	159GT1068	Hoisting Sling Rudder Assembly	Used to hoist Rudder Assembly.
(58)	159SEM10002	Compressor Assembly (Decelostat)	Used to replace tires on decelostat flywheel assemblies.

Item Number	Part Number	Name	Remarks
(35)	159GT1044	Plug Turbine Gas Compressor-Exhaust	(Aircraft with compressor only) Used to plug turbine gas compressor exhaust outlet in rear right door to protect from foreign matter. (Plastic)
(36)	159GT1046	Plug Air Conditioning - Ram Air Scoop Inlet	Used to plug ram air, inlet scoop to protect from foreign matter. (Left and right required)
(37)	159GT1047	Cover Pitot Tube	Used to cover pitot tube to protect from foreign matter. (Two required)
(38)	159GT1049	Tow Bar Nose Wheel	Attaches to nose wheel lower strut.
(39)	159GT1050	Rigging Template Flap Drive Link - Inboard and Outboard	Used to adjust the flap drive links inboard, and outboard respectively. Two of each (-3 and -5) are required for rigging.
(40)	159GT1051	Contour Board Fin and Rudder	Used with 159GT1039 throw board to check rudder at zero.
(41)	159GT1052	Centering Fixture Nose Wheel - Steering	Used for setting up the centering torque link steering and nose wheel alignment.
(42)	159GT1053	Adapter Stewart Warner - Lubrication Fitting	Used on standard alemite gun when lubricating various wing hinge and landing gear fittings.
(43)	159GT1054	Jury Cable Air Stair - Support	Used to support air stair when towing aircraft.
(44)	159GT1055	Kit Nose Landing Gear Bushing - Installation and Removal	Used to remove the nose landing gear yoke bushing.
(45)	159GT1056	Plug Engine	Used to protect the ram air scoop inlet from foreign matter. (Inlet on right side of right engine only.)
(46)	159GT1057	Cover Propeller Blade	Used to protect the propeller blades from damage and foreign matter. (Four covers required for each propeller.)
(47)	159GT1058	Hoisting Sling Engine	Used to hoist Rolls Royce Dart 529 type engine with or without propeller.

Item Number	Part Number	Name	Remarks
(48)	159GT1059	Adapter Oxygen System to Supply	Used to adapt aircraft oxygen system to the ground oxygen filling system.
(49)	159GT1060	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10305-11 interconnector nut at Wing Sta 348-1/2 and 364. (For additional fuel provisioning only ASC 125.)
(50)	159GT1061	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10306-11 interconnector nut at Wing Sta 348-1/2 and 364. (For additional fuel provisioning only ASC 125.)
(51)	159GT1062	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10302-11 interconnector nut at Wing Sta 331, 402-5/16, and 382-1/2. (For additional fuel provisioning only.)
(52)	159GT1063	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10311-11 interconnector nut at Wing Sta 440-9/16 left and right, 414-9/16, 421-3/4 right only. (For additional fuel provisioning only.)
(53)	159GT1064	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10315-11 interconnector nut at Wing Sta 414-9/16 (left wing only). (For additonal fuel provisioning only.)
(54)	159GT1065	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10332 interconnector nut at Wing Sta 307. (For additional fuel provisioning only.)
(55)	159GT1066	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10315-11 interconnector nut at Wing Sta 421-3/4 (left wing only). (For additional fuel provisioning only.)
(56)	159GT1067	Spanner Wrench Fuel Tank Interconnector Retaining Nut	Used on 159PM10302-11 interconnector nut at Wing Sta 313-1/4. (For additional fuel provisioning only.)
(57)	159GT1068	Hoisting Sling Rudder Assembly	Used to hoist Rudder Assembly.
(58)	159SEM10002	Compressor Assembly (Decelostat)	Used to replace tires on decelostat flywheel assemblies.

C. Rolls Royce Recommended Tools.

NOTE: The following tools are listed for information only and are quoted by the engine manufacturer as required for engine maintenance.

(1) Standard Tools. (Obtain through local tool supplier)

(a) Socket Wrenches

Socket Wrenches	Quantity
2 BA	1
3/16" BSF	1
1/14" BSF	1
5/16" BSF	1
3/8" BSF	1
7/16" BSF	1

Plus extension bars, ratchet handles and universal joints.

(b) Open End Wrenches

Open End Wrenches	Quantity
1/4" AF	1
7/16" x 1/2" AF	1
1/2" x 9/16" AF	1
7/8" x 15/16" AF	1
3 BA x 2 BA	2
4 BA x 5 BA	1
3/16" x 1/4" BSF	2
5/16" x 3/8" BSF	1
3/8" x 7/16" BSF	1
7/16" x 1/2" BSF	1
1/2" x 9/16" BSF	1
9/16 x 5/8" BSF	1
5/8" x 11/16" BSF	1
3/4" x 11/16" BSF	1

(c) Miscellaneous

Miscellaneous	Quantity
Hammer, Fibre Tipped	1
Screwdriver, 8"	1
Screwdriver, 4"	1
Screwdriver, Offset	1
Pliers, Combination 6"	1
Pliers, Sidecutting 6"	1
Pliers, Long Nose	1
Torque Wrench, 0-600 Inch Pounds	1

(2) Rolls Royce Engine Service Tools: (Obtain from Rolls Royce or authorized Rolls Royce Agency)

(a) Dart Maintenance Tool Kit AK966A.

(3) Rolls Royce Special Tools and Equipment. (Obtain from Rolls Royce or authorized Rolls Royce Agency)

(a) Special tools for use on splined oil tubes:

Tool	Part #
Spanner, Splined, 3/4" Splines	K10707
Spanner, Splined, 7/8" Splines	K10708
Spanner, Splined, 1-1/8" Splines	K10710

(b) Special tools for use on oil cooler air duct.

Tool	Part #
Spanner, Holding Assembly for bolts securing Oil Cooler Air Duct to Air Intake Casing	RK. 1014

(c) Miscellaneous Tools

Tool	Part #
Servicing Kit (Shock Loading Tools)	RK. 23182
Tee Fitting - Burner Pressure Check	CAT. 11874

(c) Miscellaneous Tools (Continued)

Tool	Part #
Engine Sling	J. 54595
Jointing Compound - 4 oz. Tubes	SQ. 32L
Vulcaprene Service Repair Kit - Nose Cowl	SQ. 33
Vulcaprene Service Repair Kit - L. T. Harness	SQ. 34

(4) Wrenches required for Vickers Ltd. attachments to Rolls Royce Dart Engine (can be obtained from source indicated or equivalent).

Tool	Part # & Mfg.
Engine Mount Nuts (0. 713) Flats Dimension	"Snap On" WSW-102
Engine Mount Nuts (1. 102) Flats Dimension	"Snap On" WSW-106
Fire Protection Line (1. 187) Flats Dimension	"Snap On" S-3842
Heat Exchange Unit (1. 668) Flats Dimension	"Snap On" WS-1822
Heat Exchange Unit (1. 625) Flats Dimension Open to 1. 668	"Bonney" 1212 (2)
Main Engine Fuel Overboard Drain (1. 480) Flats Dimension	"Bonney" 1248

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CHAPTER 21

AIR CONDITIONING

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CHAPTER 21

AIR CONDITIONING

AIR CONDITIONING AND PRESSURIZATION SYSTEMS — DESCRIPTION / OPERATION

1. General

The Gulfstream I environmental control system provides for pressurization, heating, cooling, ventilation and the means for reduction of humidity in flight or on the ground. True air conditioning is classified as heating or cooling as necessary to maintain a specific level of temperature within the occupied areas of the aircraft, regardless of the ambient temperatures or the operating conditions. Pressurization is the control over the pressure within the occupied areas.

2. Normal Flight Air Conditioning, Operation

(See Figure 1).

During normal in-flight operations, ambient air is scooped in and compressed by a supercharger (hereafter referred to as blower), producing hot compressed air. The blower is mounted on, and is driven by the right hand engine accessory gear box.

Cooling is provided by air cycle cooling equipment, consisting of a primary and secondary heat exchanger, and a cooling turbine (hereafter referred to as bootstrap), which are capable of reducing the temperature of the air from the blower to valves below ambient temperature. This air is distributed to the cabin and cockpit through separate distribution systems to provide air conditioning of the occupied areas. The pressurization system provides control over pressure in the occupied areas utilizing the same airflow.

The aircraft is heated solely by the heat of compression of the blower discharge air. There is no fuel burning heater, exhaust gas heat exchanger, or engine bleed air heat exchanger.

Humidity reduction is provided by a mechanical water separator.

Temperature control of the occupied areas is accomplished by varying the amount of hot blower discharge air which bypasses the cooling equipment. Separate temperature control is provided for the cabin and cockpit. For both compartments, temperature control may be exercised, either automatically or manually from the temperature control section of the upper overhead panel above the copilots station.

3. Ground Air Conditioning and Emergency Flight Operation

A feature of the air conditioning system is that it functions independently when the aircraft is on the ground. Even with the engines shut down, no ground cooling air cart or external electrical supply is required. Complete air conditioning is provided by the auxiliary power unit (hereafter referred to as the APU) supplying hot compressed air to the air conditioning system utilizing the same system as the blower. The necessary flow of coolant air across the heat exchangers of the bootstrap air cycle system is produced during ground operation by the ground fan located in the ram air ducting immediately downstream of the heat exchangers. This fan is driven pneumatically by a portion of the APU output. (In flight, air is forced across the heat exchangers by ram effect and the ground fan windmilling.) The APU is also operable in flight as an alternate pressure source in the event of a blower or right engine malfunction. When so operated, it provides all of the previously described functions of the blower.

4. Distribution System

The cabin distribution system consists of a continuous duct running the full length of the cabin at floor level on both sides. Each duct leads to louvered outlets spaced along its length which direct the air into the compartment. The cabin outlets are not adjustable by passengers. However, maintenance personnel can adjust these louvers individually in order to balance airflow distribution in the compartment. A number of holes are provided in the top surface of the duct at each window. These holes direct a portion of the air up the side walls to protect against "cold wall" effect and they also direct some cool air upward during the cooling operation. The flight station distribution system has two non-controllable foot outlets at the forward pressure bulkhead, one for the pilot and one for the copilot. Each has a side controllable outlet in the cockpit wall just above the side console. Aircraft 1 - 200, 322 and 323 having ASC 127 have valves which provide manual adjustment of the quantity of air applied to the foot outlets.

The cabin and cockpit are provided with "eyeball" supply ducts carrying cool air taken directly from the cold air line downstream of the water separator. "Eyeball" outlets are connected to this line when the aircraft is furnished.

The air conditioning system is a bootstrap, air cycle type, in which control over the cabin and cockpit temperature is accomplished by means of mixing hot compressed air and refrigerated compressed air. The system also employs water separation for humidity reduction. By "air cycle" it is meant that cooling is provided by means of a thermodynamic cycle, using only air as the medium (as opposed to "vapor cycle" systems, which employ Freon or other similar gasses).

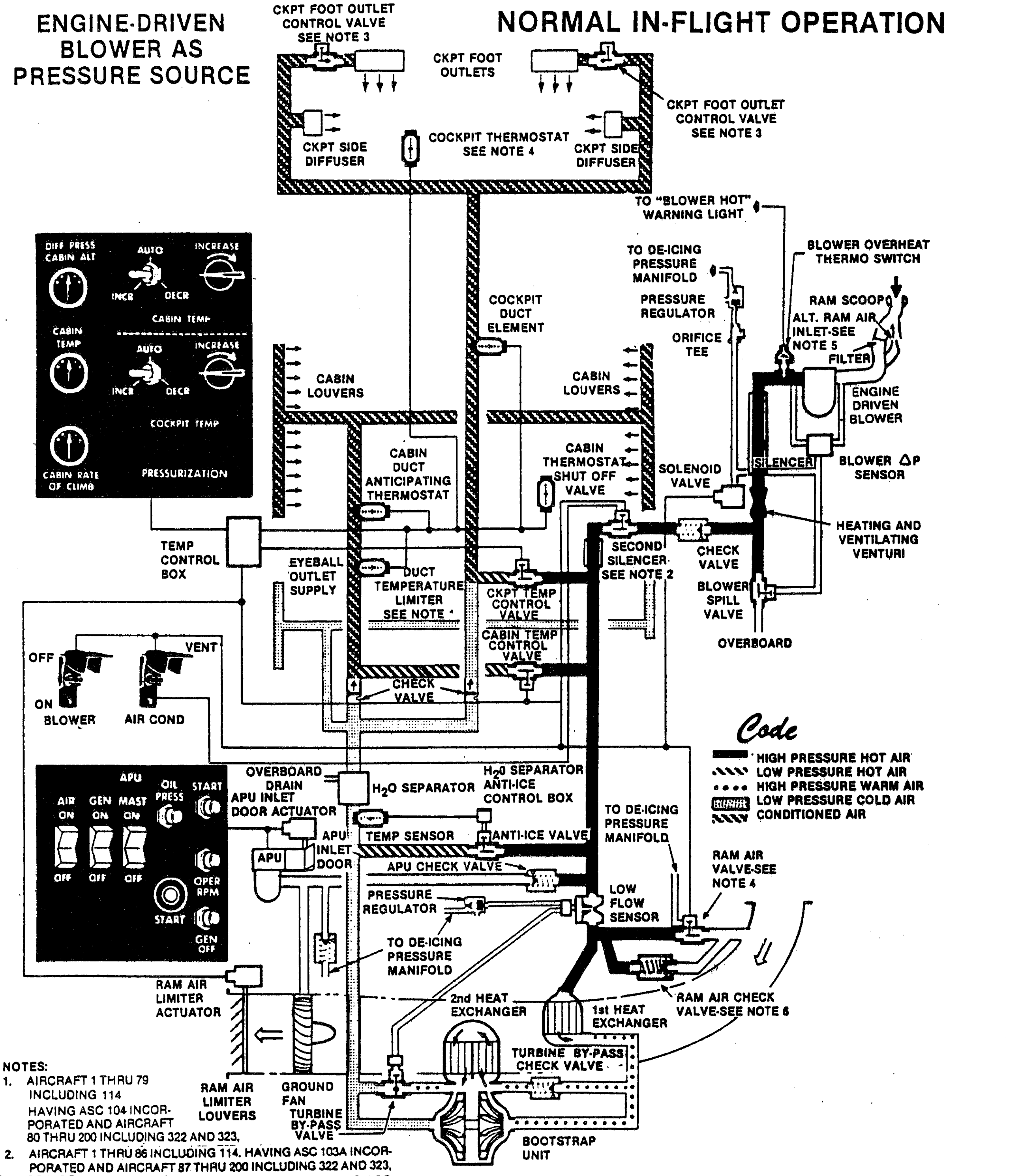
The pressurization system of the Gulfstream I consists of electronic sensing and regulating devices which control the amount of air leaving the aircraft. A pneumatic safety valve is also incorporated, which will automatically regulate the maximum pressure in the aircraft should the automatic, electronic control system fail to function properly. If required, the safety valve will limit the maximum pressure buildup within the pressurized areas (safety pressure relief), and also is the primary control device over the maximum negative pressure (vacuum differential). Suitable controls and indications for the system are located in the cockpit to enable the crew to observe the status of the system with appropriate indicators and be alerted to possible malfunctions by means of appropriate warning devices.

5. **Controls**

For the location of components and controls of the air conditioning and pressurization system See Figure 2.

ENGINE-DRIVEN BLOWER AS PRESSURE SOURCE

NORMAL IN-FLIGHT OPERATION



NOTES:

1. AIRCRAFT 1 THRU 79 INCLUDING 114 HAVING ASC 104 INCORPORATED AND AIRCRAFT 80 THRU 200 INCLUDING 322 AND 323.
2. AIRCRAFT 1 THRU 86 INCLUDING 114, HAVING ASC 103A INCORPORATED AND AIRCRAFT 87 THRU 200 INCLUDING 322 AND 323.
3. AIRCRAFT 1 THRU 200 INCLUDING 322 AND 323 HAVING ASC 127A INCORPORATED.
4. AIRCRAFT 1 THRU 148 INCLUDING 322 AND 323.
5. AIRCRAFT 1 THRU 200 INCLUDING 322 AND 323 HAVING ASC 161A INCORPORATED.
6. AIRCRAFT 149 THRU 200 EXCLUDING 322 AND 323.

Air Conditioning System — Schematic
Figure 1.

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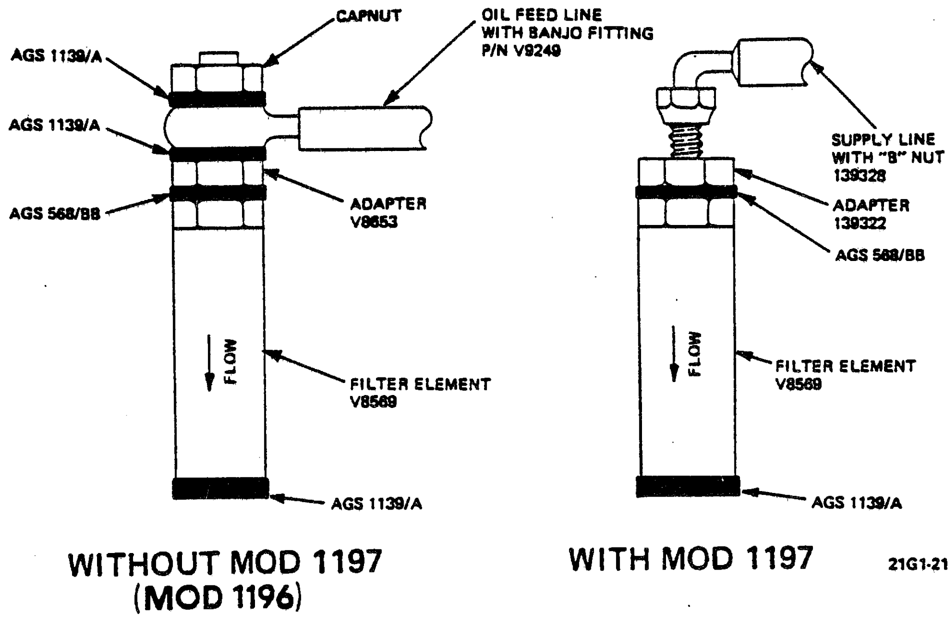
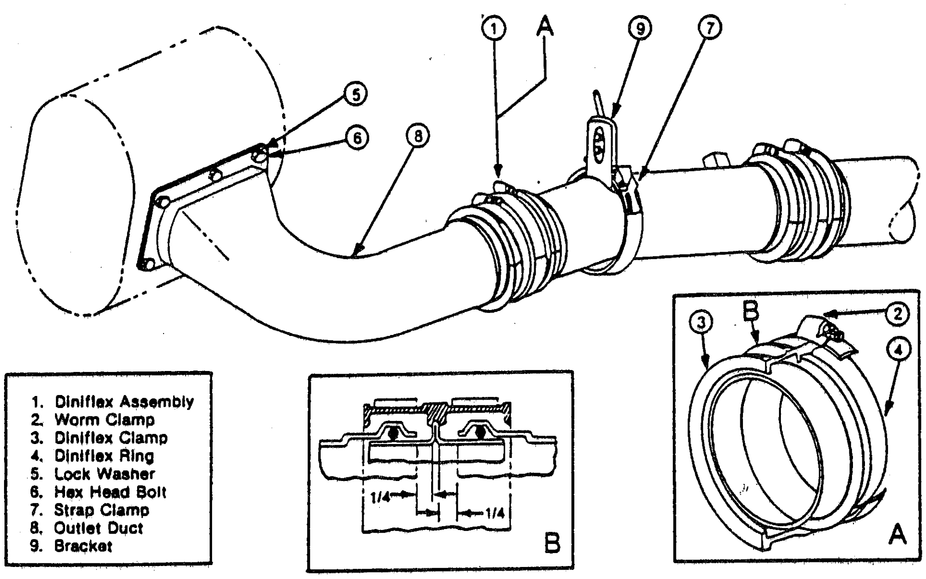


Banjo Fittings
Figure 201.



Blower Outlet Duct
Figure 202.

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(18) Adjust CABIN adjustment screw to face of cabin and cockpit temperature control box (floorboard 15) until a null (no movement) is obtained on cabin bypass valve. Valve must not be against stop.

(19) Replace dome nut when adjustment is complete. Remove resistor. Connect plug.

R. To check the proper resistance of the Temperature Control Resistance Elements, use the following chart:

TEMPERATURE CONTROL RESISTANCE ELEMENTS

TEMPERATURE IN °F VS RESISTANCE IN OHMS AT AMBIENT TEMPERATURES

Temp °F	Cabin Thermo AYLF 4500	Cabin Anticipator AYLF 6916 Unlagged	Cabin Anticipator AYLF 6916 Lagged	Cockpit Therm AYLF3358 OR BYLF—5922	Cockpit Ant. AYLF 6147 OR Cockpit Duct Element AYLF 5623	H_2O Sep Sen AYLF6632
	PINS: A-B OR C-D	PINS: C-D	PINS: A-B	PINS: A-B OR C-D	PINS: A-B OR C-D	PINS: A-B
0		82.5	89.0		82.0	168.0
10		83.5	90.0		83.5	172.0
20		85.0	91.0		85.0	176.0
30		87.0	91.5	770.0	87.0	180.0
40	565.0	89.0	92.5	565.0	89.0	184.0
50	415.0	91.0	93.5	415.0	91.0	188.0
60	320.0	93.0	94.5	320.0	93.0	192.0
70	250.0	95.0	96.5	250.0	95.0	196.0
80	195.0	96.5	97.0	195.0	97.5	200.0
90	150.0	99.5	98.0	150.0	99.5	204.0
100	120.0	102.0	99.0	120.0	102.0	208.0
110	100.0	104.0	100.0		104.5	212.0
120	80.0	107.0	101.5		107.0	216.0

NOTE: Cockpit anticipator AYLF 6147 and cabin anticipator AYLF6916 is used in aircraft 1 through 60 including 114.

Cabin anticipator AYLF6916 is used in the cabin and cockpit in aircraft 61 through 148 including 322 and 323, excluding 114.

Cabin duct element AYLF 5623 is used in aircraft 149 through 200 excluding 322 and 323. There is no cockpit thermostat in these aircraft.

Cockpit thermostat AYLF 3358 is installed in aircraft 1 through 79 including 114 not having ASC 100.

Cockpit blower type thermostat BYLF5922 is used in aircraft 1 through 79 including 114 having ASC 100 in stalled and aircraft 80 through 200 including 322 and 323.

Cabin thermostat AYLF 4500 and BYLF 5922 are identical in resistance vs temperature reaction.

S. The following table gives the approximate resistance of cabin air temperature bulb at surrounding temperature ranges from − 35 to + 120 degrees Fahrenheit and also degrees Centigrade.

OHMS C	TEMP	OHMS F	OHMS C	TEMP	OHMS F
78.97	−35	78.25	106.49	45	92.84
80.57	−30	79.13	108.39	50	93.80
82.17	−25	80.01	110.33	55	94.76
83.77	−20	80.90	112.28	60	95.73
85.39	−15	81.79	114.27	65	96.71
87.04	−10	82.68	116.27	70	97.70
88.69	− 5	83.58	118.31	75	98.70
90.38	0	84.48	119.13	77	99.11
92.08	5	85.39	120.36	80	99.70
93.80	10	86.30	122.45	85	100.71
95.55	15	87.22	124.55	90	101.73
97.31	20	88.14	126.70	95	102.75
99.11	25	89.07	128.85	100	103.78
100.91	30	90.00	131.05	105	104.82
101.65	32	90.38	133.26	110	105.86
102.75	35	90.94	135.51	115	106.91
104.60	40	91.89	137.78	120	107.97

2. **Air Conditioning Components—Servicing**

CAUTION: DO NOT INTERMIX ONE BRAND OF OIL WITH THAT OF ANOTHER BRAND WHEN ADDING OR CHANGING OIL.

A. Lubrication of Bootstrap Unit

(1) Check bootstrap oil level in accordance with times specified in the GI Inspection Schedule. If oil is low replenish with oil approved by Garrett. (See Chapter 12 SERVICING for specific manufacturers of lubricants approved by Garrett.)

(2) Change bootstrap oil in accordance with times specified in the GI Inspection Schedule as described in the following steps:

(a) Remove filler cap on bootstrap unit in order to vent and facilitate draining.

(b) Remove drain fitting at bottom of bootstrap unit and drain oil into suitable container.

(c) Install, and tighten drain fitting and safety wire.

(d) Fill bootstrap unit to full mark on twist-lock type dipstick, with oil approved by Garrett. (Refer Chapter 12 SERVICING for specific manufacturers of lubricants conforming to this specification.)

(e) Install filler cap.

NOTE: If it is evidence, on draining the bootstrap lubricant, that water is present in the drained fluid, it is recommended that the lubricant be drained and replaced more frequently.

B. Cleaning Heat Exchanger Unit

(1) To check heat exchanger unit for excessive accumulation of solid matter, remove forward end cover of primary heat exchanger.

(2) If unit requires cleaning, it can be accomplished in the following manner:

(a) Remove ducting from forward and aft end of heat exchanger unit.

(b) Remove aft end cover of primary heat exchanger and forward and aft end covers of secondary heat exchanger.

(c) Use air hose to blow out accumulated matter from each heat exchanger tube and ducts; at the same time use a vacuum cleaner to take in dirt being blown out by air hose.

(d) When cleaning has been completed, install forward and aft end covers of primary and secondary heat exchangers.

(e) Attach ducting to heat exchanger unit using seal and clamp assemblies.

C. Primary Heat Exchanger Screen — Clean / Inspect / Service

NOTE: If excessive debris is evident on screen, it is suggested that the heat exchangers and bootstrap compressor impeller be inspected and/or cleaned.

(1) Gain access to primary heat exchanger (smaller of two) in tail compartment.

(2) Remove duct clamp from forward primary heat exchanger port. Remove duct, retain clamp and discard O-ring.

(3) Remove bolts securing forward primary heat exchanger pan. Remove pan.

(4) Remove screen with screen gaskets.

(5) Clean screen with vacuum cleaner and inspect for condition, replace as necessary.

(6) Clean pan and visible heat exchanger interior of any debris.

(7) Inspect screen gaskets and replace as necessary.

(8) Install gasket, screen, gasket and pan and secure with bolts. Torque bolts to 20 - 25 inch-pounds.

(9) Inspect duct clamp and replace if necessary.

(10) Install new O-ring and connect duct with clamp. Torque clamp to 25 inch-pounds.

(11) Start APU and apply air load. Check heat exchanger seal joints and clamps for leaks.

D. Water Separator Bag Cleaning

(1) Disassemble water separator to gain access to internal components; remove V-band clamp securing plenum section to condenser. Remove two O-rings and retain if serviceable.

(2) Remove filter element from separator body. If inspection indicates cleaning is not required or if bag is damaged, replace bag and disregard Steps (3) and (4) below.

(3) Brush and vacuum filter.

NOTE: Do not disassemble condenser assembly.

(4) If required, wash filter element in mild detergent. Install bag when wet.

(5) Remove screen covering, drain plenum.

(6) Clean screen, drain plenum, and drain fitting.

(7) Install plenum screen.

(8) Check relief valve in aft end of separator condenser for freedom of movement.

(9) Inspect large O-rings and replace if necessary.

(10) Install O-rings in grooves of separator body.

(11) Place cover on body ensuring O-rings are properly seated.

(12) Install V-band clamp to secure water separator parts.

(13) Install water separator in aircraft as described in removal and installation Section of this Chapter.

E. Boostrap Unit Inboard Impeller — Clean / Inspect

(1) Gain access to bootstrap unit in tail compartment. (See Figure 202)

(2) Remove duct from bootstrap, compressor inlet, retaining clamp and discard O-ring.

(3) Remove duct from bootstrap, compressor outlet, retaining clamp and discard O-ring.

NOTE: If excessive amount of debris is evident at compressor, it is recommended that the heat exchangers be inspected and cleaned.

(4) Apply vacuum cleaner nozzle to compressor outlet to remove any debris. Use stiff nylon bristle brush at compressor inlet to dislodge any debris from impeller.

(5) Apply vacuum to compressor inlet to remove any debris from impeller.

(6) Inspect compressor impeller and surrounding area for damage.

(7) Inspect duct clamps for damage. Replace clamp(s) if any damage is noted.

(8) Install new O-ring at compressor outlet. Position duct and install clamp. Torque clamp to 35 - 40 inch-pounds.

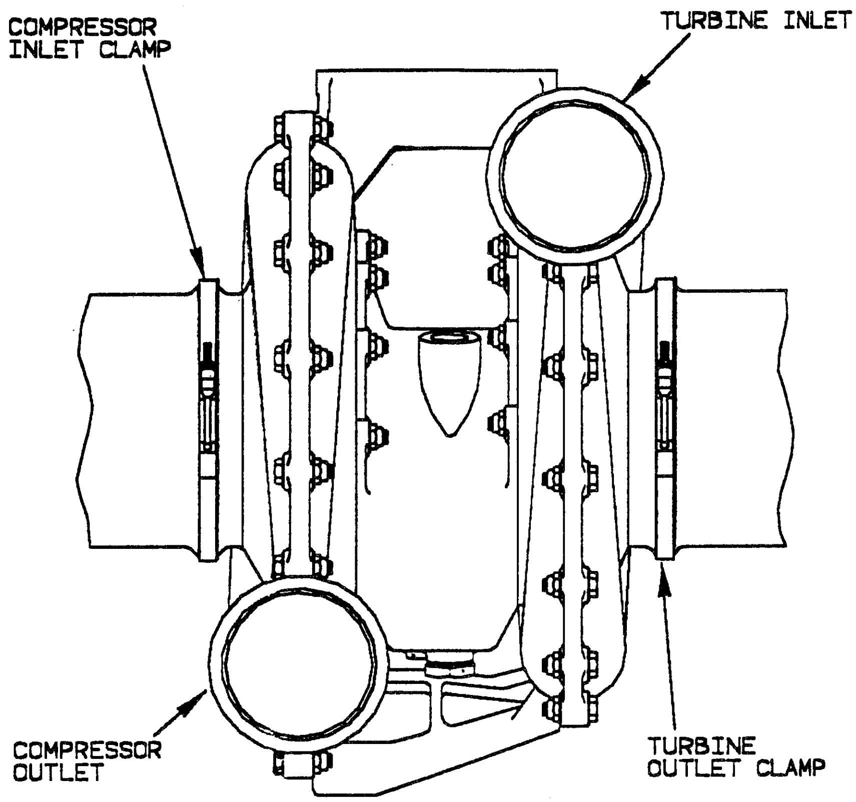
(9) Install new O-ring at compressor inlet. Position duct and install clamp.

(10) Torque compressor inlet clamp to one of the following values. NUCO clamp 40 - 50 inch-pounds, Aeroquip clamp 35 - 40 inch-pounds.

(11) Inspect area for foreign objects, leaks and damage.

(12) Start APU and apply air load. Inspect ducting for leaks.

(13) Shut off air and shut down APU.



TURBINE OUTLET CLAMP

Bootstrap
(View Looking Aft)
Figure 202

AIR CONDITIONING SYSTEM — REMOVAL/INSTALLATION

CAUTION: BEFORE PERFORMING ANY OF THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

1. **Water Separator — Removal / Installation**
   A. Removal
      (1) Gain access to water separator by removing Floorboard 18.
      (2) Remove screws, washer, and nuts securing control cable fairlead to floor support at fuselage station 448, inboard of water separator.
      (3) Disconnect duct coupling at aft end of separator.
      (4) Disconnect three duct couplings at forward end of duct section attached to water separator.

      NOTE: Duct section and water separator are removed as a unit.

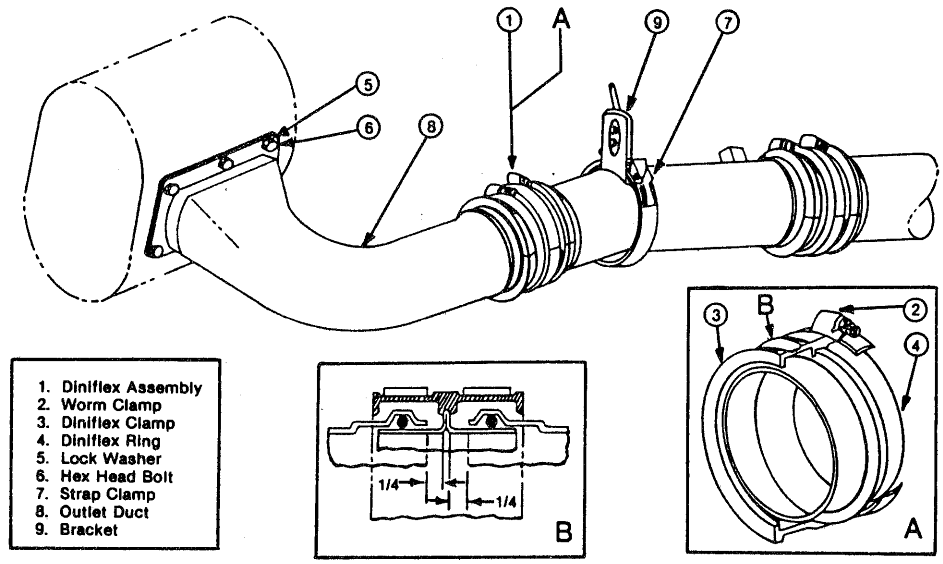
      (5) Disconnect overboard drain flex line from front lower end of separator.
      (6) Disconnect two clamps supporting duct section and two clamps supporting separator to fuselage.
      (7) Pull control cables inboard to provide clearance.
      (8) Remove water separator and duct section.
   B. Installation

      NOTE: If replacement separator is to be installed, transfer the ducts and drain fitting to new unit and orient in same direction.

      (1) Inspect duct sleeves for condition and replace if necessary.
      (2) Pull control cables between Fuselage Stations 448 and 478 to provide clearance for installing water separator and duct section.
      (3) Install water separator and duct section in fuselage support clamps. Position drain fitting at forward bottom most position.
      (4) Secure support clamps around duct section and clamps around separator.
      (5) Connect overboard drain line to front end of separator.
      (6) Connect duct couplings at forward end of duct section.
      (7) Connect duct coupling at aft end of separator.
      (8) Install control cable fairlead to floor support at Fuselage Station 448 and secure with screws, washers and nuts.
      (9) Start APU and airload leak check duct couplings; shut down APU.
      (10) Install Floorboard 18.

2. **Blower Outlet Duct Seals — Removal / Installation** (See Figure 401.)
   A. Removal
      (1) Remove right engine accessory gearbox access panels.
      (2) Remove worm-type clamps from coupling and duct clamp halves.
      (3) Remove bolts attaching 90° elbow duct to blower. Separate duct from blower.
      (4) Inspect gasket between duct and blower. Replace as required.
      (5) Separate ducting by lowering duct so that insert can be removed, remove and discard O-rings seals. Loosen strap clamp if necessary.
      (6) Carefully inspect clamp halves and coupling insert for signs of excessive wear, cracking, etc.; replace as required.
   B. Installation
      (1) Install new O-rings into duct end recesses.
      (2) Install coupling insert and position with insert band centered between duct ends.
      (3) Secure with clamp halves and worm-type clamps.
      (4) Secure ducting strap clamp.
      (5) Replace gasket and attach 90° elbow duct to blower with six bolts. Torque bolts to standard torque value.
      (6) Install gearbox access covers.

3. **Bootstrap Unit — Removal / Installation**
   A. Removal
      (1) Gain access to bootstrap unit in tail compartment.
      (2) Remove duct couplings connecting bootstrap to ducting. Check condition of clamps, replace if damaged.



Blower Outlet Duct
Figure 401

(3) Support unit and remove mounting bolts; remove bootstrap being careful not to tip bootstrap as oil can run out of breather tube.
(4) Remove four O-rings and discard.
(5) Drain oil if unit is being replaced with another unit and remove breather tube and adapter fitting.

B. Installation

NOTE: If new or replacement unit is to be installed, transfer adapter fitting to top port and install as in old unit. Connect breather tube to adapter and, if required, use new O-ring before installing adapter fitting.

(1) Mount unit using hardware previously removed; shim with washers as required to ensure snug fit and tighten bolts.
(2) Connect ducts to unit using new O-rings (2 each).
(3) Install V-band clamps and secure. Torque NUCO clamp 40-50 inch pounds and Aeroquip clamp 35-40 inch pounds.

CAUTION: BOOTSTRAP UNITS ARE SHIPPED DRY AND MUST BE LUBRICATED BEFORE PLACING IN SERVICE. USE ONLY OILS SPECIFICALLY APPROVED BY BRAND NAME AND NUMBER BY GARRETT FOR THIS UNIT. DO NOT INTERMIX DIFFERENT BRANDS OF OIL EVEN IF BOTH ARE ON APPROVED LIST.

(4) Remove twist-lock type dipstick and fill to dipstick FULL mark with oil, install dipstick and ensure that it is locked in position.
~~(5) Start and airload APU; check for leaks at duct couplings.~~
(6) Shut down air and APU if no longer required.

4. **Blower Differential Pressure Sensor — Removal / Installation**
   A. Removal
      (1) Gain access to blower differential sensor by removing right gearbox access cover. Sensor is located on frame directly behind blower.
      (2) Tag and disconnect flexible line and two rigid lines.
      (3) Remove hardware securing sensor to frame and remove sensor.
   B. Installation

   NOTE: If a replacement sensor is to be installed, transfer fittings to new unit and orient in same direction and location.

      (1) Install sensor and secure with hardware.
      (2) Remove tags and connect flexible line and two rigid lines.
      (3) Perform on operational test as outlined in AIR CONDITIONING SYSTEM — MAINTENANCE PRACTICES, paragraph 1, page 201 this Section.

5. **Blower Dump Solenoid Valve — Removal / Installation**
   A. Removal
      (1) Gain access to solenoid by removing right gearbox access cover. Solenoid is located on frame aft of gearbox.
      (2) Disconnect electrical connector.
      (3) Disconnect tubing.
      (4) Remove hardware securing solenoid valve to frame and remove solenoid valve.
   B. Installation

   NOTE: If replacement unit is to be installed, transfer fitting to new unit and orient in same direction and location.

      (1) Install solenoid on frame and secure with hardware.
      (2) Connect tubing and tighten.
      (3) Perform Operational Test of the Air Conditioning Components, this Section.

6. **Spill Valve — Removal / Installation**
   A. Removal
      (1) Gain access to spill valve by removing aft section of nacelle cowling and exposing aft tailpipe section; remove aft tailpipe from aircraft.
      (2) Disconnect ¼-inch line from valve head chamber.
      (3) Disconnect duct couplings from both ends of valve; remove valve, O-ring and sleeve.
   B. Installation

   NOTE: If new or replacement valve is to be installed, transfer fitting from old valve to new unit.

      (1) Inspect O-ring and sleeve and replace if necessary; install valve, O-ring, and sleeve.
      (2) Connect line to fitting.
      (3) Secure duct couplings at each end of valve.
      (4) Perform Operational Test of the Air Conditioning Components, this Section.

7. **Low Flow Bypass Valve — Removal / Installation**
   A. Removal
      (1) Gain access to low flow bypass valve located in duct connecting bootstrap unit to secondary heat exchanger.
      (2) Disconnect line from valve.
      (3) Disconnect duct clamps at both ends of valve.
      (4) Remove valve and two O-rings.
   B. Installation
      (1) Inspect O-rings and replace as required.
      (2) Connect regulated air source to valve fitting. With no air pressure applied to valve, position indicator should indicate closed.
      (3) Slowly apply increasing air pressure to valve, valve should open before 7 psi is applied.
      (4) Remove air pressure from valve, valve should close. Disconnect air supply from valve.
      (5) Install valve with two O-rings in duct and secure with clamps.
      (6) Connect line to valve.
      (7) Perform Operational Test of the Air Conditioning Components, this Section.

8. **Turbine Bypass Check Valve — Removal / Installation**
   A. Removal
      (1) Gain access to turbine bypass check valve, in duct connecting bootstrap unit to primary heat exchanger in tail compartment; this is an in-line type check valve.
      (2) Remove duct couplings at both ends of check valve; remove valve and associated seal and gasket.
   B. Installation
      (1) Inspect gasket and seal and replace if necessary.
      (2) Install valve with associated seal and gasket, and tighten couplings at both ends.
      (3) Start APU and apply air using normal procedure.
      (4) Check valve couplings for leaks.
      (5) Shutdown APU air and APU if no longer required.

9. **Cabin/Cockpit Temperature Control Box — Removal / Installation**
   A. Removal
      (1) Gain access to control box by removing Floorboard 18.
      (2) Disconnect electrical connector from control box.
      (3) Remove and retain hardware attaching control box to mount.
      (4) Remove control box.
   B. Installation
      (1) Install control box on mount with hardware previously removed and secure.
      (2) Connect electrical connector to control box.
      (3) Perform Operational Test of the Air Conditioning Components, this Section.

10. **Engine Driven Blower — Removal / Installation**
   A. Removal
      (1) Gain access to blower, mounted on right engine accessory gearbox, by removing gearbox access covers.
      (2) Remove right alternator.
      (3) Disconnect (at inlet duct end) small flex line connecting blower inlet duct fitting to differential pressure sensor; cap line and fitting.
      (4) Loosen two clamps securing inlet duct assemblies at sleeve and disconnect sleeve from inlet duct.
      (5) Disconnect oil feed line from gearbox; cap port and line.
      (6) Remove bolts and then remove outlet duct assembly from blower, move outlet duct away from blower.
      (7) Disconnect blower drain line from bottom rear of blower.
      (8) Provide clearance to blower removal by disconnecting rigid tubing at differential pressure sensor HI PORT and fire detector element connector adjustment to differential pressure sensor; cap all lines, ports, and fire detector fittings.

      CAUTION: SUPPORT BLOWER TO PREVENT WEIGHT FROM DAMAGING QUILL SHAFT.

      (9) Supporting blower, remove and retain nuts securing blower to gearbox studs.
      (10) Carefully move blower aft and away from aircraft.
   B. Installation

   CAUTION: IF BLOWER IS BEING REMOVED DUE TO INTERNAL FAILURE AND DISINTEGRATION OF THE ROTORS, THE DUCTS SHOULD BE INSPECTED FOR FOREIGN OBJECTS FROM BLOWER TO HEAT EXCHANGER AND BLOWER INLET AREA PRIOR TO OPERATION OF NEW BLOWER.

      (1) Install blower drain line fitting at bottom rear of blower, using two new crush washers.
      (2) Install inlet duct with removed hardware and new gasket; safety wire bolts.

      NOTE: If blower has not been used for over a month, either in storage or on an aircraft, then it must be primed with oil as follows:

      (3) Remove priming plugs from banjo fittings of two oil transfer lines at rear of blower. Force 60 cc of gearbox oil into fittings using pressure oil can. After priming each fitting, reinstall priming plugs and lockwire.

      CAUTION: SUPPORT BLOWER WEIGHT, ENSURE QUILL SHAFT DOES NOT BEAR BLOWER WEIGHT.

      (4) Carefully position blower to mate quill shaft and gearbox splines.

(5) Secure blower to gearbox with washers and nuts previously removed. Torque nuts to 160 inch pounds.
(6) On those blowers without MOD #1197 incorporated, connect oil feed line loosely to oil metering unit at rear, using new crush washers. On those blowers incorporating MOD #1197, no washers are required, install line loosely on fitting.
(7) Using pressure oil can force gearbox oil into gearbox and end of oil feed line until oil seeps out of blower end fitting; tighten fitting at blower end to a torque of 90 to 100 inch-pounds and safety wire.
(8) Connect other end of oil feed line to gearbox and tighten.
(9) Connect blower drain line at aft bottom of blower.
(10) Using new gasket, connect outlet duct to blower and tighten bolts.
(11) Connect inlet duct sleeve and clamp, check alignment of inlet and outlet ducts and adjust if required.
(12) Connect small flex line from inlet duct elbow fitting to LO side of differential pressure sensor.
(13) Connect rigid tubing to HI PORT of differential pressure sensor (remove to step 20.H).
(14) Discard copper sealing washer from fire detector element disconnected in step 20.H; install new washer and connect element to bulkhead fitting; tighten nut to a torque of 50 to 70 inch-pounds and safety wire.
(15) Install alternator.
(16) Perform engine run with access covers off and check for leaks.

CAUTION: DO NOT EXCEED 11,000 RPM WITH ACCESS COVERS OFF.

(17) If no oil leaks after engine run, secure gearbox access covers.

11. **Cabin and Cockpit Temperature Control Valves — Removal / Installation**

A. Removal
(1) Gain access to temperature control valves, under Floorboard 15 (cockpit valve) or Floorboard 18 (cabin valve).
(2) Disconnect electrical plug.
(3) Disconnect two duct couplings from valve.
(4) Remove and retain hardware holding valve to mount.
(5) Remove valve and O-rings.

B. Installation
(1) Install valve on mount with hardware previously removed and secure.
(2) Inspect two O-rings and replace if required; install in couplings and connect ducts.
(3) Connect electrical plug.
(4) Perform Operational Test of the Air Conditioning Components, this Section.
(5) Install floorboard previously removed.

12. **Blower Isolation Valve — Removal / Installation**

A. Removal
(1) Gain access to blower isolation valve by removing foorboard 14 in cabin.
(2) Disconnect electrical connector.
(3) Remove duct clamps on both sides of valve.
(4) Remove hardware holding valve to structure.
(5) Remove valve and O-rings.

B. Installation
(1) Inspect O-rings and replace as required.
(2) Install valve with O-rings; flow arrow must face aft.
(3) Mount valve.
(4) Connect forward and aft couplings, ensuring tight couplings.
(5) Connect electrical connector.
(6) Perform Operational Test of the Air Conditioning Components, this Section.

13. **Cabin and Cockpit Anticipators and Cockpit Duct Element — Removal / Installation**

NOTE: Aircraft 1 through 148, including 322 and 323 have a cabin and cockpit anticipator. Aircraft 149 through 200 have a cabin anticipator and a cockpit duct element in place of a cockpit anticipator; removal and installation procedures are the same.

(1) Remove Floorboard 15 to gain access to anticipators and/or duct element.

NOTE: Cabin anticipator is just forward of refrigerated air line junction and duct downstream of cabin temperature control bypass valve; cockpit anticipator (or duct element) is just to the left of flap central gearbox in cockpit distribution ducting.

(2) Disconnect electrical plug; remove mounting hardware and remove unit.

B. Installation
(1) Install unit into duct using hardware previously removed.
(2) Connect electrical plug.
(3) Install Floorboard 15.
(4) Start and air load APU and after compartment temperature stabilizes in AUTO, check that temperature control rheostat operation reacts normally for that compartment involved.
(5) Shut down air and APU if no longer required.

14. **Cabin and Cockpit Thermostat — Removal / Installation**

A. Removal
(1) Gain access to thermostat as follows:
(a) Cabin thermostat, all aircraft. Location is determined by outfitting agency.
(b) Cockpit thermostat for Aircraft 1 through 79 including 114 not having ASC 100 incorporated is located on the copilot's relay box (Fuselage Station 133).
(c) Cockpit thermostat for Aircraft 80 through 148, including 322 and 323, excluding 114 having ASC 100 incorporated is located on right side of pedestal kick plate.
(d) Aircraft 149 through 200, excluding 322 and 323 have no cockpit thermostat.
(2) Disconnect electrical connector.
(3) Remove hardware securing thermostat to structure and remove thermostat.

B. Installation

NOTE: If cabin thermostat is to be replaced, transfer cabin temperature bulb to new unit.

(1) Install thermostat on structure and secure with hardware.
(2) Connect electrical connector to thermostat.
(3) Energize main and essential dc buses.
(4) Check that blower in blower type thermostat operates when associated TEMP CONTROL SELECTOR is placed in AUTO and does not operate in OFF.
(5) If cabin thermostat assembly is replaced, ensure cabin air temperature gage is indicating compartment temperature.
(6) Start and operate APU and air load using normal procedure
(7) With associated system in AUTO operation, and after temperature in associated compartment stabilizes, check that temperature control rheostat reacts normally for compartment involved.
(8) Shut down air and APU.
(9) Deenergize main and essential dc buses.
(10) Close access previously opened.

15. **Cabin and Cockpit Temperature Control Rheostats — Removal / Installation**

A. Removal

NOTE: The cabin and cockpit rheostats are identical units.

(1) Remove knobs from cabin and cockpit temperature control rheostats.

CAUTION: DO NOT USE FORCE TO REMOVE LIGHTING PANEL AND AVOID TWISTING OR BENDING PANEL.

(2) Remove screws that secure edge lighting panel to right side overhead panel and remove panel.
(3) Remove screws securing overhead panel to structure and move panel away from structure. Support panel to prevent strain on wiring and lines.
(4) Tag and disconnect electrical leads from rheostat.
(5) Remove locknut from front of shaft and remove rheostat.

B. Installation
(1) Install rheostat in panel and secure with locknut.
(2) Connect electrical leads to rheostat and remove tags.
(3) Install overhead panel on structure and secure with screws.

(4) Install edge lighting panel on overhead panel and secure with screws.
(5) Install knobs on cabin and cockpit temperature control rheostats.
(6) Energize main and essential dc buses.
(7) Start APU and air load using normal procedures.
(8) With TEMP CONTROL SELECTOR for compartment which has rheostat replaced, place system in the AUTO operation. Set rheostat at midpoint (12 o'clock position) and allow temperature to stabilize.
(9) When temperature is stabilized turn rheostat to full decrease (counterclockwise). Check that applicable compartment gradually receives cooler air.
(10) Turn rheostat to full increase (clockwise). Check that applicable compartment gradually receives warmer air.
(11) Return rheostat to midpoint. System should return to original temperature.
(12) Shut down air and APU.
(13) Check overhead edge lighting panel for proper operation, use lighting control on overhead panel.
(14) De-energize main and essential dc buses.

16. **Water Separator Anti-Ice Valve — Removal / Installation**

A. Removal

(1) Gain access to water separator anti-ice valve as follows :

(a) Aircraft 1 through 80 including 114 not having ASC 128A incorporated, valve is in tail compartment between APU closure and pressure dome, lower section.

(b) Aircraft 1 through 80 including 114 having ASC 128A incorporated and Aircraft 81 through 200 including 322 and 323, valve is in the baggage compartment (remove either floorboard 24 or 25).

(2) Disconnect electrical plug.
(3) Disconnect two duct couplings at each end of valve.
(4) Remove hardware securing valve to structure.
(5) Remove valve and two O-rings.

B. Installation

(1) Inspect O-rings and replace as necessary.
(2) Install valve to structure with O-rings and secure with hardware previously removed.
(3) Connect duct couplings and tighten securely.
(4) Connect electrical plug.
(5) Gain access to water separator control box under Floorboard 21.
(6) Remove control box plug.
(7) Jumper control box plug pins C to A.
(8) Energize main and essential dc buses.
(9) Valve should move toward close stop.

NOTE: Valve will not pulse during this check.

(10) Pull H_2O SEP circuit breaker (pilot's circuit breaker panel).
(11) Move jumper at control box plug to pins B and C.
(12) Push in H_2O SEP circuit breaker; valve should move to open.

NOTE: Valve will not pulse.

(13) Pull H_2O SEP circuit breaker.
(14) Remove jumper and reconnect plug to water separator anti-ice control box.
(15) Push in H_2O SEP circuit breaker.
(16) Start and air load APU; check valve for leaks at couplings.
(17) Shut down APU air and APU if no longer required.
(18) De-energize main and essential dc buses.

17. **Water Separator Anti-Ice Control Box — Removal / Installation**

A. Removal

(1) Remove Floorboard 21 to gain access to control box.
(2) Disconnect electrical plug.
(3) Remove screws securing box to structure; remove box.

B. Installation

(1) Install control box using hardware previously removed.
(2) Install electrical plug.

(3) Disconnect electrical plug from water separator anti-ice sensor (under Floorboard 22).

NOTE: Remove Floorboard 22 only if required, as it may be possible to reach sensor through Floorboard 21 (sensor is in main duct just aft of water separator unit).

(4) Place a jumper from pin A to B of sensor plug.
(5) Energize main and essential dc buses.
(6) Water separator anti-ice valve should go to full open (valve may pulse).
(7) Pull H_2O SEP circuit breaker (pilot's circuit breaker panel—cockpit).
(8) Remove jumper from sensor plug; do not connect plug to sensor.
(9) Push in H_2O SEP circuit breaker; water separator anti-ice valve should move to full closed (valve may pulse).
(10) Deenergize main and essential dc buses.
(11) Connect sensor plug to sensor.
(12) Replace Floorboard 21 and 22 (if removed) and/or other access opened.

18. **Water Separator Anti-Ice Sensor — Removal / Installation**

A. Removal
(1) Gain access to sensor by removing Floorboard 22.
(2) Disconnect electrical connector from sensor.
(3) Remove screws securing sensor to duct boss and remove sensor.

B. Installation
(1) Connect electrical connector to sensor.
(2) Energize main and essential dc buses.
(3) Gain access to water separator anti-ice valve.
(a) Aircraft 1 through 80 including 114 not having ASC 128A incorporated the anti-ice valve is located in tail compartment between APU enclosure and pressure dome lower station.
(b) Aircraft 1 through 80 including 114 having ASC 128A incorporated and Aircraft 81 through 200 including 322 and 323, anti-ice valve is in the baggage compartment (remove either Floorboards 24 or 25).
(4) With sensor in air temperature above 40°F, anti-ice valve should be closed (valve cam against closed stop).
(5) Immerse sensor probe in ice water. Anti-ice valve should open (valve cam against open stop).

NOTE: A pulsating operation of valve is normal.

(6) Remove sensor from ice water.
(7) Deenergize main and essential buses.
(8) Install sensor in duct boss and secure with screws.
(9) Connect electrical connector.
(10) Install floorboards previously removed.

19. **Ram Air Limiter Actuator — Removal / Installation**

A. Removal
(1) Close CABIN/COCKPIT TEMP CONT circuit breaker on pilot's circuit breaker panel.
(2) Energize main and essential dc buses.
(3) Place and hold CABIN TEMP SELECTOR switch on right side overhead panel to INCREASE; hold for approximately 50 seconds. No air supply on aircraft.

CAUTION: DO NOT POSITION CABIN TEMP SELECTOR SWITCH TO AUTO.

(4) Position CABIN TEMP SELECTOR switch on right side overhead panel to OFF.
(5) Deenergize main and essential dc buses.
(6) Gain access to ram air limiter actuator in tail compartment at base of ram air exhaust duct.
(7) Disconnect electrical connector from actuator.
(8) Remove screws securing actuator to mounting base; pull actuator spline from louvered spline recess and remove actuator.

B. Installation

NOTE: If replacement actuator is to be installed, remove the four bolts securing actuator mounting brackets to actuator. Transfer the brackets in same location to new actuator. Leave bolts slightly loose, actuator must be aligned with louver on installation. The elongated holes are for spline alignment purposes.

(1) Connect electrical connector to actuator. Do not install actuator into louvered box at this time.

(2) Energize main and essential dc buses.
(3) Actuator will run to full closed louvered position. When actuator stops mate spline with louver box and align spline by adjusting brackets.

CAUTION: LOUVERS MUST BE IN FULL CLOSED POSITION BEFORE MATING ACTUATOR TO LOUVER BOX.

(4) Tighten through bolt when spline is aligned. Secure actuator and brackets to mounting base with screws.
(5) Operate CABIN TEMP SELECTOR switch to DECREASE (manual) and hold for 50 seconds. Have assistant observe actuator; actuator will pulsate to open louvered position and stop with louvers approximately full open.
(6) Operate CABIN TEMP SELECTOR switch to INCREASE and hold for 50 seconds. Have assistant observe actuator; actuator will pulsate to closed louvered position and stop with louvers fully closed. Ensure actuator cycles without binding or abnormal operation.
(7) Deenergize main and essential dc buses.

20. **Ram Air Valve — Removal / Installation** (Aircraft 1 thru 148 including 322 and 323, Aircraft 149 and Subsequent, see Procedure below)

A. Removal

NOTE: This valve is electro-pneumatic.

(1) Gain access to ram air valve, on right side of ram air duct in tail compartment.
(2) Remove electrical plug.
(3) Remove duct coupling halves at both valve ports.
(4) Remove rigid tubing to valve control chamber.
(5) Remove hardware securing valve to bracket.
(6) Remove valve and two H-ring seals.
(7) If new or replacement valve is to be installed, remove fitting at valve control chamber, inspect O-ring, and replace as necessary; install fitting in new valve and orient in same direction as olf valve.

B. Installation

(1) Place H-ring on ram air duct fitting, fit valve to bracket, and secure.
(2) Move H-ring into proper position and install coupling halves, ensuring that H-ring is properly seated in clamp recesses; tighten clamp, do not install other coupling at this time.
(3) Connect electrical plug.
(4) Apply pressure to valve using either of the following methods:
  (a) If external air source is readily available at deicing system pressure manifold in nacelle, connect small line to ram air valve in tail (disconnected in Step 20.A.(4)), and pressurize deicing system pressure manifold until cockpit gage reads 15 to 18 psi; valve remains closed.
  (b) If nacelle external supply is not readily available, connect regulated source of clean air directly to valve control chamber port, leaving rigid tubing line disconnected and apply 15 to 18 psi to valve; valve remains closed.
(5) Energize essential dc bus.
(6) With AIR COND-VENT switch (copilot's console—cockpit) in AIR COND (guard down) position, valve should remain closed.
(7) Place AIR COND-VENT switch to VENT position; valve should move to open.
(8) Place AIR COND-VENT switch to AIR COND position; valve should return to closed.
(9) Deenergize essential dc bus and remove pneumatic power.
(10) If source 20.B.(4)(b) was used, remove test line from valve fitting and connect rigid tubing aircraft line to valve fitting.
(11) Install H-ring and other end of duct to valve; secure with clamp halves, ensuring seal is properly placed in clamp recesses before closing clamp halves.
(12) Start and air load APU using normal procedure.
(13) Check valve couplings for leaks.

NOTE: If couplings leak, seal is cut due to improper installation. Replace with new seal and reconnect properly.

(14) Shut down APU air and APU if no longer required.

21. **Ram Air Check Valve — Removal / Installation** (Aircraft 149 and Subsequent)
    A. Removal
        (1) Gain access to ram air check valve (ram air duct just above heat exchanger) in tail compartment.
        (2) Remove duct clamp halves and coupling H-ring.
        (3) Remove other duct coupling and remove valve.
    B. Installation
        (1) Inspect flappers for cleanliness and freedom of movement; clean any debris from flappers and valves. Inspect H-ring seal and replace as necessary.
        (2) Install valve.
        (3) Install duct coupling, ensure H-ring is properly seated in clamp half recesses, secure clamp halves.
        (4) Start and air load APU; check couplings for leaks.
        (5) Shut down air and APU.

22. **Ground Cooling Fan — Removal / Installation**
    A. Removal
        (1) Gain access to ground cooling fan, below heat exchangers in ram air duct in tail compartment.
        (2) Disconnect inlet duct coupling from APU on side of ground cooling fan.
        (3) Disconnect band clamps and remove coupling sleeves.
        (4) Remove ground cooling fan.
    B. Installation
        (1) Check condition of duct coupling seal and sleeves, replace as necessary.
        (2) Position fan in place with sleeves and clamps and tighten into place.
        (3) Install APU inlet duct on side of fan.
        (4) Start APU and air load using normal procedures; check duct couplings for leaks, ensuring that fan operates and draws sufficient air across heat exchangers for ground cooling operation.
        (5) Shut down APU air and APU if no longer required.

23. **Nacelle Silencer — Removal / Installation**
    A. Removal

        NOTE: The nacelle silencer is installed only in right nacelle. It is necessary to remove right engine tailpipe in order to remove or install silencer.

        (1) On right nacelle, remove aft top access cover and inboard side access cover.
        (2) Remove aft tailpipe.
        (3) Remove forward and aft duct clamps and coupling halves from silencer couplings.
        (4) Support silencer and remove bolts securing silencer to structure.
        (5) Remove silencer and discard O-rings from each end.

        NOTE: It may be necessary to loosen duct clamps from aft duct line, to free silencer for removal.

    B. Installation
        (1) Install new O-rings and position silencer to pick up ducts at both ends.
        (2) Orient and bolt silencer to structure.
        (3) Install duct coupling halves and secure with worm type clamps.
        (4) Install aft tailpipe.
        (5) Install access covers removed previously.

24. **Primary and Secondary Heat Exchanger — Removal / Installation**
    A. Removal

        NOTE: Both heat exchangers are removed as a unit from aircraft, then further separation can take place.

        (1) Remove APU from aircraft.
        (2) Disconnect three duct couplings from rear end of heat exchanger package and one duct coupling from forward end.
        ~~(3) Loosen band clamp securing large sleeve to overboard duct assembly below the package.~~
        (4) At top of package, remove bolts securing ram air inlet duct assembly.
        (5) Support heat exchanger package and remove 22 bolts (11 top and 11 bottom), which secure package to support brackets.
        (6) Remove bolts from forward and aft support brackets.
        (7) Remove package from aircraft through opening used for APU removal.
        (8) Separate primary from secondary heat exchanger as required by removing associated hardware.

B. Installation

(1) Assemble primary and secondary heat exchangers and associated assemblies to form package (if assembly was broken down).

NOTE: If primary heat exchanger is to be replaced, remove screen and gasket from inlet panel. Clean screen and install in new unit before installing in aircraft. Screen and extra gasket on inlet side are installed in following sequence; gasket, screen, gasket, pan.

(2) Install heat exchanger package through APU opening and position to pick up forward and aft support brackets.
(3) Secure forward and aft supports.
(4) Install 11 top and 11 bottom bolts which secure package to upper and lower support brackets.
(5) Install bolts at top of package, securing ram air inlet to package.
(6) At bottom, secure large sleeve to bottom of package with band clamp.
(7) Inspect four seals which were removed from couplings and replace as necessary.
(8) Install seals and connect three rear couplings and one forward coupling.
(9) Install APU.
(10) Start APU and air load using normal procedures.
(11) Check all duct couplings and heat exchanger package joints for leaks.
(12) Shut down APU air and APU if no longer required.

D. With sensor in air temperature above 40°F, anti-ice valve should be closed (valve cam against closed stop).

E. Immerse sensor probe in ice water. Anti-ice valve should open (valve cam against open stop).

NOTE: A pulsating operation of valve is normal.

F. Remove sensor from ice water.

G. Deenergize main and essential buses.

38. **Installation of Water Separator Anti-Ice Sensor**

A. Install sensor in duct boss and secure with screws.

B. Connect electrical connector.

C. Install floorboards previously removed.

39. **Removal of Ram Air Limiter Actuator**

A. Close CABIN/COCKPIT TEMP CONT circuit breaker on pilot's circuit breaker panel.

B. Energize main and essential dc buses.

C. Place and hold CABIN TEMP SELECTOR switch on right side overhead panel to INCREASE; hold for approximately 50 seconds. No air supply on aircraft.

CAUTION: DO NOT POSITION CABIN TEMP SELECTOR SWITCH TO AUTO.

D. Position CABIN TEMP SELECTOR switch on right side overhead panel to OFF.

E. Deenergize main and essential dc buses.

F. Gain access to ram air limiter actuator in tail compartment at base of ram air exhaust duct.

G. Disconnect electrical connector from actuator.

H. Remove screws securing actuator to mounting base; pull actuator spline from louvered spline recess and remove actuator.

40. **Installation of Ram Air Limiter Actuator**

NOTE: If replacement actuator is to be installed, remove the four bolts securing actuator mounting brackets to actuator. Transfer the brackets in same location to new actuator. Leave bolts slightly loose, actuator must be aligned with louver on installation. The elongated holes are for spline alignment purposes.

A. Connect electrical connector to actuator. Do not install actuator into louvered box at this time.

B. Energize main and essential dc buses.

C. Actuator will run to full closed louvered position. When actuator stops mate spline with louver box and align spline by adjusting brackets.

CAUTION: LOUVERS MUST BE IN FULL CLOSED POSITION BEFORE MATING ACTUATOR TO LOUVER BOX.

D. Tighten through bolt when spline is aligned. Secure actuator and brackets to mounting base with screws.

E. Operate CABIN TEMP SELECTOR switch to DECREASE (manual) and hold for 50 seconds. Have assistant observe actuator; actuator will pulsate to open louvered position and stop with louvers approximately full open.

F. Operate CABIN TEMP SELECTOR switch to INCREASE and hold for 50 seconds. Have assistant observe actuator; actuator will pulsate to closed louvered position and stop with louvers fully closed. Ensure actuator cycles without binding or abnormal operation.

G. Deenergize main and essential dc buses.

41. **Removal of Ram Air Valve** (Aircraft 1 through 148 including 322 and 323)

NOTE: This valve is electro-pneumatic.

A. Gain access to ram air valve, on right side of ram air duct in tail compartment.

B. Remove electrical plug.

C. Remove duct coupling halves at both valve ports.

D. Remove rigid tubing to valve control chamber.

E. Remove hardware securing valve to bracket.

F. Remove valve and two H-ring seals.

G. If new or replacement valve is to be installed, remove fitting at valve control chamber, inspect O-ring, and replace as necessary; install fitting in new valve and orient in same direction as olf valve.

42. **Installation of Ram Air Valve** (Aircraft 1 through 148 including 322 and 323)

A. Place H-ring on ram air duct fitting, fit valve to bracket, and secure.

B. Move H-ring into proper position and install coupling halves, ensuring that H-ring is properly seated in clamp recesses; tighten clamp, do not install other coupling at this time.

C. Connect electrical plug.

D. Apply pressure to valve using either of the following methods:

(1) If external air source is readily available at deicing system pressure manifold in nacelle, connect small line to ram air valve in tail (disconnected in step 41.D.), and pressurize deicing system pressure manifold until cockpit gage reads 15 to 18 psi; valve remains closed.

(2) If nacelle external supply is not readily available, connect regulated source of clean air directly to valve control chamber port, leaving rigid tubing line disconnected, and apply 15 to 18 psi to valve; valve remains closed.

E. Energize essential dc bus.

F. With AIR COND-VENT switch (copilot's console—cockpit) in AIR COND (guard down) position, valve should remain closed.

G. Place AIR COND-VENT switch to VENT position; valve should move to open.

H. Place AIR COND-VENT switch to AIR COND position; valve should return to closed.

I. Deenergize essential dc bus and remove pneumatic power.

J. If source 42.D.(2) was used, remove test line from valve fitting and connect rigid tubing aircraft line to valve fitting.

K. Install H-ring and other end of duct to valve; secure with clamp halves, ensuring seal is properly placed in clamp recesses before closing clamp halves.

L. Start and air load APU using normal procedure.

M. Check valve couplings for leaks.

NOTE: If couplings leak, seal is cut due to improper installation. Replace with new seal and reconnect properly.

N. Shut down APU air and APU if no longer required.

43. **Removal of Ground Cooling Fan**

A. Gain access to ground cooling fan, below heat exchangers in ram air duct in tail compartment.

B. Disconnect inlet duct coupling from APU on side of ground cooling fan.

C. Disconnect band clamps and remove coupling sleeves.

D. Remove ground cooling fan.

44. **Installation of Ground Cooling Fan**

A. Check condition of duct coupling seal and sleeves, replace as necessary.

B. Position fan in place with sleeves and clamps and tighten into place.

C. Install APU inlet duct on side of fan.

D Start APU and air load using normal procedures; check duct couplings for leaks, ensuring that fan operates and draws sufficient air across heat exchangers for ground cooling operation.

E. Shut down APU air and APU if no longer required.

45. **Removal of Nacelle Silencer**

NOTE: The nacelle silencer is installed only in right nacelle. It is necessary to remove right engine tailpipe in order to remove or install silencer.

A. On right nacelle, remove aft top access cover and inboard side access cover.

B. Remove aft tailpipe.

C. Remove forward and aft duct clamps and coupling halves from silencer couplings.

D. Support silencer and remove bolts securing silencer to structure.

E. Remove silencer and discard O-rings from each end.

NOTE: It may be necessary to loosen duct clamps from aft duct line, to free silencer for removal.

46. **Installation of Nacelle Silencer**

A. Install new O-rings and position silencer to pick up ducts at both ends.

B. Orient and bolt silencer to structure.

C. Install duct coupling halves and secure with worm type clamps.

D. Install aft tailpipe.

E. Install access covers removed previously.

47. **Removal of Primary and Secondary Heat Exchanger**

NOTE: Both heat exchangers are removed as a unit from aircraft, then further separation can take place.

A. Remove APU from aircraft.

B. Disconnect three duct couplings from rear end of heat exchanger package and one duct coupling from forward end.

C. Loosen band clamp securing large sleeve to overboard duct assembly below the package.

D. At top of package, remove bolts securing ram air inlet duct assembly.

E. Support heat exchanger package and remove 22 bolts (11 top and 11 bottom), which secure package to support brackets.

F. Remove bolts from forward and aft support brackets.

G. Remove package from aircraft through opening used for APU removal.

H. Separate primary from secondary heat exchanger as required by removing associated hardware.

48. **Installation of Primary and Secondary Heat Exchanger**

A. Assemble primary and secondary heat exchangers and associated assemblies to form package (if assembly was broken down).

NOTE: If primary heat exchanger is to be replaced, remove screen and gasket from inlet panel. Clean screen and install in new unit before installing in aircraft. Screen and extra gasket on inlet side are installed in following sequence; gasket, screen, gasket, pan.

B. Install heat exchanger package through APU opening and position to pick up forward and aft support brackets.

C. Secure forward and aft supports.

D. Install 11 top and 11 bottom bolts which secure package to upper and lower support brackets.

E. Install bolts at top of package, securing ram air inlet to package.

F. At bottom, secure large sleeve to bottom of package with band clamp.

G. Inspect four seals which were removed from couplings and replace as necessary.

H. Install seals and connect three rear couplings and one forward coupling.

I. Install APU.

J. Start APU and air load using normal procedures.

K. Check all duct couplings and heat exchanger package joints for leaks.

L. Shut down APU air and APU if no longer required.

RAM AIR SYSTEM — DESCRIPTION / OPERATION

1. Ram Air Ventilation

On Aircraft 1 - 148, 322 and 323 (See Figure 1) emergency ventilation air is supplied by the ram air valve. It is an electrically controlled spring-loaded closed, pneumatic shutoff valve. It is opened against the spring by admitting air from the pneumatic boot deicing system pressure manifold to the control head of the valve by means of normally closed, energize-to-open solenoid integral within the valve. This solenoid is energized by lifting the guard on the AIR-COND-VENT switch on the right console and moving the switch to the VENT position. (See Figure 2). With the switch in the VENT position; the blower spill valve opens fully to spill the blower air overboard; the cabin outflow valves of the pressurization control system open fully, dumping the cabin pressure; the ram air valve opens fully, permitting ram air to enter the aircraft distribution system from the tap off in the coolant (ram air) duct, and the blower shutoff (isolation) valve closes preventing blower air from entering the system.

Once this has been accomplished, control over the quantity of ram air fed to the cabin and cockpit is provided by the manual switch positions of the cabin and cockpit temperature control system. In this case the switch labeled INCREASE and DECREASE apply to quantity of flow rather than temperature. When INCREASE is selected for the cabin, it has double effect, not only of opening the cabin temperature control bypass valve, but also closing the ram air limiter louvers which provides more ram pressure recovery in the coolant (ram air) duct, thus forcing more ram air into the distribution system.

With the AIR COND-VENT switch in VENT, pressurization or air conditioning of the aircraft by the blower or APU is impossible. When the AIR COND-VENT switch is placed in the AIR COND position, this closes the ram air valve's solenoid. With the pressure supply cut off, the air inside the ram air valve's control head bleeds out through an orifice permitting the spring to close the valve. This switch position also opens the blower shutoff (isolation) valve and allows the blower spill valve and the cabin outflow valves to return to normal operation. Both pressurization and air conditioning are now available from either the blower or the APU as applicable.

Ram air ventilation is available in emergency dc operation since the AIR COND-VENT switch is fully operative

On Aircraft 149 - 200 differs from earlier aircraft in ram air ventilation mode of operation in that the electrically controlled, pneumatically operated ram air valve is replaced by the ram air check valve. The operation of the system is essentially the same except that no pneumatic pressure or electrical power is necessary to operate the ram air check valve. When the aircraft is pressurized, the ram air check valve prevents loss of pressurization into the ram air duct, thus ensuring the pressure integrity of the system. If the aircraft is depressurized and the sources of pressure are cut off, such as placing the AIR COND-VENT switch in the VENT position, the blower dumps, isolation valve closes, and the pressurization outflow valves open as in earlier aircraft, relieving all pressure from the aircraft. As the pressure drops, the air flowing through the dorsal fin ram air duct will force the ram air valve open and supply air for ventilation purposes to the system. Control over the amount of ram air is the same as in earlier aircraft, that is, operation of the cabin and cockpit temperature control switches to INCREASE or DECREASE in the manual mode.

2. Ram Air Limiting (Coolant Air)

Ram air scooped into an opening at the base of the fin is used as a coolant for the heat exchangers. It passed down through the primary and secondary heat exchangers which, in the aircraft, are mounted in parallel, and exits overboard through a set of fixed louvers in the bottom skin. Prior to reaching these fixed louvers, the coolant air passes through a set of movable louvers whose function is to limit the amount of ram airflow to that required for cooling.

This limiting is necessitated by the fact that the physical size of the ram air path is determined by a low altitude, low speed, hot day condition. This means that during normal operation at cruise altitudes and speeds on an average day, an unnecessarily large amount of air would flow through the coolant (ram air) path and would create unnecessary drag and reduce the aircraft performance. It is, therefore, desirable to reduce this coolant (ram air) flow and hence, the drag it produces, to a minimum.

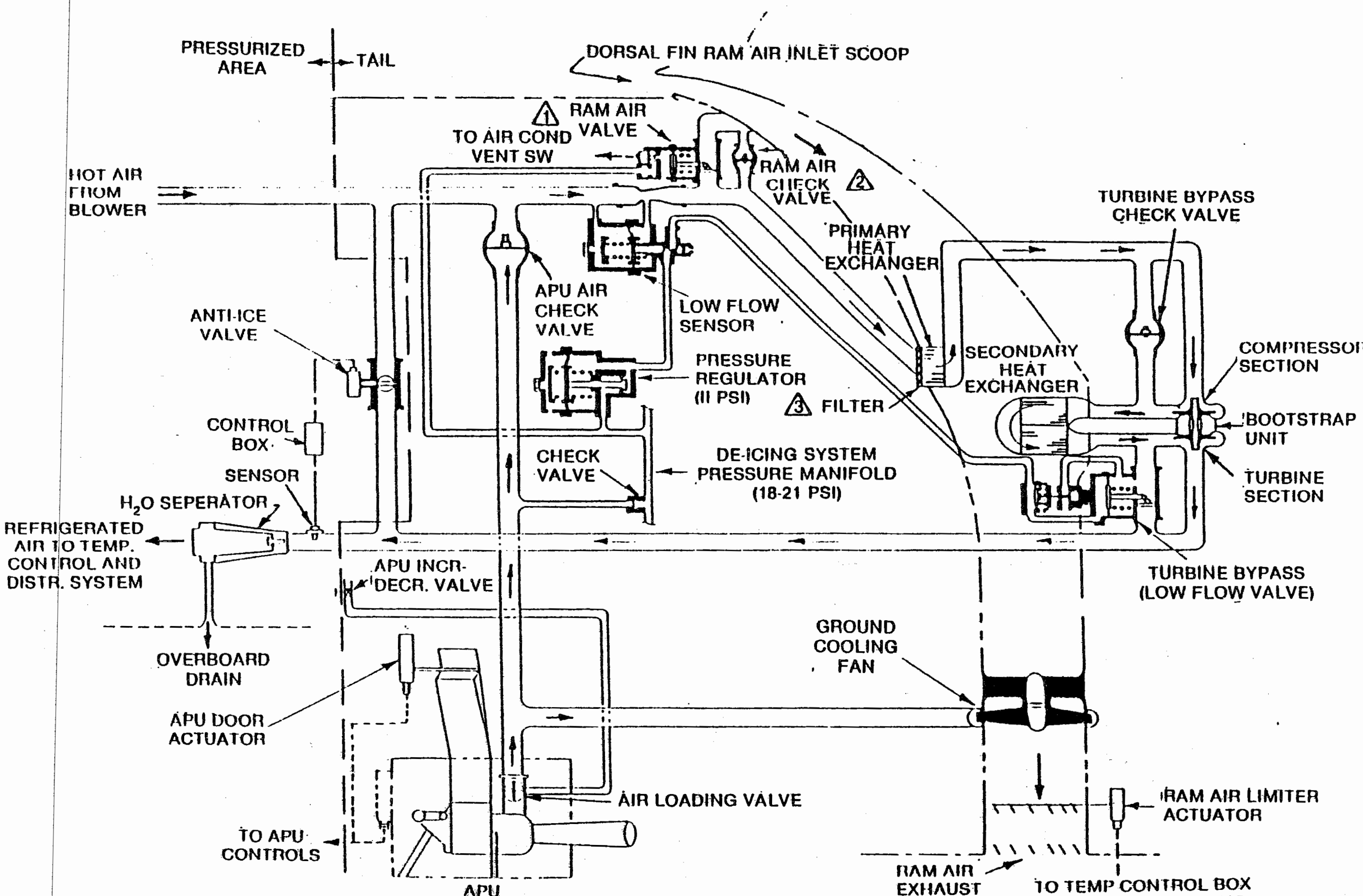
This reduction is accomplished by a set of movable louvers in the coolant air path downstream of the heat exchangers. These louvers are positioned by a linkage system operated by an electrical actuator. The actuator

is connected to the cabin temperature control valve by means of a synchronizing circuit so that when cabin cooling is required the louvers run open, admitting more coolant air at the same time the cabin temperature control valve runs toward closed. The reverse situation applies when cabin heating is required. The position of the cabin temperature control bypass valve and the louvers are always inversely proportional to each other.

3. **Ram Air Limiter Synchronizing System**

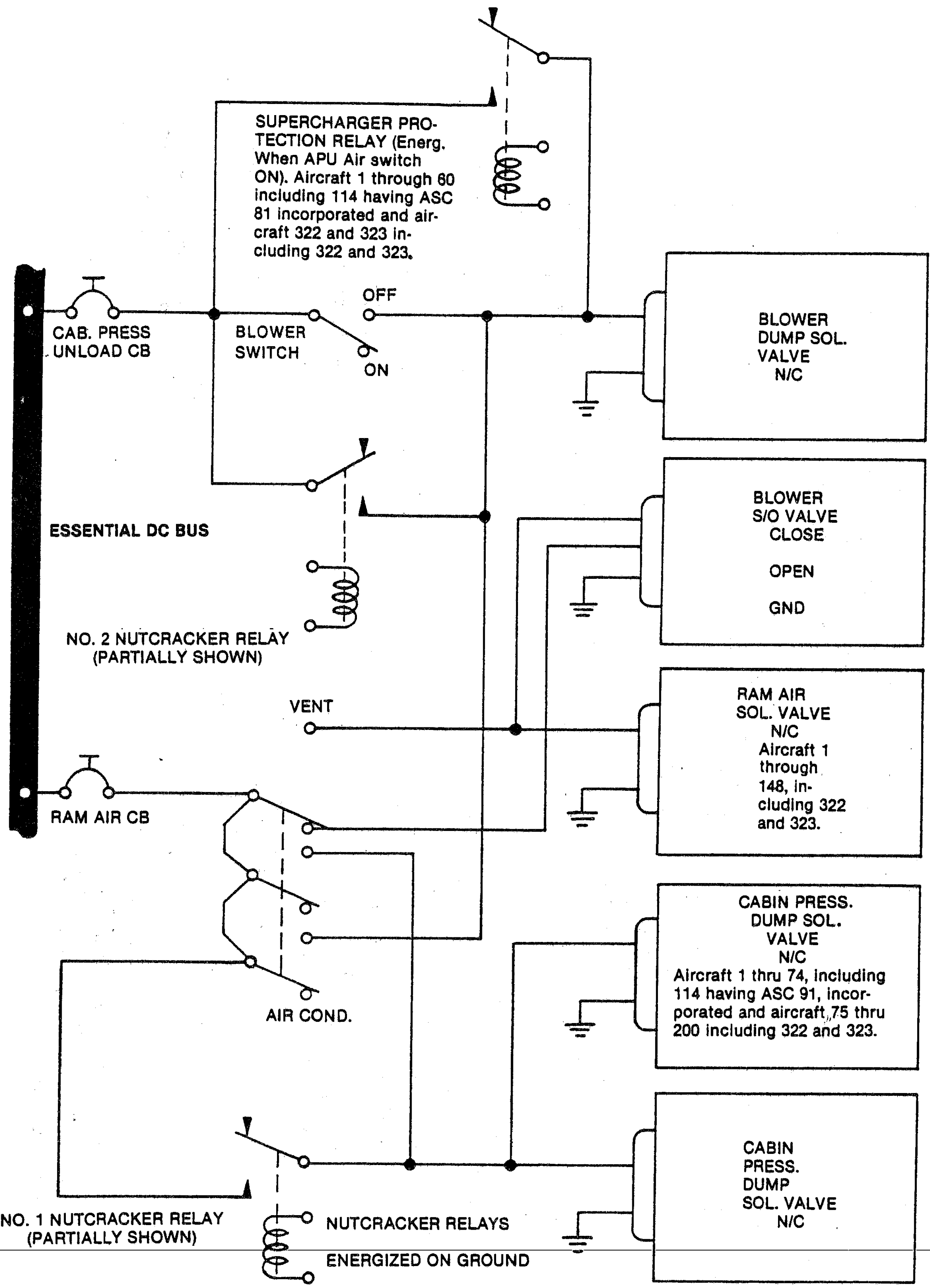
(See Figure 3)

Movable ram air limiter louvers controlling coolant airflow across the heat exchangers in the refrigeration equipment are operated by the cabin temperature control system. This is accomplished by the ram air limiter synchronizing circuit in the automatic temperature control box acting through potentiometers which are part of the cabin temperature control valve and ram air limiter louver actuator. When the cabin temperature control valves move toward open, the actuator closes the louvers proportionally and vice versa.

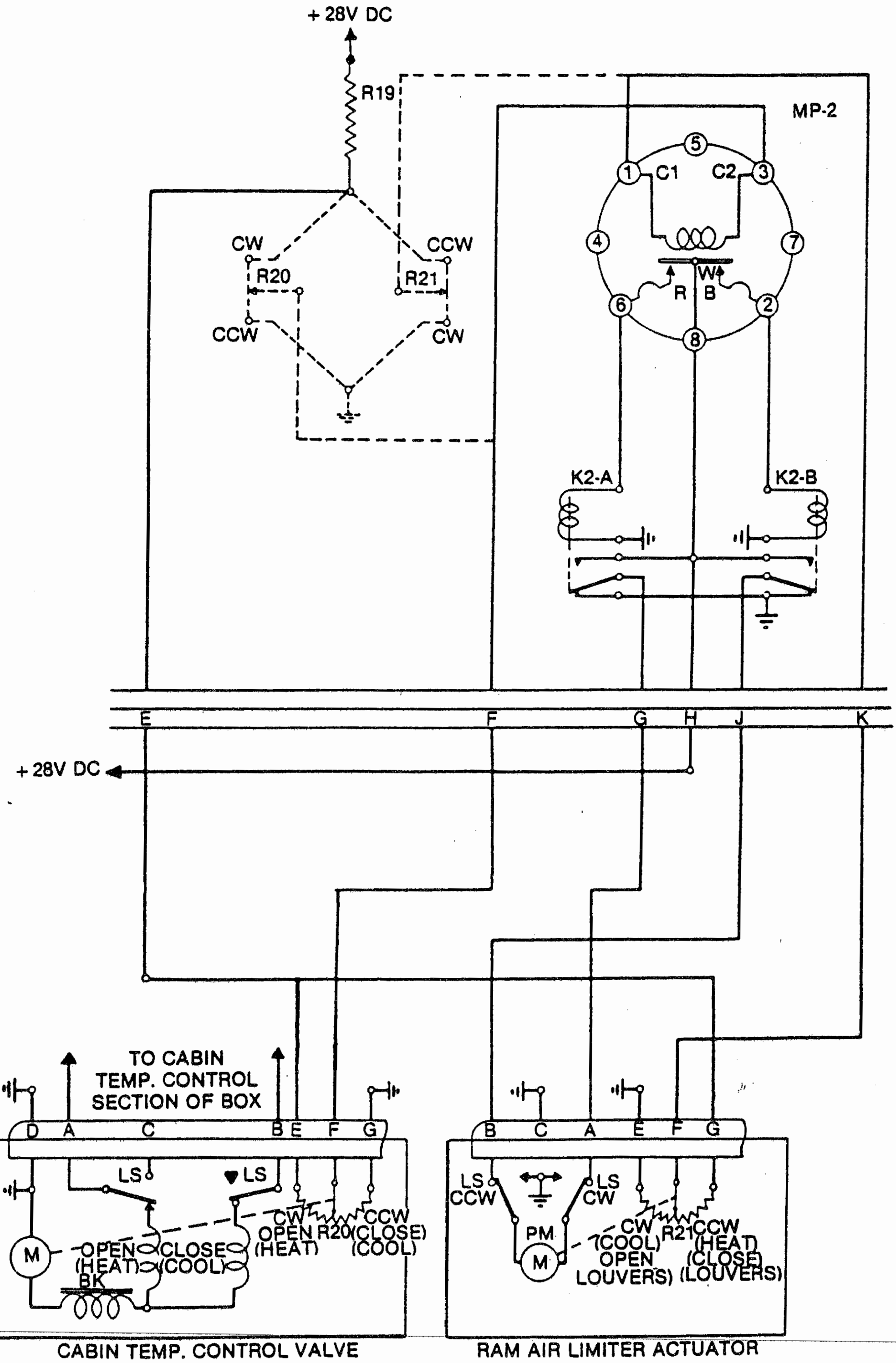


1. Aircraft 1 through 148, including 322 and 323
2. Aircraft 149 through 200, excluding 322 and 323
3. Aircraft 1 through 84, including 114 having ASC 132 incorporated and aircraft 85 through 200, including 322 and 323.

Refrigeration, Dehumidification and APU Supply Systems — Schematic
Figure 1.



Air Conditioning System Blower and Vent System — Schematic
Figure 2.



Ram Air Limiter Synchronizing circuit
Figure 3.

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RAM AIR SYSTEM — MAINTENANCE PRACTICES

1. Ram Air Limiter Actuator — Removal / Installation

A. Removal

(1) Depress CABIN/COCKPIT TEMP CONT circuit breaker on pilots circuit breaker panel.

(2) Energize main and essential dc buses.

(3) Place and hold CABIN TEMP SELECTOR switch on right side overhead panel to INCREASE; hold for approximately 50 seconds. No air supply on aircraft.

CAUTION: DO NOT POSITION CABIN TEMP SELECTOR SWITCH TO AUTO.

(4) Position CABIN TEMP SELECTOR switch on right side overhead panel to OFF.

(5) De-energize main and essential dc buses.

(6) Gain access to ram air limiter actuator in tail compartment at base of ram air exhaust duct.

(7) Disconnect electrical connector from actuator.

(8) Remove screws securing actuator to mounting base; pull actuator spline from louvered spline recess and remove actuator.

B. Installation

NOTE: If replacement actuator is to be installed, remove the four bolts securing actuator mounting brackets to actuator. Transfer the brackets in same location to new actuator. Leave bolts slightly loose, actuator must be aligned with louver on installation. The elongated holes are for spline alignment purposes.

(1) Connect electrical connector to actuator. Do not install actuator into louvered box at this time.

(2) Energize main and essential dc buses.

(3) Actuator will run to full closed louvered position. When actuator stops mate spine with louver box and align spline by adjusting brackets.

CAUTION: LOUVERS MUST BE IN FULL CLOSED POSITION BEFORE MATING ACTUATOR TO LOUVER BOX.

(4) Tighten through bolt when spline is aligned. Secure actuator and brackets to mounting base with screws.

(5) Operate CABIN TEMP SELECTOR switch to DECREASE (manual) and hold for 50 seconds. Have assistant observe actuator; actuator will pulsate to open louvered position and stop with louvers approximately full open.

(6) Operate CABIN TEMP SELECTOR switch to INCREASE and hold for 50 seconds. Have assistant observe actuator; actuator will pulsate to closed louvered position and stop with louvers fully closed. Ensure actuator cycles without binding or abnormal operation.

(7) De-energize main and essential dc buses.

2. Ram Air Valve — Removal / Installation

CAUTION: BEFORE PERFORMING THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

(Aircraft 1 - 148, 322 and 323, Aircraft 149 - 200)

A. Removal

NOTE: This valve is electro-pneumatic.

(1) Gain access to ram air valve, on right side of ram air duct in tail compartment.

(2) Remove electrical plug.

(3) Remove duct coupling halves at both valve ports.

(4) Remove rigid tubing to valve control chamber.

(5) Remove hardware securing valve to bracket.

(6) Remove valve and two H-ring seals.

(7) If new or replacement valve is to be installed, remove fitting at valve control chamber, inspect O-ring, and replace as necessary; install fitting in new valve and orient in same direction as old valve.

B. Installation

(1) Place H-ring on ram air duct fitting, fit valve to bracket, and secure.

(2) Move H-ring into proper position and install coupling halves, ensuring that H-ring is properly seated in clamp recesses; tighten clamp, do not install other coupling at this time.

(3) Connect electrical plug.

(4) Apply pressure to valve using either of the following methods:

(a) If external air source is readily available at deicing system pressure manifold in nacelle, connect small line to ram air valve in tail (disconnected in Step A.(4) above), and pressurize deicing system pressure manifold until cockpit gage reads 15 - 18 psi, valve remains closed.

(b) If nacelle external supply is not readily available, connect regulated source of clean air directly to valve control chamber port, leaving rigid tubing line disconnected and apply 15 - 18 psi to valve; valve remains closed.

(5) Energize essential dc bus.

(6) With AIR COND-VENT switch (copilots console-cockpit) in AIR COND (guard down) position, valve should remain closed.

(7) Place AIR COND-VENT switch to VENT position; valve should move to open.

(8) Place AIR COND-VENT switch to AIR COND position; valve should return to closed.

(9) De-energize essential dc bus and remove pneumatic power.

(10) If source (4)(b) was used, remove test line from valve fitting and connect rigid tubing aircraft line to valve fitting.

(11) Install H-ring and other end of duct to valve; secure with clamp halves, ensuring seal is properly placed in clamp recesses before closing clamp halves.

(12) Start and air load APU using normal procedure.

(13) Check valve couplings for leaks.

NOTE: If couplings leak, seal is cut due to improper installation. Replace with new seal and reconnect properly.

(14) Shut down APU air and APU if no longer required.

3. Ram Air Valve — Functional Test (Aircraft 1 - 148, 322 and 323)

(See Figure 201).

A. Gain access to ram air valve in tail compartment.

B. Remove safety wire and one screw of bottom bearing cover; loosen other screw and swing cover aside to expose bearing and shaft.

C. With AIR COND-VENT switch in AIR COND position, and no deicing system manifold pressure air available, mark the bottom of butterfly valve shaft with a light pencil mark. This will indicate closed position of the valve. (See Figure 202)

D. Apply air pressure to valve using either of the following methods.

(1) If external air source is available at deicing system pressure manifold in nacelle, pressurize manifold in increments until cockpit gage reads 15 - 18 psi.

(2) If external air source is unavailable,connect regulated source of clean air directly to valve control chamber port by disconnecting rigid tubing line to electrical plug on valve; apply 15 to 18 psi directly to valve.

E. Connect external dc power unit to aircraft and energize main and essential dc buses (BATT SW-NORMAL, EXT PWR SW-ON)

F. Ensure RAM AIR circuit breaker is depressed.

G. Place AIR COND-VENT switch to VENT; mark on shaft (Step (3) above) should move approximately 90° (valve open).

H. Place AIR COND-VENT switch to AIR COND. Mark should rotate back to original closed position.

I. Repeat Steps F. and G. above several times; place switch to AIR COND (valve closed).

J. De-energize main and essential dc buses and pneumatic source. If air was directly applied to valve by method in Step 4.(2) above, remove external source of air and connect rigid tubing line to valve; ensure connection is tight.

K. Swing valve bearing cover back and install screw; tighten both screws and safety wire.

NOTE: On some aircraft the ram air coolant duct is fitted with a removable access plate on the aft end of the upper portion of the duct, approximately six inches above the top of the heat exchangers. The valve butterfly may be observed directly by removing the eight bolts securing the access plate to the duct, and observing the action of the butterfly utilizing an inspection mirror and flashlight. Follow Steps C., thru E, of above procedure. Operate AIR COND-VENT switch several times. In AIR COND the butterfly should be CLOSED. In VENT the butterfly should be OPEN. Replace access plate after check is completed.

4. Ram Air Check Valve — Functional Test (Aircraft 149 - 200 (only)

A. Gain access to ram air check (ram air duct just above heat exchanger) in tail compartment.

B. Remove inboard coupling halves, roll H-ring into duct and move duct so that valve flappers can be seen.

C. Inspect flappers for cleanliness and freedom of movement; clean any debris from flappers and valve, inspect H-ring seal and replace as necessary.

D. Install duct coupling, ensure H-ring is properly seated in clamp half recesses, secure clamp halves.

E. Start and air load APU; check valve couplings for leaks.

F. Shut down air and APU.

5. Ram Air Check Valve — Removal / Installation

(Aircraft 149 - 200)

CAUTION: BEFORE PERFORMING THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to ram air check valve (ram air duct just above heat exchanger) in tail compartment.

(2) Remove duct clamp halves and coupling H-ring.

(3) Remove other duct coupling and remove valve.

B. Installation

(1) Inspect flappers for cleanliness and freedom of movement; clean any debris from flappers and valves. Inspect H-ring seal and replace as necessary.

(2) Install valve.

(3) Install duct coupling, ensure H-ring is properly seated in clamp half recesses, secure clamp halves.

(4) Start and air load APU; check couplings for leaks.

(5) Shut down air and APU.

6. Ground Cooling Fan — Removal / Installation

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to ground cooling fan, below heat exchangers in ram air duct in tail compartment.

(2) Disconnect inlet duct coupling from APU on side of ground cooling fan.

(3) Disconnect band clamps and remove coupling sleeves.

(4) Remove ground cooling fan.

B. Installation

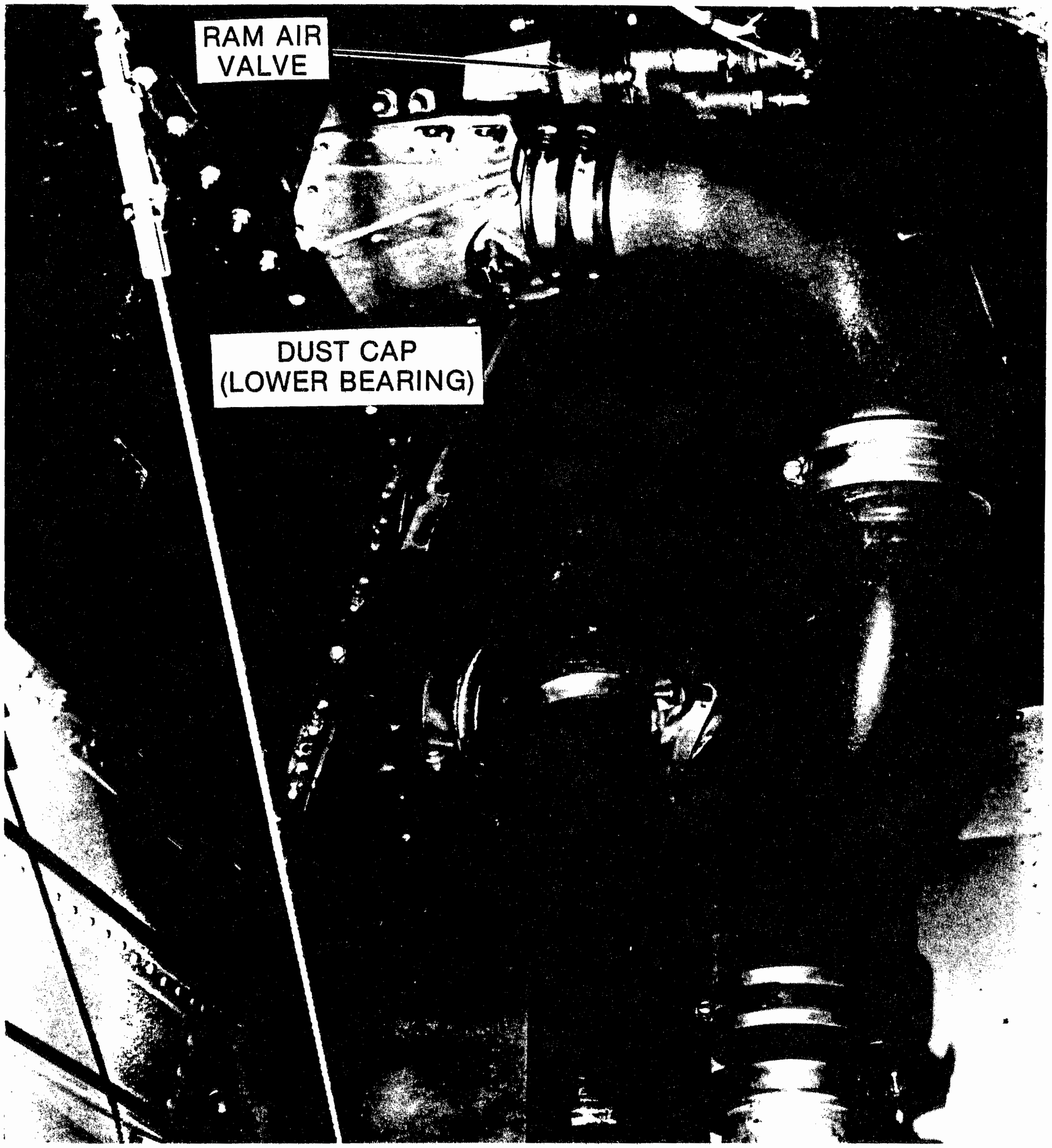
(1) Check condition of duct coupling seal and sleeves, replace as necessary.

(2) Position fan in place with sleeves and clamps and tighten into place.

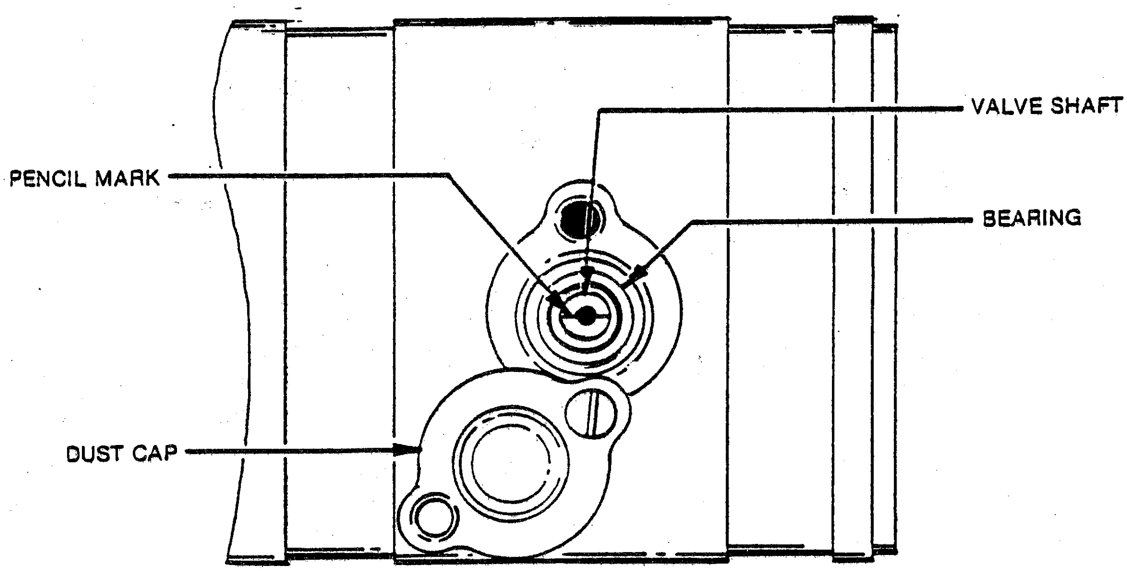
(3) Install APU inlet duct on side of fan.

(4) Start APU and air load using normal procedures. Check duct couplings for leaks, ensuring that fan operates and draws sufficient air across heat exchangers for ground cooling operation.

(5) Shut down APU air and APU if no longer required.



Ram Air Valve Installation
(Aircraft 1 - 148, 322 and 323)
Figure 201.



Valve Shaft
Figure 202.

TEMPERATURE CONTROL SYSTEM — DESCRIPTION / OPERATION

1. Temperature Control (Bypass) System

Once past the check valve (and the second silencer, aircraft having ASC 103), the direction of the air is controlled by the bypass valves on the cabin and cockpit temperature control system. (See Figure 1) The closing of the cabin and cockpit temperature control bypass valves force the air aft through the ducting below the floor on the right side to the air cycle cooling equipment located in the tail behind the pressure bulkhead. The air is then cooled and brought forward beneath the cabin floor on the left side and distributed to the cabin and cockpit outlets after mixing with the portion of the hot air which is bypassed directly to the outlet ducting. The hot compressed (bypassed) air, plus refrigerated compressed air, constitutes temperature controlled air, which in turn emits from the cabin and cockpit outlets.

Opening the cabin and cockpit temperature control (bypass) valves permit the air from the blower (hot), to bypass the cooling equipment and enter the cabin cockpit outlet ducting directly. If the cabin is under pressure, the blower sees a high resistance flow path, and develops discharge pressure, thus the air delivered is hot. If the cabin is unpressurized, thus offering low resistance to blower flow, the resistance of the ducting between the blower and the cabin is increased by the action of the flow limiter and heating venturi, thus assuring sufficient back pressure to produce heat. When this is done, the blower again sees a high resistance path and delivers hot air. All heating of this aircraft is solely by heat of compression.

Separate temperature control is provided for both the cabin and the cockpit. (See Figure 2, Figure 3 and Figure 4) Control can be exercised for both compartments either automatically or manually from the temperature control section of the upper overhead panel above the copilots station. Manual operation is selected by moving the toggle switch involved out of the AUTO position. Momentarily holding the toggle switch in the INCREASE position sends a signal to the particular temperature control valve involved. This signal runs the valve toward open, thereby allowing more hot air to bypass, thus increasing compartment temperature. Conversely, momentarily holding the toggle switch in the DECREASE position closes the valve which forces the air to first pass through the cooling equipment. These manual signals go directly to the valves and do not pass through the control box of the automatic system. Full travel of the temperature control valves which are motor-driven in both directions requires approximately 40 seconds when manually controlled.

The automatic temperature control system for the cabin is identical in all aircraft, consisting of a thermostat in the compartment, a duct element (anticipator) in the supply duct, a temperature selector rheostat in the CABIN section of the upper overhead panel, a selector switch adjacent to the rheostat, and a control box under the cabin floor.

When in AUTO mode of operation, the anticipator duct element, thermostat, and the selector rheostat form the legs of a Wheatstone bridge circuit and feed information into the CABIN section of the temperature control box, located under floorboards 15, to control the cabin temperature control bypass valve. (See Figure 5 and Figure 6)

Two different types of cockpit systems are installed as follows:

A. Aircraft 1 - 148, 322 and 323 a system similar to the cabin in all respects.

B. Aircraft 149 - 200 only have a cockpit system which consists of a duct element, selector rheostat, control switch and a transistorized amplification unit in the COCKPIT section of the temperature control box (See Figure 7). THERE IS NO THERMOSTAT IN THE COCKPIT SYSTEM. The manual mode of operation is the same as earlier aircraft.

CAUTION: THE CABIN AND COCKPIT TEMPERATURE CONTROL BOX FOR AIRCRAFT 1 - 148, 322 AND 323 IS DIFFERENT FROM THE BOX INSTALLED IN AIRCRAFT 149 - 200, AND ARE NOT INTERCHANGEABLE. THE EARLIER CONTROL BOXES DO NOT CONTAIN THE TRANSISTORIZED COCKPIT TEMPERATURE CONTROL AMPLIFIER.

In AUTO mode of operation, the bridge circuits compare the temperature (as sensed by the various elements) to the selected temperature (as determined by selector rheostat position). If an error exists, the control box section involved will then power the appropriate bypass valve in the correct direction.

Suitable time delay factors are incorporated into each system to prevent overshoot of the valve and to avoid hunting. These functions allow the air to stabilize without excessive temperature changes in the compartment. The cabin and cockpit temperature control system receives power in AUTO and MANUAL modes of operation from the main dc bus through the CABIN/COCKPIT TEMP circuit breaker. Therefore, all temperature control is inoperative in EMERGENCY DC OPERATION, the valves remaining in their previously selected position before going to emergency dc. Emergency ram air ventilation (AIR COND-VENT SWITCH) and blower unloading (BLOWER ON-OFF SWITCH) however, are fed from the ESSENTIAL DC BUS and are operative in emergency dc.

NOTE: (Reference Customer Bulletin 138) This temperature control box is shock mounted and allowed to move on the isolation tray on which it is mounted. This Customer Bulletin gives information to reduce failure of the temperature control box and connecting wires caused by improper cable routing if box, mounting, or wiring is relocated. Proper cable routing to the shock mounted box allows a minimum of nine inches of free cable with a three inch minimum radius of curvature.

2. **Cabin / Cockpit Thermostat Modifications**

The cabin compartment thermostat contains a small fan to ensure an adequate flow of cabin air over the thermostat. In some instances the cabin temperature sensing element fan produces radio noise. This noise is minimized by the inclusion of a radio noise filter in the unit in which cabin temperature sensor part number AYLF4500 has been replaced by, or modified to become part number BYLF5922.

NOTE: Cabin temperature sensor part number BYLF5922 is the same as the cockpit temperature sensor element installed in aircraft 1-79, 114 having ASC 100 and Aircraft 80 - 200, 322 and 323.

Aircraft 1 - 79, 114 having ASC 100 and Aircraft 80 - 200, 322 and 323 puts an internal blower in the cockpit thermostat similar to the type utilized in the cabin system. The unit was previously located behind the copilots seat, mounted on the cover of the right station 133 relay panel. The blower type unit is relocated into the right side of the pedestal, just inboard of the copilots seat. Due to the fact that the blower-type thermostat requires dc power to energize the blower motor, this change not only relocates the existing thermostat leads, but adds the necessary wiring to power the blower motor. It is wired in the same way as the cabin thermostat, and operates in the same manner. The element which is installed is identified as BYLF5922, which is the manufacturer's part number. As discussed above, this unit is identical to the AYLF4500 element utilized in the cabin system, with the exception that it incorporates a special radio noise filter to minimize the pickup of radio noise from this blower motor. Aircraft 149 - 200 only has no cockpit thermostat.

3. **Duct Temperature Limiter 150°F**

(Cabin System Only)

Aircraft 1 - 79, 114 having ASC 104 and Aircraft 80 - 200, 322 and 323, have a 150°F thermal switch installed in the duct adjacent to the cabin anticipator between Fuselage Station 384 and 392.5 under floorboard 15. This switch enables the temperature control system for the cabin to correct without time delay when duct temperature rises above 150°F. This temperature rise usually occurs when the blower is cut into a system which has not been preconditioned by use of the APU. This limiter electrically shorts out the "lagged" portion of the anticipator in the cabin system. This action enables the system to move the cabin temperature control bypass valve toward cold without the time delay afforded by the "lagged" portion of the anticipator. When the duct temperature falls below 150°F, the limiter contacts open and returns the system to normal time-delay operation. (This only works in auto-temp control.)

4. **Cabin Air Temperature Indication**

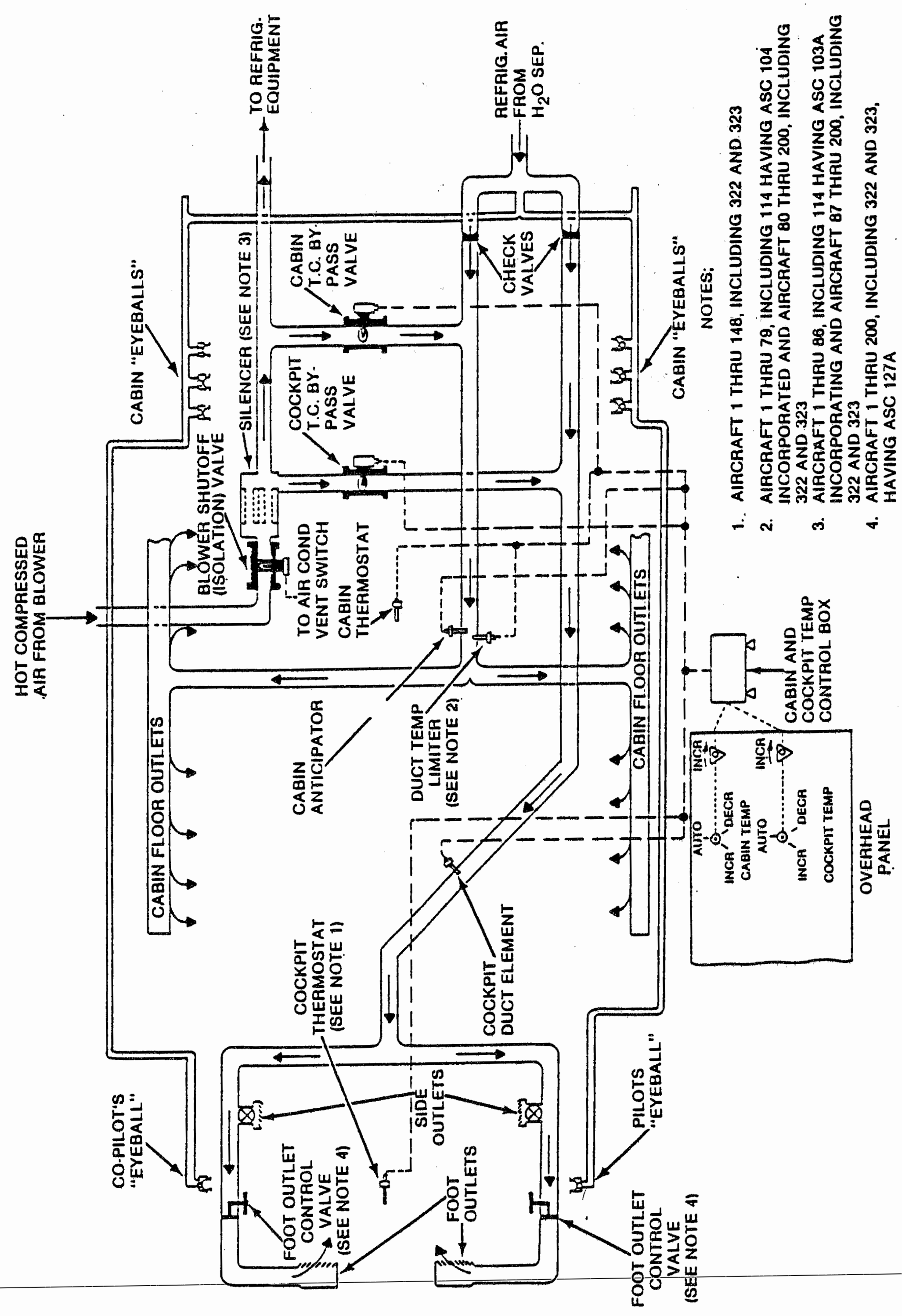
A cabin air temperature indicator, located in the upper overhead panel, adjacent to the temperature control switches, indicates the cabin temperature in degrees fahrenheit. It is of the electrical resistance type and measures the temperature by a bulb located near the cabin thermostat element. Two wires connect the bulb with the indicator. The indicator reads cabin temperature directly in degrees fahrenheit from − 20 to + 120. It receives power from the main dc bus through the CABIN TEMP circuit breaker. The cabin temperature indicator is inoperative in emergency dc operation.

5. Distribution System

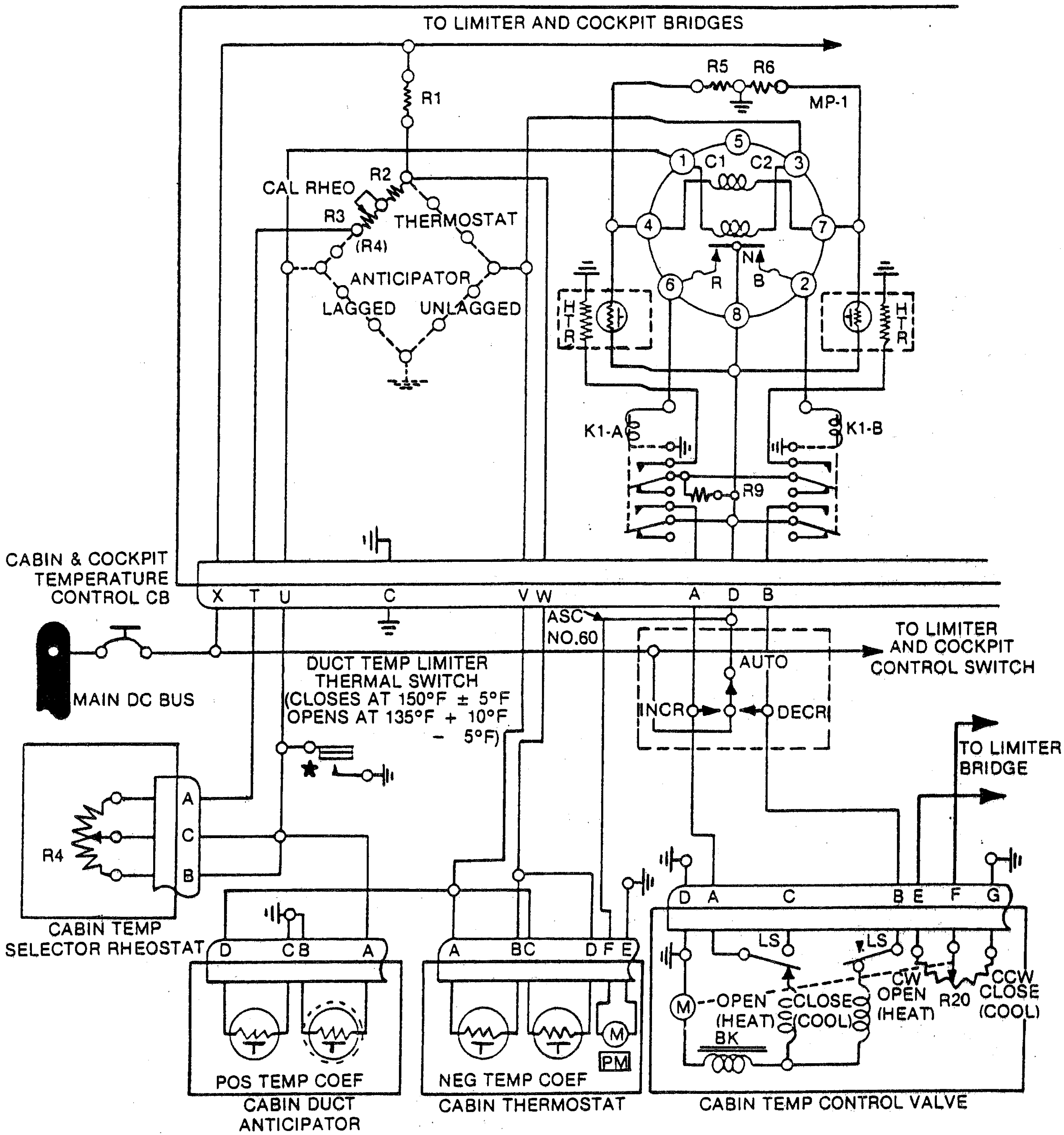
(See Figure 7)

The cabin distribution system consists of a continuous duct running the full length of the cabin at floor level on both sides. Each duct leads to louvered outlets spaced along its length which direct the air into the compartment. The cabin outlets are not adjustable by passengers. However, maintenance personnel can adjust these louvers individually in order to balance airflow distribution in the compartment. A number of holes are provided in the top surface of the duct at each window. These holes direct a portion of the air up the side walls to protect against "cold wall" effect and they also direct some cool air upward during the cooling operation. The flight station distribution system has two non-controllable foot outlets at the forward pressure bulkhead, one for the pilot and one for the copilot. Each has a side controllable outlet in the cockpit wall just above the side console. Aircraft 1 - 200, 322 and 323 having ASC 127 have valves which provide manual adjustment of the quantity of air applied to the foot outlets.

The cabin and cockpit are provided with "eyeball" supply ducts carrying cool air taken directly from the cold air line downstream of the water separator. "Eyeball" outlets are connected to this line when the aircraft is furnished.

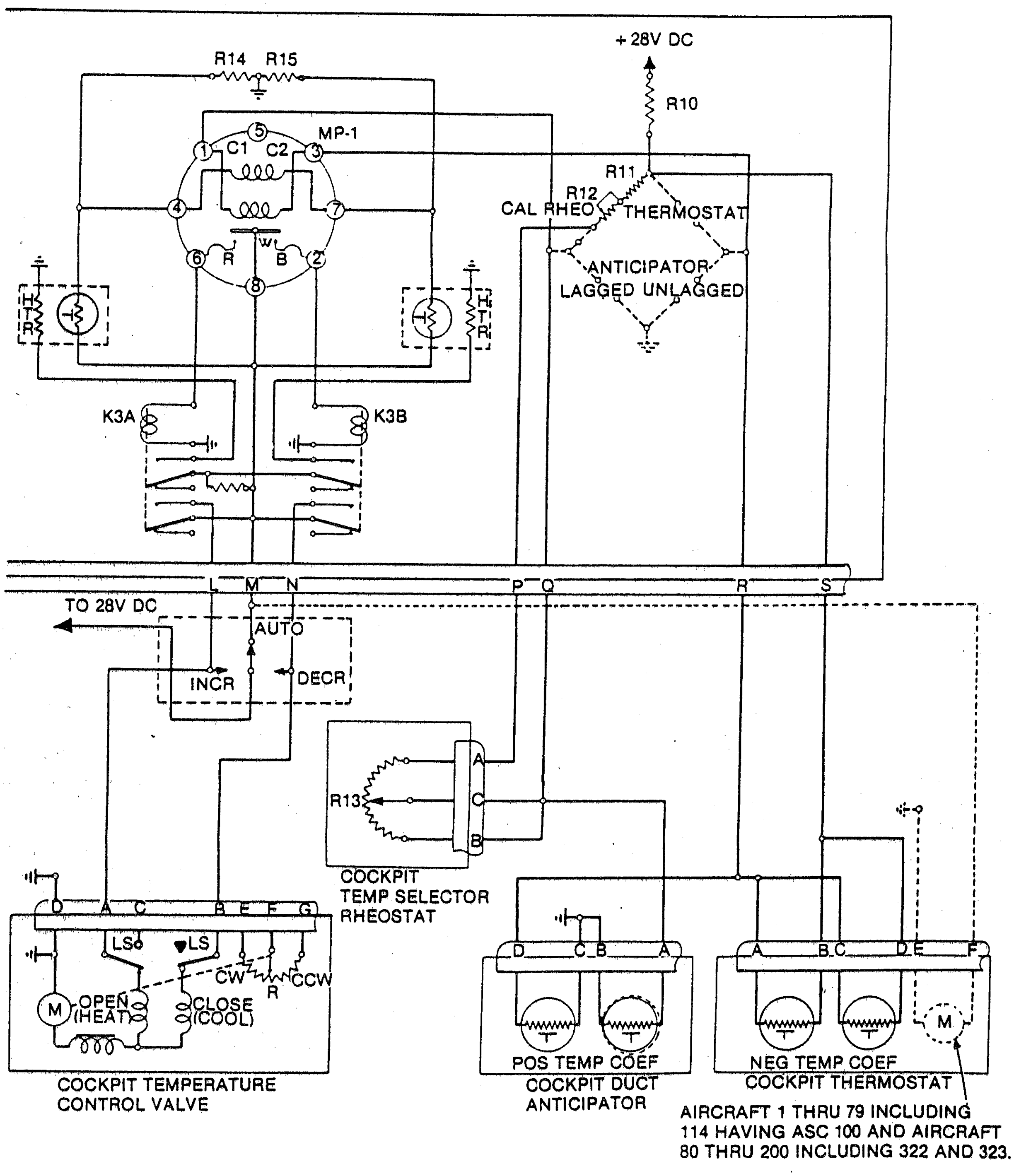


Temperature Control and Distribution System, Simplified Schematic Diagram
Figure 1.

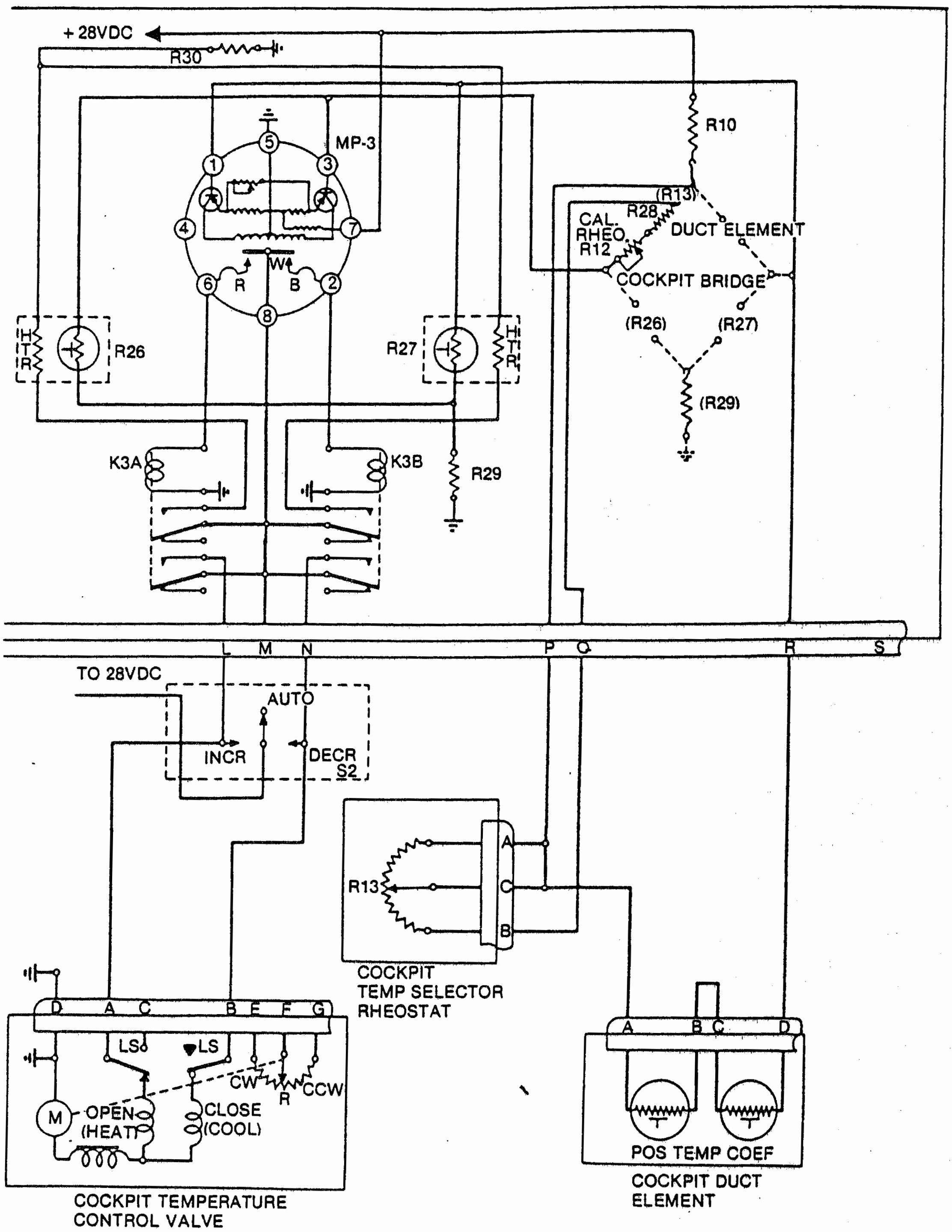


NOTE:
AIRCRAFT 1 THRU 79 INCLUDING
114 HAVING ASC 104 INCORPORATED
AND AIRCRAFT 80 THRU 200 INCLUDING
322 AND 323

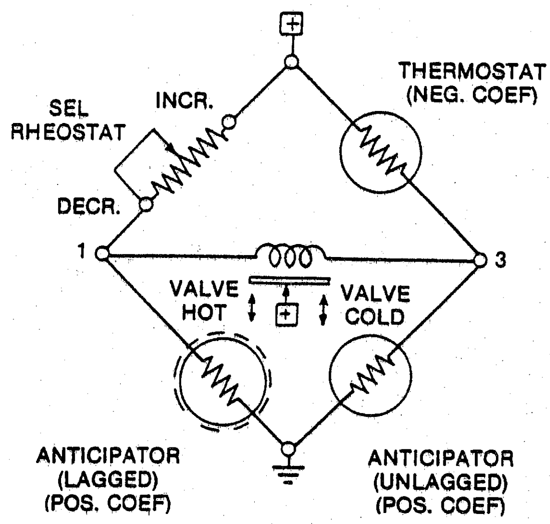
Cabin Temperature Control — Schematic
Figure 2.



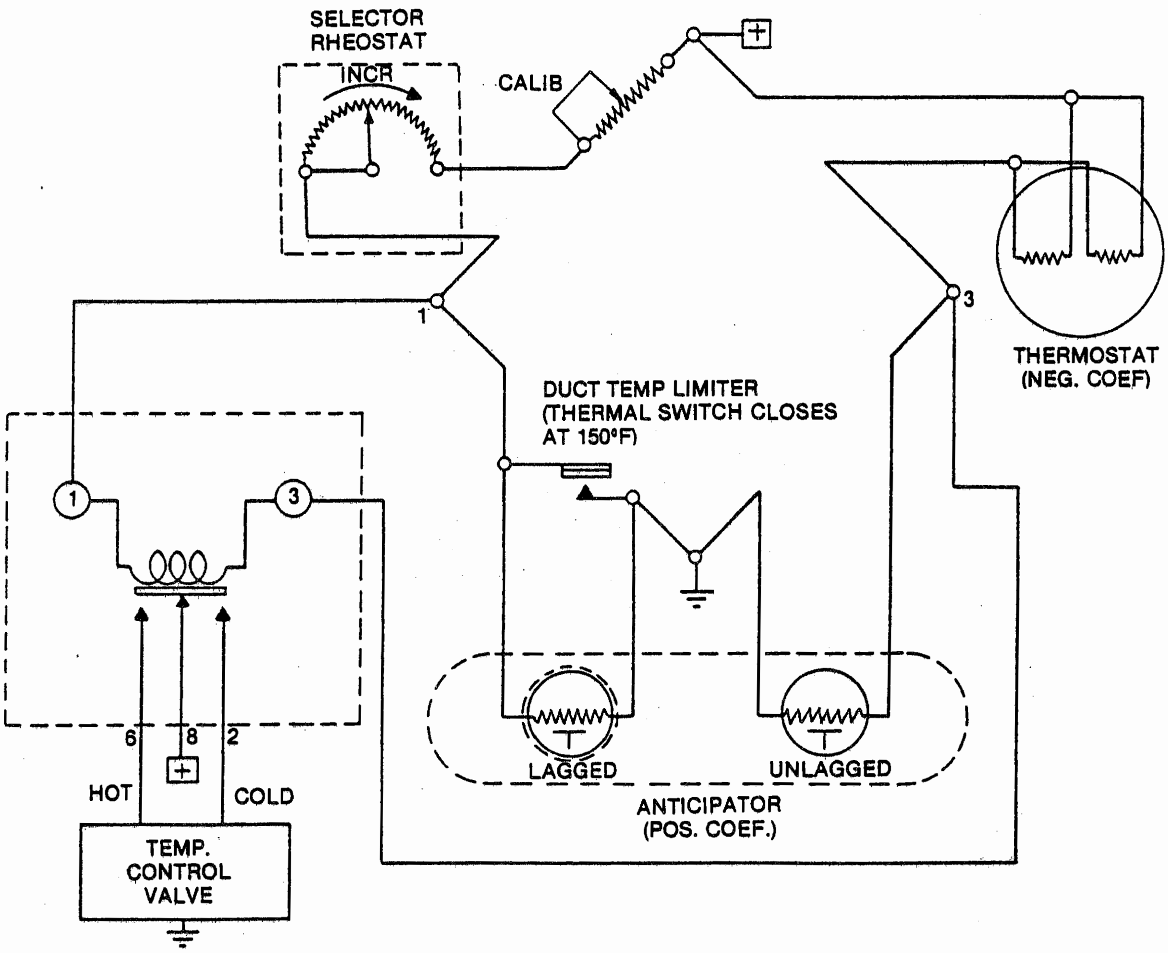
Cockpit Temperature Control — Schematic
Aircraft 1 - 148, 322 and 323
Figure 3.



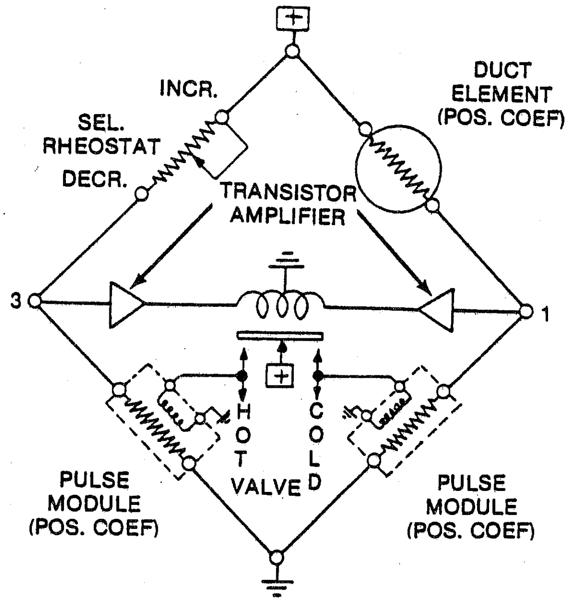
Cockpit Temperature Control — Schematic
Aircraft 149 - 200 only
Figure 4.



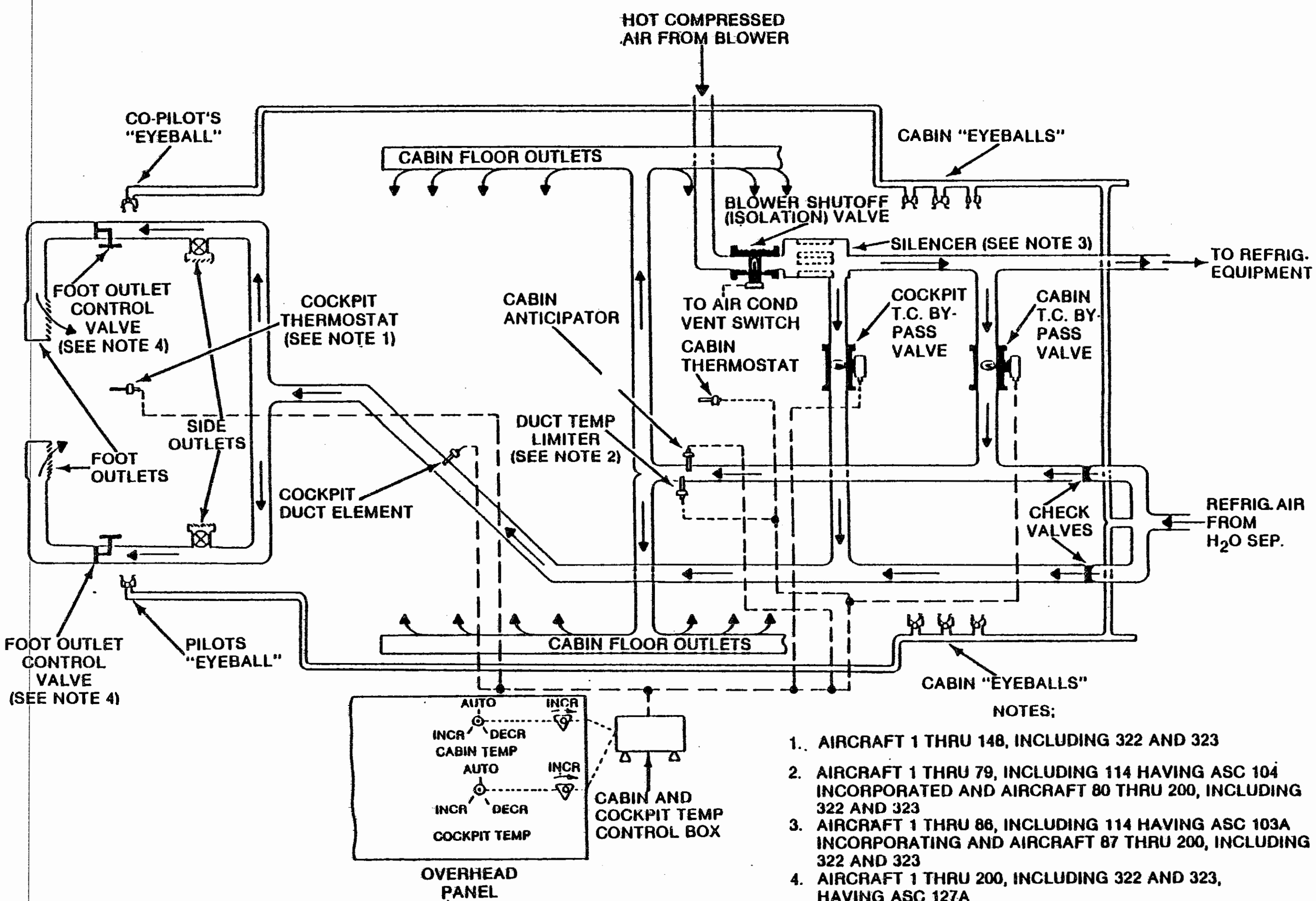
Basic Bridge — Schematic
Figure 5.



Actual Cabin Bridge — Schematic
Figure 6.



Transistorized Bridge — Schematic
Figure 7.



Temperature Control and Distribution System, Simplified Schematic Diagram
Figure 8.

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TEMPERATURE CONTROL SYSTEM — MAINTENANCE PRACTICES

1. Low Flow Bypass Valve — Removal / Installation

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to low flow bypass valve located in duct connecting bootstrap unit to secondary heat exchanger.

(2) Disconnect line from valve.

(3) Disconnect duct clamps at both ends of valve.

(4) Remove valve and O-rings.

B. Installation

(1) Inspect O-rings and replace as required.

(2) Connect regulated air source to valve fitting. With no air pressure applied to valve, position indicator should indicate closed.

(3) Slowly apply increasing air pressure to valve, valve should open before 7 psi is applied.

(4) Remove air pressure from valve, valve should close. Disconnect air supply from valve.

(5) Install valve with O-rings in duct and secure with clamps.

(6) Connect line to valve.

(7) Perform Low Flow Bypass Pressure Regulator — Operational Test, this Section.

2. Low Flow Bypass Pressure Regulator — Functional Test

A. Gain access to low flow bypass pressure regulator located in forward right side of tail compartment.

B. Locate two drain fittings for low flow sensor system,(bottom,aft, center of APU enclosure) and remove cap from right hand fitting. Install pressure gage on fitting.

C. Locate T-fitting inboard of pressure regulator and disconnect vertical line that connects to low flow sensor.

D. Cap off T-fitting using cap removed in Step B. above.

E. Pressurize deicing system pressure manifold 17 to 19 psi, as indicated on pressure gage in cockpit, using regulated external air source (25 to 40 psi) in either nacelle.

F. Pressure gage in tail compartment should read 11 $\pm$ 1.0 psi. If not , replace pressure regulator and repeat test.

G. Remove air source from deicing system pressure manifold and return system to normal configuration.

H. Remove cap from T-fitting and install vertical line.

I. Remove pressure gage from right hand drain fitting and reinstall cap removed in Step H above.

J. Remove external air source from deicing system manifold.

K. Inspect area for foreign objects, security of all attachments.

L. Close access cover.

3. Bypass Check Valve — Removal / Installation

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to turbine bypass check valve, in duct connecting bootstrap unit to primary heat exchanger in tail compartment; this is an in-line type check valve.

(2) Remove duct couplings at both ends of check valve; remove valve and associated seal and gasket.

B. Installation

(1) Inspect gasket and seal and replace if necessary.

(2) Install valve with associated seal and gasket, and tighten couplings at both ends.

(3) Start APU and apply air using normal procedure.

(4) Check valve couplings for leaks.

(5) Perform Low Flow Bypass System — Operational Test, this Section.

(6) Shutdown APU air and APU if no longer required.

4. Low Flow Bypass System — Operational Test

A. In tail compartment, with no pressure in pneumatic deicing system pressure manifolds, position indicator on turbine low-flow bypass valve should indicate a closed valve.

B. Disconnect 1/4 inch pneumatic control line from valve fitting.

C. Apply a slowly increasing regulated air pressure to valve fitting. By time this pressure reaches 7 psi, valve position indicator should indicate an open valve.

D. Remove rig and reconnect pneumatic control line to valve fitting.

E. Connect source of pressurized air to pneumatic deicing system in either nacelle. Deicing system pressure gage should read 15 - 18 psi.

F. With no air flowing through air conditioning system, bypass valve position indicator should indicate an open valve.

5. Cabin / Cockpit Temperature Control Box — Removal / Installation

CAUTION: BEFORE PERFORMING THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to control box by removing FLR-18.

(2) Disconnect electrical connector from control box.

(3) Remove and retain hardware attaching control box to mount.

(4) Remove control box.

B. Installation

(1) Install control box on mount with hardware previously removed and secure.

(2) Connect electrical connector to control box.

(3) Perform Temperature Control System — Operational Test, this Section.

6. **Cabin / Cockpit Temperature Control Valves — Removal / Installation**

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to temperature control valves, under FLR-15 (cockpit valve) or FLR-18 (cabin valve).

(2) Disconnect electrical plug.

(3) Disconnect duct couplings from valve.

(4) Remove and retain hardware holding valve to mount.

(5) Remove valve and O-rings.

B. Installation

(1) Install valve on mount with hardware previously removed and secure.

(2) Inspect O-rings and replace if required; install in couplings and connect ducts.

(3) Connect electrical plug.

(4) Perform Temperature control System — Operational Test, this Section.

(5) Install floorboard previously removed.

7. **Cabin / Cockpit Duct Element / Anticipators — Removal / Installation**

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

NOTE: Aircraft 1 - 148, 322 and 323 have a cabin and cockpit anticipator. Aircraft 149 - 200 have a cabin anticipator and a cockpit duct element in place of a cockpit anticipator; removal and installation procedures are the same.

A. Removal

(1) Remove FLR-15 to gain access to anticipators and/or duct element.

NOTE: Cabin anticipator is just forward of refrigerated air line junction and duct downstream of cabin temperature control bypass valve; cockpit anticipator (or duct element) is just to the left of flap central gearbox in cockpit distribution ducting.

(2) Disconnect electrical plug; remove mounting hardware and remove unit.

B. Installation

(1) Install unit into duct using hardware previously removed.

(2) Connect electrical plug.

(3) Install FLR-15.

(4) Start and air load APU and after compartment temperature stabilizes in AUTO, check that temperature control rheostat operation reacts normally for that compartment involved.

(5) Shut down air and APU if no longer required.

8. **Cabin / Cockpit Thermostat — Removal / Installation**

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to thermostat as follows:

(a) Cabin thermostat, all aircraft. Location is determined by outfitting agency.

(b) Cockpit thermostat for Aircraft 1 - 79 and 114 not having ASC 100 is located on the copilots relay box (Fuselage Station 133).

(c) Cockpit thermostat for Aircraft 80 - 148, 322 and 323, excluding 114 having ASC 100, is located on right side of pedestal kick plate.

(d) Aircraft 149 - 200, have no cockpit thermostat.

(2) Disconnect electrical connector.

(3) Remove hardware securing thermostat to structure and remove thermostat.

B. Installation

NOTE: If cabin thermostat is to be replaced, transfer cabin temperature bulb to new unit.

(1) Install thermostat on structure and secure with hardware.

(2) Connect electrical connector to thermostat.

(3) Energize main and essential dc buses.

(4) Check that blower in blower type thermostat operates when associated TEMP CONTROL SELECTOR is placed in AUTO and does not operate in OFF.

(5) If cabin thermostat assembly is replaced, ensure cabin air temperature gage is indicating compartment temperature.

(6) Start and operate APU and air load using normal procedure

(7) With associated system in AUTO operation, and after temperature in associated compartment stabilizes, check that temperature control rheostat reacts normally for compartment involved.

(8) Shut down air and APU.

(9) De-energize main and essential dc buses.

(10) Inspect area for foreign objects, security of all attachments.

(11) Close access previously opened.

9. Cabin Thermostat — Cleaning

A. Remove electrical power from aircraft.

B. Gain access to thermostat.

C. Disconnect plugs and remove thermostat. **CAUTION:** DO NOT USE ANY SHARP TOOLS OR SCRAPING ACTION ON THERMOSTAT ELEMENT (LOOKS LIKE A WIRE WOUND COIL) THE CABIN AIR TEMPERATURE BULB MAY BE ATTACHED TO THERMOSTAT BRACKET AND CARE SHOULD BE TAKEN NOT TO DENT OR DAMAGE SENSING PORTION OF PROBE.

D. Depending on amount of dirt on thermostat element, clean using a small amount of cleaning fluid and blowing out with air. Particular care should be taken when cleaning airflow passages and actual sensing element.

E. Connect plugs and install thermostat.

F. Check operation of thermostat blower and thermostat using APU air with CABIN TEMP CONTROL SELECTOR switch in AUTO position.

G. Shut down air and APU.

10. Cabin Air Temperature Indicator — Removal / Installation

CAUTION: BEFORE PERFORMING THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Remove knobs from cabin and cockpit temperature control rheostats.

CAUTION: DO NOT USE FORCE TO REMOVE LIGHTING PANEL AND AVOID TWISTING OR BENDING PANEL.

(2) Remove screws that secure right side edge lighting panel to overhead panel and remove panel.

(3) Remove screws securing overhead panel to structure and move panel away from structure. Support panel to prevent strain on wiring and lines.

(4) Disconnect electrical connector from indicator.

(5) Remove screws securing indicator and remove indicator from panel.

B. Installation

(1) Install indicator in panel and secure with screws.

(2) Connect electrical connector to indicator.

(3) Install overhead panel on structure and secure with screws.

(4) Install edge lighting panel on overhead panel and secure with screws,

(5) Install knobs on cabin and cockpit temperature control rheostats.

(6) Energize main and essential dc buses.

(7) Check cabin temperature indicator for proper indication.

(8) Check overhead edge lighting panel for proper operation, use lighting control on overhead panel.

(9) De-energize main and essential dc buses.

11. Cabin / Cockpit Temperature Control Rheostats — Removal / Installation

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

NOTE: The cabin and cockpit rheostats are identical units.

(1) Remove knobs from cabin and cockpit temperature control rheostats.

CAUTION: DO NOT USE FORCE TO REMOVE LIGHTING PANEL AND AVOID TWISTING OR BENDING PANEL.

(2) Remove screws that secure edge lighting panel to right side overhead panel and remove panel.

(3) Remove screws securing overhead panel to structure and move panel away from structure. Support panel to prevent strain on wiring and lines.

(4) Tag and disconnect electrical leads from rheostat.

(5) Remove locknut from front of shaft and remove rheostat.

B. Installation

(1) Install rheostat in panel and secure with locknut.

(2) Connect electrical leads to rheostat and remove tags.

(3) Install overhead panel on structure and secure with screws.

(4) Install edge lighting panel on overhead panel and secure with screws.

(5) Install knobs on cabin and cockpit temperature control rheostats.

(6) Energize main and essential dc buses.

(7) Start APU and air load using normal procedures.

(8) With TEMP CONTROL SELECTOR for compartment which has rheostat replaced, place system in the AUTO operation. Set rheostat at midpoint (12 o'clock position) and allow temperature to stabilize.

(9) When temperature is stabilized turn rheostat to full decrease (counterclockwise). Check that applicable compartment gradually receives cooler air.

(10) Turn rheostat to full increase (clockwise). Check that applicable compartment gradually receives warmer air.

(11) Return rheostat to midpoint. System should return to original temperature.

(12) Shut down air and APU.

(13) Check overhead edge lighting panel for proper operation, use lighting control on overhead panel.

(14) De-energize main and essential dc buses.

12. Temperature Control System — Operational Test

A. Energize main and essential dc bus.

B. Depress CABIN/COCKPIT TEMP circuit breaker.

C. With man in cockpit and man observing ram air limiter louvers from outside aircraft (look in ram air duct external exhaust outlet).

D. Place CABIN TEMP CONTROL switch to full decrease and hold at least 50 seconds, louvers should open.

NOTE: Louver actuator may pulse, this is normal.

E. Place CABIN TEMP CONTROL switch to full increase and hold for at least 50 seconds, louvers should close. Outside man returns to cabin.

F. Place both CABIN and COCKPIT TEMP CONTROL switches to decrease and hold for at lease 50 seconds: leave both switches off.

G. Place both cabin and cockpit rheostats to midpoint.

H. Start and air load APU; use full air load (increase - decrease valve in baggage compartment to full increase, or flight ground switch to FLIGHT).

I. Allow air flow and temperature to stabilize and check that cold air is emitting from both cabin and cockpit outlets.

J. Place CABIN TEMP CONTROL switch to increase and hold; have second man check for increasing air temperatures in cabin outlets only, cockpit outlets remain cold.

K. Place CABIN TEMP CONTROL switch to decrease; cabin duct air temperature should gradually become cold and cockpit should remain cold.

L. Repeat same procedure for cockpit system using COCKPIT TEMP CONTROL switch; cabin should not be affected.

M. Place both CABIN and COCKPIT TEMP CONTROL switches to AUTO; observe that both cabin and cockpit outlet air temperatures gradually stabilize to prevailing conditions and conditions and midpoint temperature setting.

N. Place cabin rheostat to decrease (full counterclockwise); cabin air temperature should decrease accordingly and cockpit temperature should not change.

O. Place cabin rheostat increase (full clockwise); cabin air temperature should gradually increase and cockpit temperature should not change.

P. Place cabin rheostat to midpoint, air temperature in cabin should return to original midpoint selection.

Q. Repeat Steps N thru P using Cockpit rheostat; determining that cockpit system reacts correctly to rheostat position and cabin temperature should not change.

R. Shut down air and APU.

S. Return all controls and switches to desired position.

T. De-energize main and essential dc bus.

BOOTSTRAP UNIT — DESCRIPTION / OPERATION

1. Expansion Turbine

The last thing the air passes through in a bootstrap cycle is the expansion turbine of the bootstrap unit. As the air passes through the turbine, its pressure drops to the turbine discharge pressure and the turbine rotates. (See Figure 1). The energy to rotate the turbine comes from the air, since air is the only thing in contact with the turbine blades. The air thereby gives up its energy to the turbine. As a result, energy content of the air is decreased and its temperature drops. The greater the pressure ratio across the turbine, the greater the energy lost to the turbine (as evidenced by increased turbine speed) and the greater the temperature drop.

To provide the largest possible pressure ratio across the cooling turbine and thereby produce the lowest possible turbine discharge temperature, the bootstrap cooling unit has a centrifugal compressor attached to the same shaft as the cooling turbine. The energy taken out of the air by the turbine is used by the compressor to boost the pressure of the air being fed into the turbine above the pressure level provided by the primary pressure source for the system; (in this case the blower or APU). It is for this reason that the bootstrap unit is used with air sources of limited pressure capacity and it is for this reason that the name "bootstrap" is used. The refrigeration system, in effect, picks itself up by its own bootstraps in regards to pressure.

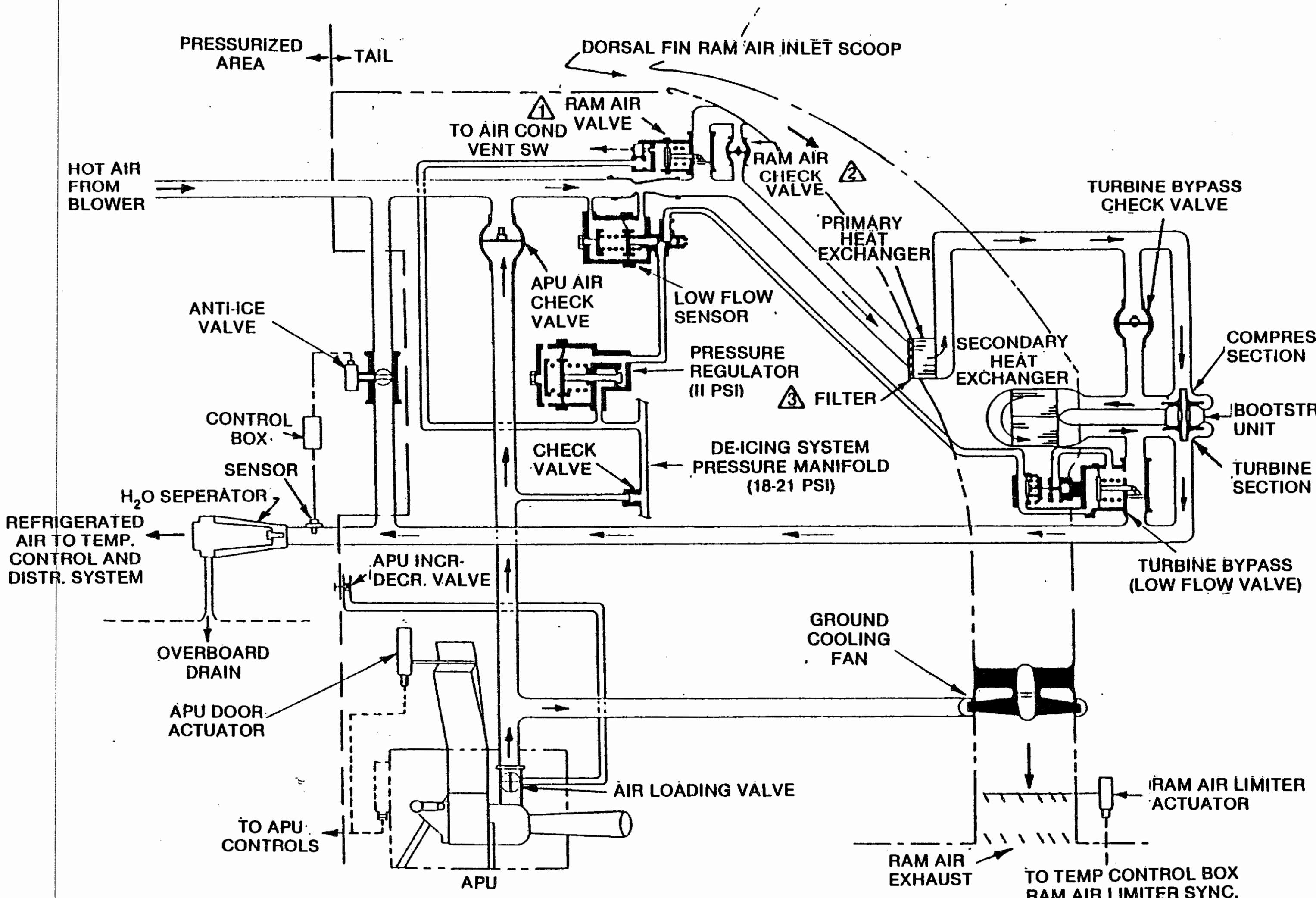
2. Turbine (Low Flow) Bypass System

The air path through the cooling equipment offers much more resistance to flow than does the path through the temperature control bypass valves. As a result, if the temperature control bypass valves are closed for full cooling so that all the blower's output is forced through the cooling equipment, blower discharge pressure increases. If this situation occurs when the cabin is pressurized, blower discharge pressure must go even higher. As a result, during the hot days at lower altitudes, the blower discharge pressures reach the limitation, (12.5 $\pm$ 0.5 psi) and the differential pressure sensor limits blower pressures by actuating the spill valve to spill excessive pressures overboard. This spillage reduces the airflow into the cabin and would produce a cabin pressure decrease if it were not for the turbine (low-flow) bypass system. At higher altitudes, blower airflow is reduced, and the blower discharge pressure does not reach its limitation.

The flow through the cooling path is measured by the low-flow sensor. This device consists of a venturi section and a spring-loaded-open slide valve actuated by a diaphragm. The low fuel sensor is located under the right side of the APU base in the tail compartment. (Aircraft 1 - 84 and 114 having ASC 130 and Aircraft 85 - 200, 322 and 323, have the low flow sensor located aft of APU.) Fed into the opposite sides of the diaphragm are the pressure upstream of the venturi and the pressure in its throat. If the flow through the cooling path is large, the difference between these pressures is large and the slide valve is forced to close off the flow path through the slide valve. With the slide valve closed, actuating pressure is prevented from reaching the control head of the low-flow (turbine) bypass valve, which (being spring-loaded closed) remains shut. With this valve shut, the air leaving the secondary heat exchanger is forced to go through the bootstrap turbine. The turbine rotation drives the bootstrap compressor which then produces a higher pressure at its outlet than at its inlet. This pressure difference holds the low-flow (turbine) check valve closed. This is the normal situation with the air passing through the bootstrap unit. (Full cooling.)

When the low-flow sensor diaphragm senses a small pressure difference between the upstream section and the throat of the venturi, indicating a small airflow, the spring moves the slide valve to open and allows actuating pressure to reach the control head of the low-flow bypass valve. The bypass valve is then forced open against a spring. Air flowing out of the secondary heat exchanger is then bypassed around the bootstrap turbine. As a result, the turbine slows and, of course, so does the bootstrap compressor. As the compressor slows, the pressure difference it creates becomes smaller and finally the check valve opens, bypassing air from the primary heat exchanger around the compressor. Thus, the entire bootstrap unit is bypassed and air flows directly from the primary heat exchanger, through the secondary heat exchanger to the cabin. Since the bootstrap unit makes up the greater portion of the cooling equipment's resistance to flow, the resistance of the flow path as seen by the blower is greatly decreased. The blower discharge pressure and temperature decrease, spilling ceases and all the blower's output is again delivered to the compartments. Airflow is maintained while at the same time so is cooling capacity, since the low-flow bypass system only goes into operation at altitude where heat exchanger cooling is increased due to the low coolant (ram air) temperature.

Actuating pressure for the low-flow bypass system is supplied from the pressure manifold of the pneumatic boot-deicing system through a pressure regulator, identical to that used in the blower safety and unload system.



1. Aircraft 1 through 148, including 322 and 323
2. Aircraft 149 through 200, excluding 322 and 323
3. Aircraft 1 through 84, including 114 having ASC 132 incorporated and aircraft 85 through 200, including 322 and 323.

Refrigeration, Dehumidification and APU Supply Systems — Schematic
Figure 1.

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BOOTSTRAP UNIT — MAINTENANCE PRACTICES

1. **Low Flow Sensor Control — Service**

   A. In tail compartment, locate 1/4 inch rigid tubing connecting low flow sensor and low flow bypass valve, also locate 1/4 inch line connecting pressure regulator to sensor.

   NOTE: On Aircraft 1 - 84 and 114 having ASC 130, Aircraft 85 - 200, lines come together and meet at low point in system and are secured to bracket located at roughly centerline of Fuselage Station 635. On Aircraft 1 - 84 not having ASC 130, locate line at lowest point in system.

   B. Remove caps from drains at low point in system and drain accumulated moisture from lines.

   C. Install caps (or connect lines) when draining is complete.

2. **Bootstrap Unit — Removal / Installation**

   CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

   A. Removal

   (1) Gain access to bootstrap unit in tail compartment.

   (2) Remove duct couplings connecting bootstrap to ducting. Check condition of clamps, replace if damaged.

   (3) Support unit and remove mounting bolts; remove bootstrap being careful not to tip bootstrap as oil can run out of breather tube.

   (4) Remove O-rings and discard.

   (5) Drain oil if unit is being replaced with another unit and remove breather tube and adapter fitting.

   B. Installation

   NOTE: If new or replacement unit is to be installed, transfer adapter fitting to top port and install as in old unit. Connect breather tube to adapter and, if required, use new O-ring before installing adapter fitting.

   (1) Mount unit using hardware previously removed; shim with washers as required to ensure snug fit and tighten bolts.

   (2) Connect ducts to unit using new O-rings (2 each).

   (3) Install V-band clamps and secure. Torque NUCO clamp (P/N N3290S5-565SH) 40 - 50 inch pounds and Aeroquip clamp (P/N 43907-565SH) 35 - 40 inch pounds.

   CAUTION: BOOTSTRAP UNITS ARE SHIPPED DRY AND MUST BE LUBRICATED BEFORE PLACING IN SERVICE. USE ONLY OILS SPECIFICALLY APPROVED BY BRAND NAME AND NUMBER BY GARRETT FOR THIS UNIT. DO NOT INTERMIX DIFFERENT BRANDS OF OIL EVEN IF BOTH ARE ON APPROVED LIST.

   (4) Remove twist-lock type dipstick and fill to dipstick FULL mark with oil, install dipstick and ensure that it is locked in position.

   (5) Start and airload APU; check for leaks at duct couplings.

   (6) Shut down air and APU if no longer required.

3. **Bootstrap Unit — Oil Check / Service**

CAUTION: DO NOT INTERMIX ONE BRAND OF OIL WITH THAT OF ANOTHER BRAND WHEN ADDING OR CHANGING OIL.

NOTE: Check bootstrap oil level in accordance with times specified in the Chapter 5. If oil is low replenish with oil approved by Garrett. (See Chapter 12 SERVICING for specific manufacturers of lubricants approved by Garrett.)

A. Gain access to bootstrap unit in tail compartment.

B. Remove twist-lock type dipstick between scroll assemblies and check oil level.

NOTE: If same brand of oil is not available, it is recommended that unit be drained, flushed and new oil installed.

C. Inspect dipstick O-ring and replace if necessary.

D. Install dipstick and ensure it is firmly locked in place.

4. **Bootstrap Unit — Oil Change**

NOTE: Change bootstrap oil in accordance with times specified in Chapter 5 (described in the following steps).

A. Gain access to bootstrap unit in tail compartment.

B. Remove filler dipstick to facilitate draining.

C. Remove safety wire and drain plug (drain plug is at the six o'clock position between two scroll assemblies below dipstick port). Discard drain plug O-ring.

D. Drain oil into glass jar.

E. Inspect drain plug and replace as necessary.

F. Install drain plug with new O-ring and safety-wire.

G. Inspect drained oil for water. If water is detected in an appreciable amount, it is suggested that oil be changed at more frequent intervals than required.

5. **Bootstrap Unit Inboard Impeller — Clean / Inspect**

A. Gain access to bootstrap unit in tail compartment. (See Figure 201)

B. Remove duct from bootstrap, compressor inlet, retaining clamp and discard O-ring.

C. Remove duct from bootstrap, compressor outlet, retaining clamp and discard O-ring.

NOTE: If excessive amount of debris is evident at compressor, it is recommended that the heat exchangers be inspected and cleaned.

D. Apply vacuum cleaner nozzle to compressor outlet to remove any debris. Use stiff nylon bristle brush at compressor inlet to dislodge any debris from impeller.

E. Apply vacuum to compressor inlet to remove any debris from impeller.

F. Inspect compressor impeller and surrounding area for damage.

G. Inspect duct clamps for damage. Replace clamp(s) if any damage is noted.

H. Install new O-ring at compressor outlet. Position duct and install clamp. Torque clamp to 35 - 40 inch pounds.

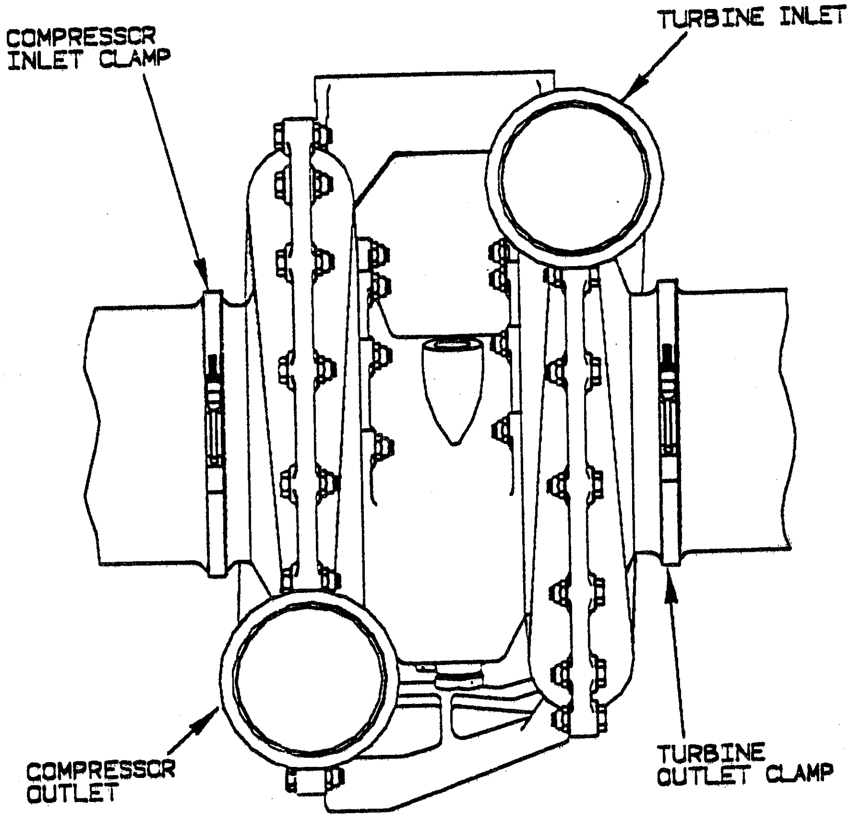
I. Install new O-ring at compressor inlet. Position duct and install clamp.

J. Torque compressor inlet clamp to one of the following values. NUCO clamp (P/N N3290S5-565SH) 40 - 50 inch pounds, Aeroquip clamp (P/N43907-565SH) 35 - 40 inch pounds.

K. Inspect area for foreign objects, leaks and damage.

L. Start APU and apply air load. Inspect ducting for leaks.

M. Shut off air and shut down APU.



Bootstrap (View Looking Aft)
Figure 201.

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HEAT EXCHANGERS — DESCRIPTION / OPERATION

1. Heat Exchangers

Since the compressor is a device that puts mechanical energy into the airstream, it raises the air temperature at the same time it is increasing the pressure. To lower the air temperature from which the expansion turbine will start the final temperature reduction, air leaving the compressor is passed through the secondary heat exchanger. Here it is cooled by losing energy through heat transfer to the coolant (ram air) stream. Thus, its high pressure is retained, for the expansion through the turbine while its temperature is reduced almost to ambient. The temperature drop through the turbine can then lower the air temperature well below ambient.

The primary heat exchanger is employed in a similar manner. The air from the blower (or APU) having been compressed, is hot and if it were supplied to the bootstrap compressor inlet at this temperature, the compressor discharge temperatures would be excessive. Instead, the primary heat exchanger cools down the blower (or APU) supplied air before it is recompressed.

At first glance the bootstrap cycle may seem like a perpetual motion machine. Of course it is not. It is continually giving up energy to the coolant (ram air) stream and naturally has frictional losses. It is up to the primary pressure source, the blower (or APU) to provide for these losses plus maintaining the cabin pressure level.

On aircraft 1 - 84, and 114 having ASC 132 and Aircraft 85 - 200, 322 and 323 a filter is added to the inlet side of the primary heat exchanger in the tail. Its purpose is to prevent any solid contaminant, from possibly plugging of the heat exchanger cores, and also keeping it out of the bootstrap cooling turbine. It is installed in the inlet flange of the primary heat exchanger.

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HEAT EXCHANGERS — MAINTENANCE PRACTICES

1. **Primary / Secondary Heat Exchanger — Removal / Installation**

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

NOTE: Both heat exchangers are removed as a unit from aircraft, then further separation can take place.

(1) Remove APU from aircraft.

(2) Disconnect duct couplings from rear end of heat exchanger package and duct coupling from forward end.

(3) Loosen band clamp securing large sleeve to overboard duct assembly below the package.

(4) At top of package, remove bolts securing ram air inlet duct assembly.

(5) Support heat exchanger package and remove bolts (11 top and 11 bottom), which secure package to support brackets.

(6) Remove bolts from forward and aft support brackets.

(7) Remove package from aircraft through opening used for APU removal.

(8) Separate primary from secondary heat exchanger as required by removing associated hardware.

B. Installation

(1) Assemble primary and secondary heat exchangers and associated assemblies to form package (if assembly was broken down).

NOTE: If primary heat exchanger is to be replaced, remove screen and gasket from inlet panel. Clean screen and install in new unit before installing in aircraft. Screen and extra gasket on inlet side are installed in following sequence; gasket, screen, gasket, pan.

(2) Install heat exchanger package through APU opening and position to pick up forward and aft support brackets.

(3) Secure forward and aft supports.

(4) Install 11 top and 11 bottom bolts which secure package to upper and lower support brackets.

(5) Install bolts at top of package, securing ram air inlet to package.

(6) At bottom, secure large sleeve to bottom of package with band clamp.

(7) Inspect seals which were removed from couplings and replace as necessary.

(8) Install seals and connect rear couplings and forward coupling.

(9) Install APU.

(10) Start APU and air load using normal procedures.

(11) Check all duct couplings and heat exchanger package joints for leaks.

(12) Shut down APU air and APU if no longer required.

2. **Heat Exchanger Unit — Cleaning**

A. To check heat exchanger unit for excessive accumulation of solid matter, remove forward end cover of primary heat exchanger.

B. If unit requires cleaning, it can be accomplished in the following manner:

(1) Remove ducting from forward and aft end of heat exchanger unit.

(2) Remove aft end cover of primary heat exchanger and forward and aft end covers of secondary heat exchanger.

(3) Use air hose to blow out accumulated matter from each heat exchanger tube and ducts; at the same time use a vacuum cleaner to take in dirt being blown out by air hose.

(4) When cleaning has been completed, install forward and aft end covers of primary and secondary heat exchangers.

(5) Attach ducting to heat exchanger unit using seal and clamp assemblies.

3. Primary Heat Exchanger Screen — Inspection

NOTE: If excessive debris is evident on screen, it is suggested heat exchangers and bootstrap compressor impeller be inspected and/or cleaned.

A. Gain access to primary heat exchanger (smaller of two) in tail compartment.

B. Remove duct clamp from forward primary heat exchanger port. Remove duct, retain clamp and discard O-ring.

C. Remove bolts securing forward primary heat exchanger pan. Remove pan.

D. Remove screen with screen gaskets.

E. Clean screen with vacuum cleaner and inspect for condition, replace as necessary.

F. Clean pan and visible heat exchanger interior of any debris.

G. Inspect screen gaskets and replace as necessary.

H. Install gasket, screen, gasket and pan and secure with bolts. Torque bolts to 20 - 25 inch pounds.

I. Inspect duct clamp and replace if necessary.

J. Install new O-ring and connect duct with clamp. Torque clamp to 25 inch pounds.

K. Start APU and apply air load. Check heat exchanger seal joints and clamps for leaks.

L. Shut down air and APU.

WATER SEPARATOR — DESCRIPTION / OPERATION

1. **Water Separator**

As noted, the expansion through the cooling turbine reduces turbine discharge temperature below ambient temperature. Unless the day is a very dry one, the low discharge temperature forces some of the moisture in the air to condense. This moisture would proceed to the cabin and produce high relative humidity if not removed.

A water separator is installed to reduce humidity in the aircraft when cooling is required. The water separator provides a mechanical means of water removal and consists of two sections. The inlet section is a "coalescer". It makes a few large drops from a lot of small droplets by passing the droplet laden airstream through a coarse mesh nylon bag. The second section accomplishes the actual removal. The airstream is forced to swirl by means of a series of vanes so that the large drops are spun to the outside walls where they drain overboard through a flush port in the lower fuselage belly. The separator removes approximately 80 percent of all water passing through it in the form of liquid water. It cannot remove water vapor. It can only remove that vapor which has condensed to liquid water. The water separator also contains a relief valve which, if the nylon bag is clogged, will bypass the air. In this case, dehumidification will not take place.

2. **Water Separator Anti-Ice System**

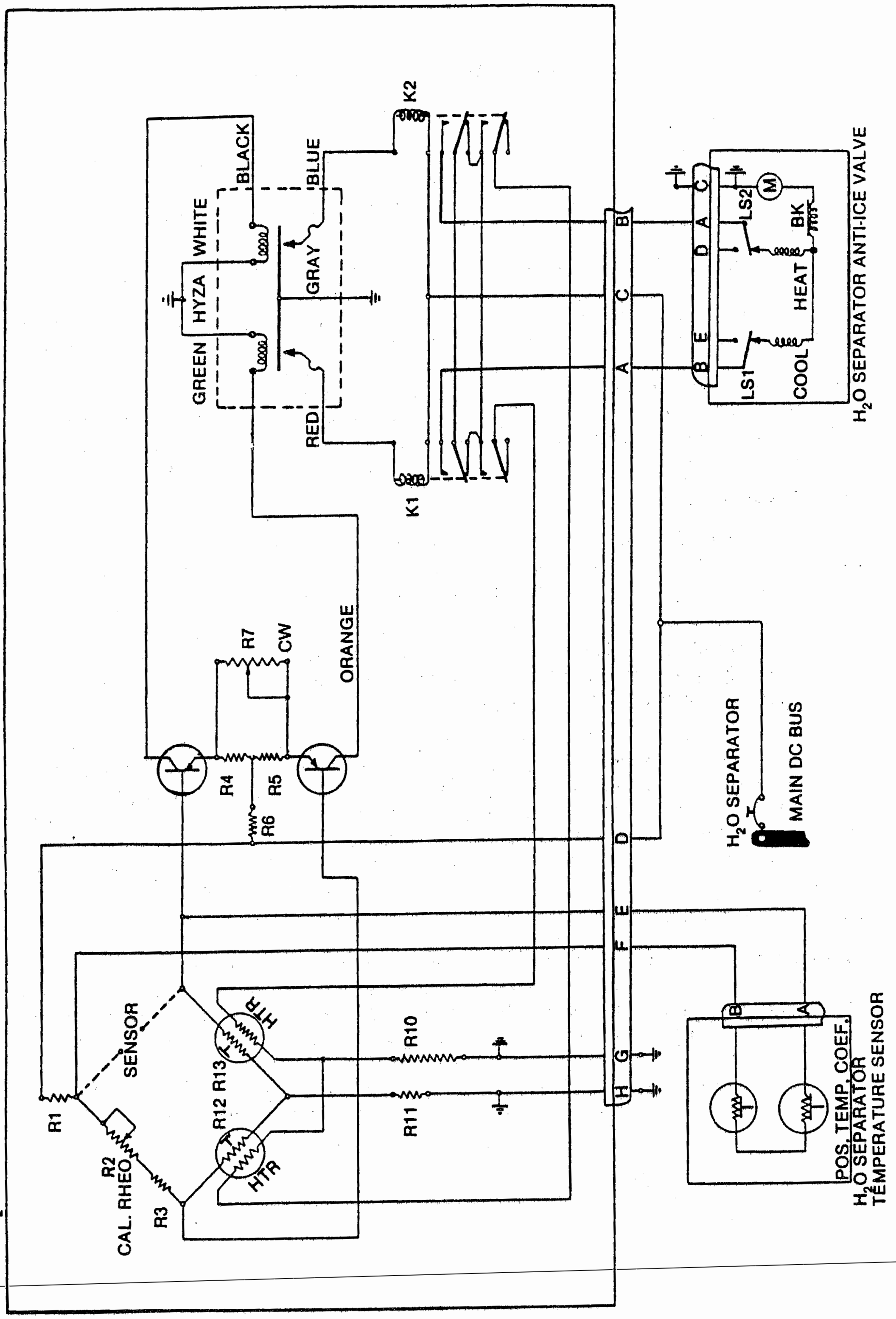
On a cool, moist day, cooling turbine discharge temperature fall low enough so that water is not only condensed, but frozen. To prevent the nylon bag of the water separator from becoming clogged with ice crystals and restricting cabin airflow, a water separator anti-icing system is installed.

This system consists of a sensor in the duct upstream of the water separator which, operating through a bridge circuit in a control box, position the water separator anti-icing valve. (See Figure 1) When opening, this motor-driven butterfly type modulating valve bypasses hot air around all of the cooling equipment and mixes it with the refrigerated air approaching the water separator.

The water separator anti-icing system is set to prevent water separator inlet air temperature from dropping below 34°F. The valve closes when the air temperature reaches 40°F, thus maintaining 34°F to 40°F temperature. In the event of a failure in the water separator anti-ice system, the separator relief valve would bypass the nylon bag before the pressure difference bursts the bag. The water separator anti-icing valve is located in the tail compartment, Fuselage Station 588. (Aircraft 1 - 80 and 114 having ASC 128A, Aircraft 81 - 200, 322 and 323, the anti-icing valve is located under floorboard 24 and 25 forward of the pressure bulkhead, Fuselage Station 539). The water separator anti-ice system is inoperative in emergency dc operation.

3. **Air-Cycle Refrigeration**

To understand air cycle refrigeration equipment, two principles of thermo-dynamics must be understood. First, the temperature of a mass of air depends upon its energy content; low energy content, low temperature; high energy content, high temperatures. The second is that energy can be removed from the air (and its temperature thereby lowered) in either of two ways. It can be removed by the transfer of heat out of the air without the air undergoing any change of pressure or it can be removed in the form of mechanical work as the air undergoes a pressure drop. In both cases air temperature decreases. Similarly, energy can be added to air and its temperature thereby raised either by heat transfer or by doing mechanical work on the air.



Water Separator Control — Schematic
Figure 1.

WATER SEPARATOR — MAINTENANCE PRACTICES

1. **Water Separator / Separator Bag — Removal / Installation**

   A. Removal

   NOTE: To clean water separator bag it is necessary to remove water separator from aircraft and disassemble it.

   (1) Gain access to water separator by removing FLR-18.

   (2) Remove screws, washers and nuts securing control cable fairlead to floor support at Fuselage Station 448, inboard of water separator.

   (3) Disconnect duct coupling at aft end of separator.

   (4) Disconnect duct couplings at forward end of duct section attached to water separator.

   NOTE: Duct section and water separator are removed as a unit.

   (5) Disconnect overboard drain line from lower end of separator.

   (6) Disconnect clamps supporting duct section and clamps supporting to fuselage.

   (7) Pull control cables inboard to provide clearance.

   (8) Remove water separator.

   B. Installation

   NOTE: If replacement separator is to be installed, transfer ducts and drain fitting to new unit and orient in same direction.

   (1) Inspect duct sleeves for condition and replace as required.

   (2) Pull control cables between Fuselage Stations 448 and 478 to provide clearance for installing water separator and duct section.

   (3) Install water separator and duct section in fuselage support clamps. Position drain fitting at forward bottom position.

   (4) Secure support clamps around duct section and clamps around separator.

   (5) Connect overboard drain flex line to front end of separator.

   (6) Connect duct couplings at forward end of duct section.

   (7) Connect duct coupling at aft end of separator.

   (8) Install control cable fairlead to floor support at Fuselage Station 448.

   (9) Start and air load APU.

   (10) Check ducts and couplings for leakage and shut down APU.

   (11) Inspect area for foreign objects, security of all attachments.

   (12) Install floor.

2. **Water Separator Anti-Ice Control Box — Removal / Installation**

   CAUTION: BEFORE PERFORMING THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

   A. Removal

   (1) Remove FLR-21 to gain access to control box.

   (2) Disconnect electrical plug.

   (3) Remove screws securing box to structure; remove box.

   B. Installation

(1) Install control box using hardware previously removed.

(2) Install electrical plug.

(3) Disconnect electrical plug from water separator anti-ice sensor (under FLR-22).

NOTE: Remove FLR-22 only if required, as it may be possible to reach sensor through FLR-21 (sensor is in main duct just aft of water separator unit).

(4) Place a jumper from pin A to B of sensor plug.

(5) Energize main and essential dc buses.

(6) Water separator anti-ice valve should go to full open (valve may pulse).

(7) Pull H_2O SEP circuit breaker. (pilots circuit breaker panel).

(8) Remove jumper from sensor plug; do not connect plug to sensor.

(9) Push H_2O SEP circuit breaker; water separator anti-ice valve should move to full closed (valve may pulse.

(10) De-energize main and essential dc buses.

(11) Connect sensor plug to sensor.

(12) Inspect area for foreign objects, security of all attachments.

(13) Install FLR-21 and 22 (if removed) and/or other access opened.

3. Water Separator Anti-Ice Valve — Removal / Installation

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to water separator anti-ice valve as follows:

(a) Aircraft 1 - 80, 114 not having ASC 128A , valve is in tail compartment between APU closure and pressure dome, lower section.

(b) Aircraft 1 - 80, 114 having ASC 128A and Aircraft 81 - 200, 322 and 323, valve is in the baggage compartment (remove either FLR 24 or 25).

(2) Disconnect electrical plug.

(3) Disconnect duct couplings at each end of valve.

(4) Remove hardware securing valve to structure.

(5) Remove valve and O-rings.

B. Installation

(1) Inspect O-rings and replace as necessary.

(2) Install valve to structure with O-rings and secure with hardware previously removed.

(3) Connect duct couplings and tighten securely.

(4) Connect electrical plug.

(5) Gain access to water separator control box under FLR 21.

(6) Remove control box plug.

(7) Jumper control box plug pins C to A.

(8) Energize main and essential dc buses.

(9) Valve should move toward close stop.

NOTE: Valve will not pulse during this check.

(10) Pull H_2O SEP circuit breaker. (pilots circuit breaker panel).

(11) Move jumper at control box plug to pins B and C.

(12) Depress H_2O SEP circuit breaker; valve should move to open.

NOTE: Valve will not pulse.

(13) Pull H_2O SEP circuit breaker.

(14) Remove jumper and reconnect plug to water separator anti-ice control box.

(15) Depress H_2O SEP circuit breaker.

(16) Start and air load APU; check valve for leaks at couplings.

(17) Perform Water Separator Anti-ice Valve — Operational Test, this Section.

(18) Shut down APU air and APU if no longer required.

(19) De-energize main and essential dc buses.

4. Water Separator Anti-Ice System — Operational Test

A. Apply external dc power, and energize main dc bus and essential dc bus by placing BATT SW to NORMAL and EXT PWR SW to ON.

B. Close H_2O SEPARATOR circuit breaker.

C. Observe position of water separator anti-ice valve (Fuselage Station 585). Valve cam should be against CLOSED stop (temperature above 40°F).

NOTE: Valve is located in tail compartment between pressure dome and APU container. (Aircraft 1 - 80, 114 having ASC 128A and Aircraft 81 - 200, 322 and 323, the anti-icing valve is located under floorboards 24 and 25 forward of the pressure Bulkhead, Station 539.)

D. Disconnect electrical plug at water separator anti ice valve.

E. Ground pin C of valve receptacle.

F. Apply 28V dc to pin A of valve receptacle. Valve should open fully.

G. Remove lead from pin A and apply 28V dc to pin B. Valve should close fully.

H. Remove test lead and reconnect electrical plug to receptacle.

I. Remove electrical plug at water separator anti-ice sensor (Fuselage Station 480, left side under FLR 22).

J. Short pins A and B at sensor plug. Water separator anti-ice valve should open fully.

K. Reconnect sensor electrical plug. Water separator anti-ice valve should close fully.

L. Remove water separator anti-ice sensor from duct.

M. Insert probe of element in ice and water mixture, and observe that water separator anti ice valve moves toward open.

N. Install water separator anti-ice element in duct.

5. Water Separator Anti-Ice Sensor — Removal / Installation

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to sensor by removing FLR-22.

(2) Disconnect electrical connector from sensor.

(3) Remove screws securing sensor to duct boss and remove sensor.

B. Installation

(1) Connect electrical connector to sensor.

(2) Energize main and essential dc buses.

(3) Gain access to water separator anti-ice valve.

(a) Aircraft 1 - 80, 114 not having ASC 128A the anti-ice valve is located in tail compartment between APU enclosure and pressure dome lower station.

(b) Aircraft 1 - 80, 114 having ASC 128A and Aircraft 81-200, 322 and 323, anti-ice valve is in the baggage compartment (remove either FLR-24 or 25).

(4) With sensor in air temperature above 40°F, anti-ice valve should be closed (valve cam against closed stop).

(5) Immerse sensor probe in ice water. Anti-ice valve should open (valve cam against open stop).

NOTE: A pulsating operation of valve is normal.

(6) Remove sensor from ice water.

(7) De-energize main and essential buses.

(8) Install sensor in duct boss and secure with screws.

(9) Connect electrical connector.

(10) Inspect area for foreign objects, security of all attachments.

(11) Install floorboards previously removed.

6. Water Separator Bag — Cleaning

A. Disassemble water separator to gain access to internal components; remove V-band clamp securing plenum section to condenser. Remove O-rings and retain if serviceable.

B. Remove filter element from separator body. If inspection indicates cleaning is not required or if bag is damaged, replace bag and disregard Steps C. and D. below.

C. Brush and vacuum filter.

NOTE: Do not disassemble condenser assembly.

D. If required, wash filter element in mild detergent. Install bag when wet.

E. Remove screen covering, drain plenum.

F. Clean screen, drain plenum, and drain fitting.

G. Install plenum screen.

H. Check relief valve in aft end of separator condenser for freedom of movement.

I. Inspect large O-rings and replace if necessary.

J. Install O-rings in grooves of separator body.

K. Place cover on body ensuring O-rings are properly seated.

L. Install V-band clamp to secure water separator parts.

M. Install water separator, this Section.

EXHAUST SYSTEM / ELECTRONIC COOLING — DESCRIPTION / OPERATION

1. Exhaust System / Electronic Compartment Equipment Cooling

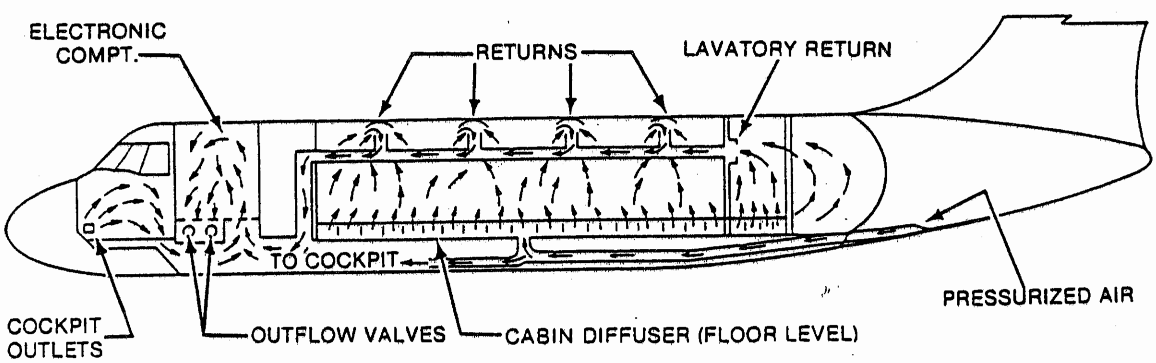
(See Figure 1)

The airflow path is from the cabin outlets, upward to a series of exhaust outlets. The exhaust air from these outlets is then ducted forward on each side of the cabin, and then down, under the floor to the electronics compartment. Cockpit airflow is from the outlets, through the console and pedestal openings, progressing under the cockpit floor to join cabin airflow to the electronics compartment. The electronics compartment is completely closed, and the exhausted air is forced to flow around the electronics gear to the only path of overboard exit the two outflow valves located in the forward lower part of the compartment, thus airflow for cooling purposes around the electronics equipment is maintained. The exhausted air is then allowed to pass overboard through the outflow valves to complete the cycle. It is to be noted that no air is recirculated in this aircraft.

CAUTION: SINCE THE ENTIRE RADIO COMPARTMENT VENTILATION DEPENDS ON EXHAUSTED AIR FROM THE OCCUPIED AREAS OF THE AIRCRAFT, IT IS ESSENTIAL THAT AIRFLOW OF SOME NATURE (BLOWER, APU, OR RAM AIR) BE SUPPLIED AT ALL TIMES WHEN ELECTRONIC GEAR IS USED. IF THIS IS DISREGARDED, RADIO AND ELECTRONIC EQUIPMENT IN THIS COMPARTMENT WILL NOT RECEIVE PROPER COOLING AND UNDER LONG PERIODS OF NO AIRFLOW, WILL OVERHEAT.

2. Radio Rack Overheat Warning

Radio rack overheat thermal switches, placed adjacent to the pressurization outflow valves, sense excessive heat in the compartment and indicate through the RADIOS HOT light on the master warning system. They are set to 200° ± 10°F. (See Chapter 26, FIRE DETECTION)



Electronic Equipment Cooling
Figure 1.

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EMERGENCY DC OPERATION — DESCRIPTION / OPERATION

1. **Emergency DC Operation — Effect on Air Conditioning Systems**

- Manual and automatic temperature control is inoperative in emergency.
- Ram air ventilation via AIR COND-VENT switch is fully operative in emergency.
- APU is fully operative in emergency.
- Blower ON-OFF switch is operative in emergency.
- Water separator anti-ice system is inoperative in emergency.
- Blower dump via the nutcracker system is inoperative in emergency, except in aircraft having ASC 108A Part 1. It is then fully operative in emergency. (Production effectivity Aircraft 94 - 200, 322 and 323)

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CABIN PRESSURIZATION CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

The pressurization control system operates on the outflow side of the airflow path to control the level of compartment pressurization. The Gulfstream I pressurization system is classified as a variable isobaric, fixed differential, with selective rate of change. (see Figure 1)

The pressurization control devices, together with their associated components control compartment pressurization by allowing air to flow at a controlled rate from the occupied areas to atmosphere through the outflow valve. Cabin altitude and ambient altitude are sensed and through automatic electronic devices, establish control over the outflow valve, in accordance with predetermined programming by the crew. The pressurization control system functions in the following ways:

A. Isobaric pressurized operation with cabin altitude as selected by the crew and performed by the cabin pressure controller.

B. Maximum pressure differential operation where the difference between the cabin and ambient pressures remains constant at 6.55 psi. In this range the cabin pressure changes in direct proportion to the ambient pressure with the cabin being maintained at a higher pressure than ambient. The maximum pressure differential is determined by the structural strength of the cabin.

C. Manual crew selection of the rate of cabin pressure change, as desired, within differential limits.

D. Manual reduction of cabin pressure in the event of automatic controller malfunction.

E. Rapid cabin pressure decrease for smoke evacuation.

F. Automatic depressurizing of the aircraft during ground operation.

The following equipment is included in the pressurization control system:

Unit	No Per A/C	Location
Outflow Valves	2	Aft of the copilots bulkhead from Fuselage Station 133 to approximately 157.
Automatic Cabin Pressure Controller	1	Copilots console
Pneumatic Relays	2	Adjacent to the outflow valves.
Manual Cabin Pressure Control Valve	1	On the copilots console, aft of the automatic cabin pressure controller.
Manual Vacuum Control Valve (Aircraft 1 - 200, 322 and 323, having ASC 203).	1	Fuselage Station 145
Pressurization Dump Solenoid Valves	2	Aft of the copilots bulkhead from Fuselage Station 133 to approximately 157.
Pressure Control Venturi	1	Fuselage Station 140.
Ground Test Valves	2	Under the copilots console.
Cabin Pressure Warning Aneroid Switch	1	Above the hydraulic reservoir in the hydraulic compartment
Cabin Altitude and Differential Pressure Gage	1	Upper overhead panel, adjacent to the air conditioning control switch.
Cabin Rate of Climb Indicator	1	Upper overhead panel, under the cabin altitude and differential pressure gage.

2. Terminology

The pressurization control system, basically, functions to relate two types of air pressures; cabin pressure, and ambient pressure. These terms are defined as follows:

CABIN PRESSURE. The actual pressure inside the pressurized area of the aircraft.

AMBIENT PRESSURE. The pressure of the air surrounding the aircraft at the particular altitude at which it is flying.

CABIN ALTITUDE. The pressure altitude within the cabin which is entirely dependent up on cabin pressure.

AIRCRAFT OR AMBIENT ALTITUDE. The pressure altitude of the aircraft, which is entirely dependent on ambient pressure.

The following terms deal with the various differentials in pressurization:

PRESSURE DIFFERENTIAL. The difference between the cabin and ambient air pressures, where in the cabin pressure is higher. This is a maximum 6.55 $\pm$ 0.2 psi.

SAFETY PRESSURE RELIEF. A safety limit pressure above the normal maximum pressure differential at which a system will automatically relieve excessive pressure within the cabin, if the normal pressure regulating devices fail to operate. This is 7.2 $\pm$ 0.2 psi.

VACUUM DIFFERENTIAL. The difference between cabin and ambient air pressures wherein the ambient pressure is higher. The maximum vacuum differential in is 0.2 psi.

The next series of terms refers to the operating ranges of a pressurization system.

UNPRESSURIZED RANGE. An operating range of a system at which the cabin pressure is maintained at approximately the same as the ambient; in other words, there is no pressurization.

ISOBARIC RANGE. An even or constant pressure range, derived from the prefix "ISO" meaning even or constant, and "BARIC" meaning pertaining to barometric pressure. In this range, the cabin pressure remains constant, regardless off ambient pressure.

DIFFERENTIAL RANGE. A range of operation in which the difference between the cabin and ambient pressures remains constant. In this range, the cabin pressure changes in direct proportion to the ambient pressure; the cabin, however, is maintained at a higher pressure than the ambient. The differential range is determined by the structural strength of the cabin.

PRESSURE RATE OF CHANGE. The rate of pressure change, within the cabin, per unit of time (usually feet per minute).

3. Detailed System Operation

A. General.

Under normal conditions the automatic cabin pressure controller, together with the pneumatic relays and the outflow valves, controls pressurization by allowing air to flow at a controlled rate from the cabin to the atmosphere through the outflow valve. The controller utilizes cabin, vacuum, and atmospheric air pressure to establish control pressure for the operation of the outflow valves. Filtered cabin air enters the head chamber through the orifice in the controller. The head chamber air is released to vacuum or atmosphere by the isobaric or differential control metering valve, establishing control pressure for application to the associated pneumatic relays. The isobaric and normal maximum (6.55 psi) differential pressure control system, together with the selected pressure rate of change control, establishes the cabin air pressurization schedule for the aircraft. As the aircraft ascends from takeoff, the rate of change control limits the rate of cabin pressurization until the cabin isobaric pressure is reached. The isobaric control system maintains the selected isobaric cabin pressure until the maximum normal cabin-to-atmosphere differential pressure is reached (6.55 psi). The small window at the bottom of the automatic cabin pressure controller dial indicates the aircraft altitude at which this will occur. As the aircraft continues to ascent the normal differential control system limits the cabin-to-atmosphere differential pressure to 6.55 $\pm$ 0.2 psi. A small pneumatic signal pressure, (the primary control signal pressure) comes from the head chamber of the automatic controller and goes to the right side of each pneumatic relay. This primary

control signal pressure (as produced by the controller) changes when more or less pressurization is needed.

On the left hand side of the pneumatic relays is a chamber vented through a needle valve to a 6 inHg. vacuum source line from the deicing system vacuum manifold. This left side of each relay is ported directly to the top chamber (control chamber) of each outflow valve. The control chamber of the outflow valves is also vented, through a small orifice, and a filter to cabin pressure. The control chamber pressure reacts upon a large diaphragm assembly, known as the control diaphragm. Also acting upon the control diaphragm is a calibration spring tending to hold the control diaphragm down, and cabin pressure tending to push it up. The control diaphragm governs the opening of the valve between the cabin air and the ambient.

In order to open the outflow valves, cabin pressure must govern the compression of the diaphragm spring and control chamber pressures tending to hold the valves closed. By applying vacuum inside of the control chamber in a sufficient amount, a difference in pressures across the control diaphragm compresses the spring and moves the diaphragm assembly upward, opening the valve. Since the amount of vacuum is governed by the primary control signal pressure from the controller matched by the pneumatic relays which will produce high flow a small change in primary signal pressure causes a rapid reaction vacuum in the control chamber of the outflow valves.

The small orifice in the vent line to the cabin filter from the control chamber restricts the cabin air from entering to replace the air as fast as it is being drawn out of the chamber through the pneumatic relays by the vacuum. Therefore, the differential is created, so that any time the pressure inside of the control chamber drops, the outflow valves open, proportionally causing a proportional amount of cabin air to exhaust overboard.

B. Unpressurized Operation.

Prior to control operation, and when the isobaric selection is above aircraft altitude, pressure on both sides of the rate control diaphragm is equal. The isobaric bellows is compressed and the isobaric metering valve is open. Filtered cabin air enters the head chamber through an orifice and flows toward vacuum through the open isobaric metering valve. Head chamber pressure is less than the atmosphere pressure providing the decrease in the head chamber pressure does not exceed the selected rate of change. The resulting head chamber pressure is transmitted to a pneumatic relay which controls the outflow valves according to the control schedule of the pressurization control unit.

C. Selected Rate of Pressure Change Operation.

As the aircraft starts to climb, the pressure in the head chamber is reduced, causing a decrease in reference chamber pressure. The rate of decrease of pressure in the reference chamber is limited to a steady schedule determined by the position of the rate metering valve which is set by the RATE control knob. Pressure change in the reference chamber lags that of the head chamber, resulting in a differential pressure across the rate control diaphragm. In response to this difference in pressure, the diaphragm moves the isobaric metering valve to a metering position to limit the rate at which the pressure in the head chamber decreases.

D. Isobaric Range.

As the pressure in the head chamber continues to decrease and the selected isobaric altitude (the cabin cruising altitude) is approached the isobaric bellows expands sufficiently to override the rate control diaphragm and cause the isobaric metering valve to start closing. With the head chamber pressure maintained at essentially a constant value, variations in pressure applied to the outflow valves causes the valves to operate to maintain cabin pressure at a selected value.

E. Normal Maximum Differential Pressure Operation (6.55 $\pm$ 0.2 psi).

The normal differential control serves to prevent cabin pressure from exceeding aircraft structural design limits. As the aircraft continues to climb to an altitude where the differential pressure between the cabin and the atmosphere approaches the calibrated differential setting of the control (6.55 $\pm$ 0.2 psi) the differential pressure between the head chamber and the atmosphere is sufficient to force the differential diaphragm to move up, opening the differential metering valve and releasing head chamber air to static, which is actually a 6.55 psi vacuum in relation to cabin pressure. As the aircraft climb continues and head chamber pressure continues to decrease, cabin pressure acts to open the outflow valves. The

outflow valves operate to maintain the calibrated differential pressure between the cabin and atmosphere as directed by the controller at 6.55 $\pm$ 0.2 psi.

As the aircraft descends and atmospheric pressure increases, the differential metering valve closes and pressure within the head chamber increases. Head chamber air bleeds into the reference pressure chamber at the rate determined by the rate control knob setting. If this increase (this increase refers to head chamber pressure, not reference chamber) exceeds the rate control setting, the rate control diaphragm opens the isobaric metering valve to reduce the rate of cabin pressure increase. As the descent continues into the isobaric range, the increase in pressure in the reference chamber causes the isobaric bellows to become effective, and the action is substantially the reverse of that which occurred during the ascent.

4. Pressurization Electrical Power Supply Sources

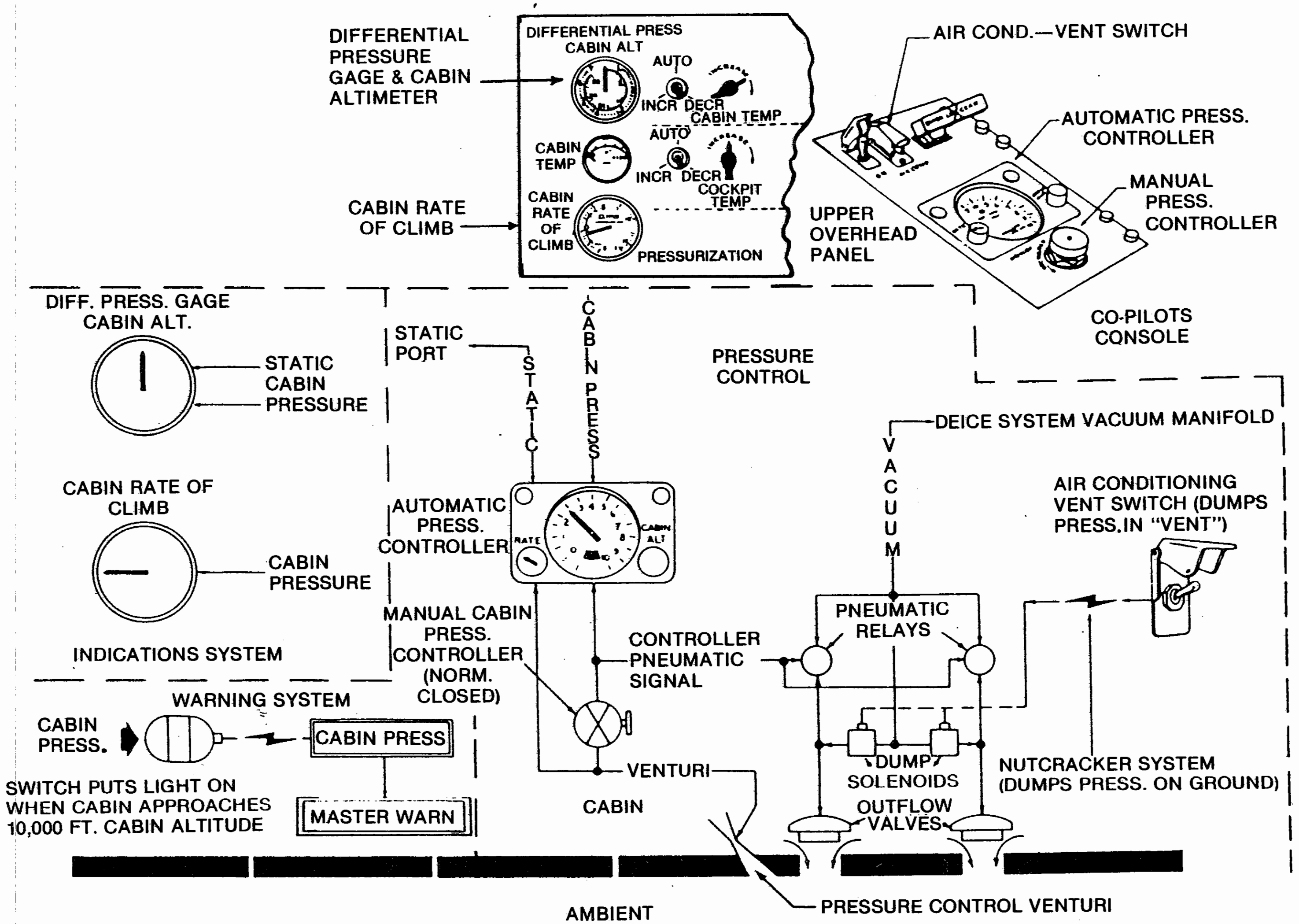
A. The CABIN PRESS and RAM AIR circuit breakers receive their feeds from the essential dc bus, therefore they are operative during emergency dc operations of the aircraft. This provides control over the pressurization functions involved with the BLOWER ON-OFF switch and the AIR COND-VENT SW. of the aircraft, even during emergency dc system configurations.

B. Aircraft 1 - 93 and 114 having ASC 108A Part 1 and Aircraft 94 - 200, 322 and 323, have the nutcracker system fed from the essential dc bus, therefore, the pressurization dump functions of the nutcracker are fully operative in emergency dc operation.

C. The APU unit receives its starting and operating power from the essential dc bus, and its generator feeds the essential dc bus when it is placed on the line. This unit can be started, controlled, and run while in emergency dc operation. The supply of pressurization air and/or APU generator power to assist in maintaining the aircraft during emergency dc operation is therefore available.

5. Ground Test Valves

Two valves are located under the copilots console outboard of the map case and are used only to test the system on the ground.



Pressure Control System — Functional Diagram
Figure 1.

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Time (Minutes)	Cabin Alt (ft)	Aircraft Alt (ft)	Ambient Press. (psi)	Cabin Press. (psi)	Cabin Differential (psi)
0	5500	25,000	5.4	12.0	6.6
1	4900	22,500	6.1	12.3	6.2
2	4300	20,000	6.7	12.6	5.9
3	3700	17,500	7.5	12.9	5.4
4	3100	15,000	8.3	13.2	4.9
5	2500	12,500	9.2	13.4	4.2
6	1900	10,000	10.1	13.7	3.6
7	1300	7,500	11.2	14.0	2.8
8	700	5,000	12.3	14.3	2.0
9	100	2,500	13.4	14.6	1.2
10	0	0	14.7	14.7	0.0

4. **Outflow Valves**

Two outflow valves, installed aft of the copilot's bulkhead from Fuselage Station 133 to approximately Fuselage Station 157, are mounted on the right skin of the aircraft under the radio rack. The valves exhaust overboard. Both valves are identical and operate in parallel for safety reasons. They are pneumatically-operated diaphragm devices which are controlled by a penumatic signal from the automatic controller, manual controller, or a dump signal from the pressurization dump solenoid valves. The outflow valves will allow the cabin air to exhaust from the pressurized area, or will close down, restricting outflow air to maintain pressurization and allow outflow air to exhaust at specific rates.

5. **Automatic Pressurization Controller**

The automatic pressurization controller unit is located on the copilot's console. By manipulation of knobs located on the unit, a selection can be made of the isobaric and rate of change pressures at which the system will operate during a particular flight.

The controller consists of a dual dial calibrated in thousands of feet, two knobs, and internal mechanisms. The controller contains three separate pressure-controlling systems: isobaric, selected pressure-rate-of-change, and normal maximum pressure differential (6.55 psi). This controller is the sensing or controlling device for the above three functions of the outflow valves. The controller senses or sets up the situation; the outflow valves actually do the work. Three input pressures are fed into the controller.

A. Static pressure, from flush static ports on the forward, right and left sides of the aircraft, near the nose.

B. Cabin pressure, through a filter beneath the controller vented to the pressurized area.

C. Vacuum from the pressure control venturi (which produces vacuum for the use of the controller). This vacuum is utilized by the controller as a portion of the output pneumatic signal, which in turn, through the pneumatic relays is used to control the outflow valves.

Manual controls for the unit are located on each side of the dial. These controls consist of the CABIN ALT. selector knob and the RATE control knob. The CABIN ALT. selector knob, in conjunction with the dial, the dial gear, and the pointer, provides a means of manually selecting the cabin altitude. The dial is calibrated in thousands of feet from −1000 to + 10,000 feet cabin altitude, and is used in selection of the isobaric cabin altitude. The dial gear is calibrated to indicate the maximum aircraft altitude at which the selected isobaric range of operation can be maintained. The maximum aircraft altitude indication is visible through the window in the dial. For example, if an isobaric cabin altitude of 3000 feet is to be programmed, the CABIN ALT. selector knob is turned until the pointer on the main dial of the controller reads 3000 feet. At the same time, through the gear train, the indication in the small window beneath the dial reads approximately 20,500 feet. This means that the isobaric range of the unit can be maintained at a 3000 foot cabin altitude until an aircraft altitude of 20,500 feet is reached, at which time the 6.55 psi pressure differential takes over. This is merely a reference; it is not an altitude indication.

Programming of this controller for a particular flight involves turning the CABIN ALT. knob until the maximum flight plan altitude appears in the cut in the dial. The hand then indicates the lowest cabin altitude (isobaric) which can be maintained, providing that the aircraft altitude does not exceed the cut in the dial, and that aircraft altitude is above the reading on the handle.

If the controller is not re-programmed on letdown, the isobaric range holds the cabin altitude until the aircraft passes through ambient altitude with the same value as the programmed altitude. From then on the aircraft is unpressurized, or goes into vacuum relief.

Re-programming to the landing field altitude is necessary to prevent running into the unpressurized range or vacuum differential on letdown. When the final adjustment of the controller is made (on the "in range check" list) it is normally set so that the hand reads landing field pressure altitude minus 100 feet This enables the aircraft to land slightly pressurized for maximum comfort, and avoids the unpressurized range.

The RATE control knob serves to select the rate of cabin pressure increase or decrease within limits established by the cabin pressure differential setting. It indicates only minimum (ccw) and maximum (cw), with a small index arrow dead ahead. Most controllers with the knob arrow dead ahead (aligned with the index arrow on the case) produce a setting of about 500 ft/min rate of change. Turning the knob full CCW produces about a 50 ft/min rate. Full CW rotation will produce about a 2000 ft/min rate of change. This knob is used during changes of pressure in conjunction with the cabin rate of climb indicator located in the upper overhead panel, right side. This instrument is merely a standard, lagged-type rate of climb indicator vented into the cabin, and measures rate of climb or dive of the cabin pressure in thousands of feet per minute. The knob on the controller is adjusted and the rate of climb indicator tells the actual rate of change of cabin altitude.

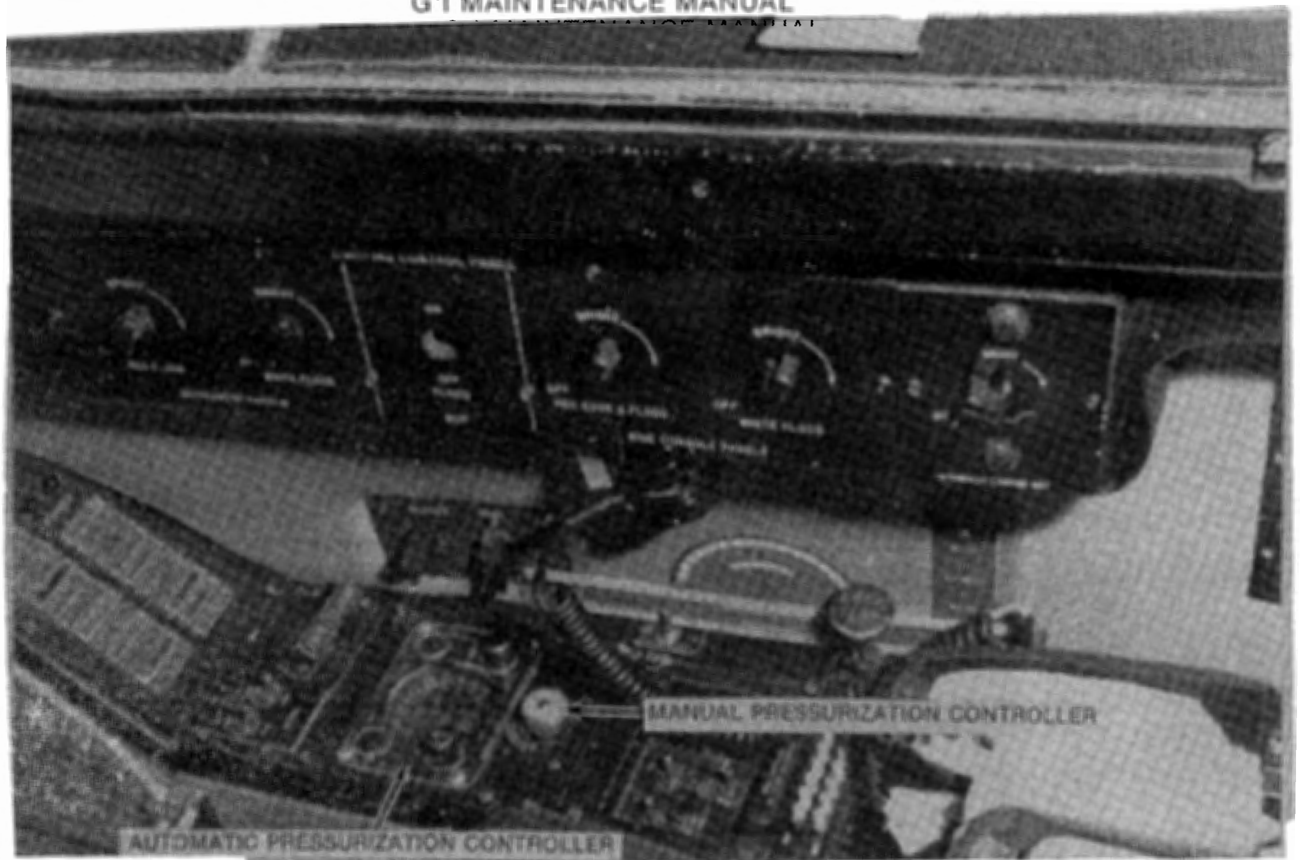
6. **Pneumatic Relays**

Each outflow valve is controlled by a pneumatic signal from the controller. This signal is matched by two pneumatic relays located adjacent to the outflow valves which by using the pneumatic deicing system vacuum can produce a high flow from and thus rapid response of the outflow valves. One relay serves each outflow valve but both receive the same signal from the controller.

7. **Manual Cabin Pressure Controller**

CAUTION: KEEP THE MANUAL CONTROLLER IN THE FULL CLOSED POSITION (FULL CW) AT ALL TIMES UNLESS THE AUTOMATIC CONTROLLER IS TO BE BYPASSED TO DECREASE PRESSURIZATION.

The manual cabin pressure controller is a small needle valve installed in the copilot's console aft of the automatic controller. (See figure 2.) It bypasses the pressure control venturi vacuum around the automatic controller to the pneumatic relays for the manual control of the system whenever the automatic controller fails. When fully open, the needle valve depressurizes the aircraft. This control can only decrease the cabin pressure being maintained by the automatic cabin pressure controller.



Co-Pilot Console Pressure Control System Controls

Figure 2

8. **Pressurization Dump Solenoid Valves**

The pressurization dump solenoid valves, located near the outflow valves and pneumatic relays, are electrically operated, normally closed, ball type solenoid valves. When energized, they depressurize the cabin by routing the deicing system vacuum manifold directly to the outflow valves opening them fully. All pressurization is dumped immediately. (This can be extremely uncomfortable, and in some cases dangerous if ambient altitude is high, and oxygen masks are not used.)

Placing the AIR COND—VENT switch on the copilot's console in the VENT position, or by weight on both main gears through the nutcracker system will energize the pressurization dump solenoids, thereby, causing depressurization of the cabin.

9. **Ground Test Valves — General**

Two valves are located under the copilot's console outboard of the map case and are used only to test the system on the ground.

10. **Cabin Pressure Warning System**

An aneroid type pressure switch is located at the top of the hydraulic reservoir compartment. This switch is normally open if the cabin altitude is below 9250 ± 750 feet. If the pressurization fails to maintain the cabin below this altitude, the switch is actuated and a CABIN PRESS light is illuminated on the master warning system, indicating a possible pressurization failure. The switch is set so that it will open on increasing pressure before 8000 feet cabin altitude is reached.

11. **Cabin Altitude and Differential Pressure Gage**

A dual indicator is installed in the upper overhead panel, directly adjacent to the air conditioning control switches. (See Figure 3.) One needle indicates cabin altitude and the other indicates cabin to ambient pressure differential as a reference for the effectiveness of the pressurization sytem.

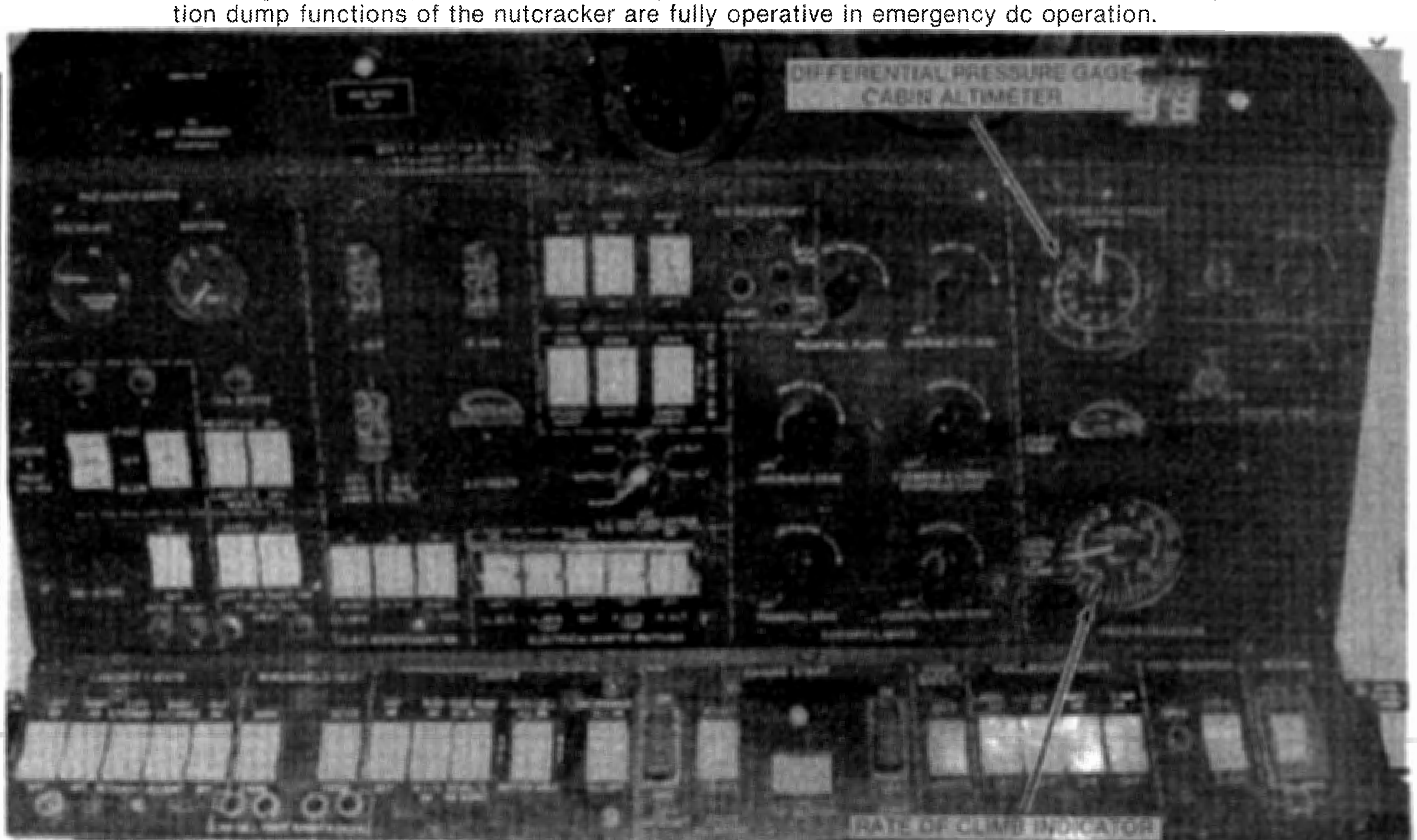
12. **Cabin Rate of Climb Indicator**

Also installed in the upper overhead panel under the differential pressure gage cabin altimeter is a standard lagged type rate of climb indicator vented to the cabin. This instrument gives a visual indication in thousands of feet of the rate of change in cabin pressurization within the aircraft.

13. **Pressurization Electrical Power Supply Sources**

A. The CABIN PRESS and RAM AIR circuit breakers receive their feeds from the essential dc bus, therefore they are operative during emergency dc operations of the aircraft. This provides control over the pressurization functions involved with the BLOWER ON-OFF switch and the AIR COND-VENT SW. of the aircraft, even during emergency dc system configurations.

B. Aircraft 1 through 93 including 114 having ASC 108A Part I incorporated and Aircraft 94 through 200 including 322 and 323, have the nutcracker system fed from the essential dc bus, therefore, the pressurization dump functions of the nutcracker are fully operative in emergency dc operation.



Upper Overhead Panel Pressure Control System Controls
Figure 3

C. The APU unit receives its starting and operating power from the essential dc bus, and its generator feeds the essential dc bus when it is placed on the line. This unit can be started, controlled, and run while in emergency dc operation. The supply of pressurization air and/or APU generator power to assist in maintaining the aircraft during emergency dc operation is therefore available.

14. Outflow Valves — Detailed

The outflow valve have the function of creating an outflow air path, safety pressure relief, and vacuum differential control of the pressurization system. The base of the outflow valve, which is a magnesium casting, contains openings through which air flows between the cabin and the atmosphere. The flow of air is controlled by a large flow valve which seats against the base. This valve is secured to a coverplate which slides on a teflon shaft when the valve moves. Air from the cabin can flow between the valve and the coverplate. The control diaphragm functions, in effect, as two separate diaphragms.

The portion of the diaphragm between the valve and the base operates on a differential pressure between the cabin and the control chamber. The other portion of the diaphragm between the base and coverplate operates on a differential pressure between the cabin and the atmosphere. This control diaphragm, and a small pressure relief diaphragm cover and cover assembly, form two chambers; a cabin pressure chamber and a control chamber. The pressure relief diaphragm separates the cabin pressure chamber and a recess in the base, thus forming a third chamber, the atmosphere sensing chamber.

The control chamber of the valve contains a spring which loads the valve towards the closed position. This is the large spring directly beneath the port from the pneumatic relay. The control chamber is vented to the cabin through a filter. The cabin pressure chamber contains a pilot shaft. This shaft guides the cover which is attached to the flow valve by means of the teflon shaft and contains an air passage for venting the control chamber to the atmoshpere through the atmosphere sensing chamber. This passage which is used in conjuction with the safety pressure relief mechanism, is operated by a ball type pressure relief valve. This relief valve is connected to the safety pressure relief diaphragm. The atmosphere sensing chamber contains a spring which loads the safety pressure relief diaphragm holding the ball valve normally closed. The spring is the very small one in the bottom center of the valve. In referring to the diagram (Figure 5), it will be noted that there are actually three connections to the outflow valve: one from the pneumatic relay into the control chamber; one from the filter, which allows cabin air to enter through the small orifice into the control chamber; and a third in the bottom, which allows static air pressure to enter the atmosphere sensing chamber. The outflow valve is mounted on the pressurized side of the cabin. The cabin air flows through the orifice in the filtered cabin air inlet port to the control chamber. This flow results in a pressure in the control chamber; this is cabin pressure minus whatever pressure loss occurs across the orifice. Cabin pressure exists in the cabin air pressure chamber through the small holes in the circumference of the flow valve. Atmosphere pressure exists in the atmosphere sensing chamber through the static port at the bottom.

In normal operation, vacuum from the pneumatic relays, controlled by the automatic or manual controller signal pressure, is ported into the control chamber. This creates a low pressure area in the outflow valve control chamber which is directly proportional to the controller demands. The cabin air pressure in the cabin air pressure chamber on the other side of the control diaphragm thus is greater than the reduced pressure in the control chamber causing the diaphragm to move the flow valve toward open. The amount will depend on the amount of vacuum applied to the control chamber. The more vacuum, the greater the opening. The less the restriction, the less pressurization is attained.

Safety pressure relief occurs when the pressure differential between the cabin pressure and the atmospheric pressure is sufficient to cause the safety pressure relief diaphragm to move against the force of its backing spring. Movement of the diaphragm unseats the ball valve. When this occurs, air escapes from the control chamber to the atmosphere, reducing the pressure in the control chamber. Cabin pressure exerted against control diaphragm from within the cabin pressure chamber forces the control diaphragm to open. This allows cabin air to flow to the ambient air, maintaining the cabin differential pressure at a safety limit of 7.2 ± 0.2 psi in case the normal maximum differential pressure control (6.55 psi) in the controller fails.

Vacuum relief occurs whenever ambient pressure exceeds cabin pressure by 0.20 psi. When this condition exists, pressure on the atmosphere side of the vacuum relief section of the control diaphragm forces the diaphragm against the coverplate and lifts the flow valve off of its seat. This allows air at atmospheric pressure to flow into the cabin.

Dump operation occurs whenever the cabin pressurization dump solenoid valves are energized electrically. This permits vacuum to decrease pressure in the outflow valve control chamber, permitting cabin pressure to push the valve open.

The solenoid dump valves can be energized to dump the system by either of the following means:

A. Weight on the main landing gear through the nutcracker system.

B. Placing the AIR COND — VENT switch on the copilot's console in VENT position. (This also accomplishes other functions in the air conditioning system.)

In-flight controlled depressurization can be accomplished by opening the manual cabin pressure control valve on the copilot's console. This permits venturi vacuum to decrease pressure in the control chamber of the controller and in the signal pressure line to the pneumatic relays. The pneumatic relays are displaced, compressing the springs and opening the needle valve on the output side of the relays. As a result deicing system vacuum decreases pressure in the outflow valve control chamber and cabin pressure opens the outflow valves.

CAUTION: IT IS TO BE NOTED THAT THE OUTFLOW VALVE PORTS ARE NOT MARKED ON THE UNIT, THEREFORE THE FOLLOWING MUST BE OBSERVED.

THE CENTER PORT OF THE TWO ON THE DOME (CABIN) SIDE OF THE VALVE ALWAYS GOES TO IT'S PNEUMATIC RELAY AND PRESSURIZATION DUMP SOLENOID.

THE PORT OFF CENTER ON THE DOME (CABIN) SIDE OF THE RELAY ALWAYS GOES TO ITS FILTER.

THE SINGLE PORT ON THE AMBIENT (OUTFLOW) SIDE OF THE VALVE MUST BE OPEN TO ALLOW STATIC AIR TO ENTER. IF CAPPED, MAKE SURE THE CAP IS REMOVED BEFORE INSTALLING VALVE IN THE AIRCRAFT. NO TUBING CONNECTION IS MADE. PORT IS LEFT OPEN.

15. **Automatic Pressurization Controller — Description**

The automatic controller has the functions of isobaric, rate of change, and normal maximum differential control of the pressurization system. The isobaric system consists of a bellows, a rocker arm supported by a fulcrum and a yoke, a compression spring loaded by the cabin altitude selector knob. a cabin altitude pointer, and gearing connection this pointer to the cabin altitude selector knob. (See Figure 4.) The bellows is of the evacuated and sealed aneroid type and is connected to the rocker arm. It is called the isobaric bellows. The CABIN ALT. selector knob travelling nut and spring permits selection of the isobaric cabin altitude. The bellows and rocker arm are connected to the rate control diaphragm by means of a leaf spring. The rate control diaphragm (this is the large diaphragm separating the two major chambers of the controller) separates the reference chamber from the head chamber, and actuates a spring loaded isobaric metering valve which is ported to vacuum from the pressure control venturi.

The selected pressure-rate-of-change control consists of a filter, a drilled passage with a capillary tube, a rate metering valve, and a valve seat. Adjustment of the rate metering valve is accomplished by means of the rate controller knob located on the front of the unit.

The normal maximum differential system (6.55 ± 2 psi) consists of a differential diaphragm, a metering valve and its seat, a diaphragm backing spring, and an adjustment screw. The force of the diaphragm spring at the metering valve opening position is adjusted when the unit is calibrated, and this adjustment remains fixed. The spring loaded side of the differential diaphragm is vented to atmosphere. The opposite side is exposed to head chamber pressure.

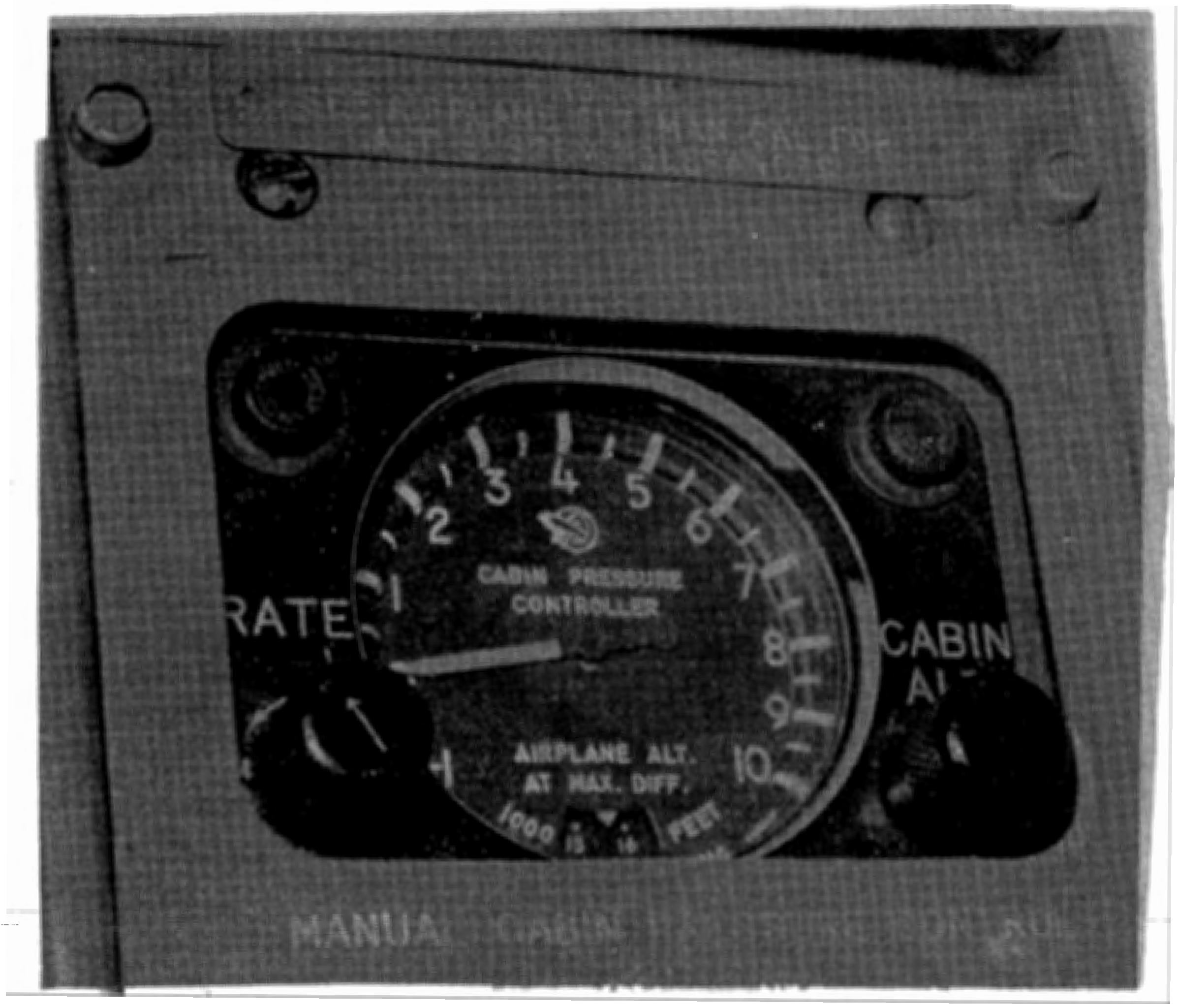
The small drilled passage in the controller has an opening into the reference pressure chamber and airflow between the two chambers is controlled by the rate control knob, which actually is a needle valve. Rapid changes in cabin pressure go directly through the filter into the head chamber, which is on one side of the rate control diaphragm. A small portion of the cabin air passes through the small drilled passage and the rate metering valve, into the reference chamber. Therefore, the pressure in the head control chamber changes almost immediately. Conversely, the rate of change in pressure in the reference chamber is directly proportional to the amount the rate control knob metering pin is opened. The magnitude of the pressure differential thus produced across the rate control diaphragm positions the isorbaric metering valve, thereby controlling the rate of pressure change in the head control chamber and therefore the outflow valves. However, the selected rate of pressure change cannot override the maximum normal differential control and vacuum relief control so that the pressure differential of 6.55 psi and the vacuum differential of 0.2 psi are not exceeded.

All controller system connections are located on underside of the unit. The cabin pressure controller is designed to be mounted on a panel in the pressurized area. The controller must be connected to a true atmospheric pressure (static), a source of vacuum (pressure control venturi), cabin pressure (through filter) and to pneumatic relays.

NOTE: This unit is not connected directly to the outflow valves. It must go through the pneumatic relays.

Tubing connections to automatic controller is as follows: (ports are marked)

Controller Port Marking	Connect To
Filter	Filter
ATMOS #1	Static Line
ATMOS #3	Pressure Control Venturi Vacuum
Outflow	Pneumatic Relays



Automatic Controller
Figure 4

16. **Manual Pressurization Controller**

The manual pressurization controller enables the crew to depressurize the aircraft by controlled manual means without resorting to an absolute, instantaneous dump, which would be the case if the pressurization dump solenoid were used. The valve consists of a housing within an inlet connection and an outlet connection, an adjustment knob, and a valve shaft. The valve shaft is turned by the knob to seat the needle valve. The needle valve is spring loaded inside the bottom end of the shaft. As the valve is opened, vacuum from the pressure control venturi is applied at the inlet connection side of the valve housing and flows through the valve seat opening and out the opposite connection. The valve is operated by turning the knob on the top of the valve counterclockwise to open the valve. To open the valve fully from the closed position requires approximately seven complete turns. A pointer on the top of the knob gives an indication of internal valve position. The pointer is 90 to 105 degrees counterclockwise from the inlet connection when the air begins to flow through the unit. Clockwise rotation of the valve shaft after the valve is seated merely compresses the spring in the end of the valve shaft, therefore it is considered a non-jamming type valve. When opened, vacuum is ported through the valve to the primary control pressure line to the pneumatic relays, thus manually controlling the outflow valves. It bypasses the automatic controller, and since it is always kept in the closed position, unless used for depressurizing, it can only reduce differential.

Completely open, the pressurization is fully dumped.

17. **Detailed System Operation**

A. General.

Under normal conditions the automatic cabin pressure controller, together with the pneumatic relays and the outflow valves, controls pressurization by allowing air to flow at a controlled rate from the cabin to the atmosphere through the outflow valve. (See Figure 5.) The controller utilizes cabin, vacuum, and atmospheric air pressure to establish control pressure for the operation of the outflow valves. Filtered cabin air enters the head chamber through the orifice in the controller. The head chamber air is released to vacuum or atmosphere by the isobaric or differential control metering valve, establishing control pressure for application to the associated pneumatic relays. The isobaric and normal maximum (6.55 psi) differential pressure control system, together with the selected pressure rate of change control, establishes the cabin air pressurization schedule for the aircraft. As the aircraft ascends from takeoff, the rate of change control limits the rate of cabin pressurization until the cabin isobaric pressure is reached. The isobaric control system maintains the selected isobaric cabin pressure until the maximum normal cabin-to-atmosphere differential pressure is reached (6.55 psi). The small window at the bottom of the automatic cabin pressure controller dial indicates the aircraft altitude at which this will occur. As the aircraft continues to ascent the normal differential control system limits the cabin-to-atmosphere differential pressure to 6.55 ± 0.2 psi. A small pneumatic signal pressure, (the primary control signal pressure) comes from the head chamber of the automatic controller and goes to the right side of each pneumatic relay. This primary control signal pressure (as produced by the controller) changes when more or less pressurization is needed.

On the left hand side of the pneumatic relays is a chamber vented through a needle valve to a 6 inHg. vacuum source line from the deicing system vacuum manifold. This left side of each relay is ported directly to the top chamber (control chamber) of each outflow valve. The control chamber of the outflow valves is also vented, through a small orifice, and a filter to cabin pressure. The control chamber pressure reacts upon a large diaphragm assembly, known as the control diaphragm. Also acting upon the control diaphragm is a calibration spring tending to hold the control diaphragm down, and cabin pressure tending to push it up. The control diaphragm governs the opening of the valve between the cabin air and the ambient.

In order to open the outflow valves, cabin pressure must govern the compression of the diaphragm spring and control chamber pressures tending to hold the valves closed. By applying vacuum inside of the control chamber in a sufficient amount, a difference in pressures across the control diaphragm compresses the spring and moves the diaphragm assembly upward, opening the valve. Since the amount of vacuum is governed by the primary control signal pressure from the controller matched by the pneumatic relays which will produce high flow a small change in primary signal pressure causes a rapid reaction vacuum in the control chamber of the outflow valves.

The small orifice in the vent line to the cabin filter from the control chamber restricts the cabin air from entering to replace the air as fast as it is being drawn out of the chamber through the pneumatic relays by the vacuum. Therefore, the differential is created, so that any time the pressure inside of the control chamber drops, the outflow valves open, proportionally causing a proportional amount of cabin air to exhaust overboard.

B. Unpressurized Operation.

Prior to control operation, and when the isobaric selection is above aircraft altitude, pressure on both sides of the rate control diaphragm is equal. The isobaric bellows is compressed and the isobaric metering valve is open. Filtered cabin air enters the head chamber through an orifice and flows toward vacuum through the open isobaric metering valve. Head chamber pressure is less than the atmosphere pressure providing the decrease in the head chamber pressure does not exceed the selected rate of change. The resulting head chamber pressure is transmitted to a pneumatic relay which controls the outflow valves according to the control schedule of the pressurization control unit.

C. Selected Rate of Pressure Change Operation.

As the aircraft starts to climb, the pressure in the head chamber is reduced, causing a decrease in reference chamber pressure. The rate of decrease of pressure in the reference chamber is limited to a steady schedule determined by the position of the rate metering valve which is set by the RATE control knob. Pressure change in the reference chamber lags that of the head chamber, resulting in a differential pressure across the rate control diaphragm. In response to this difference in pressure, the diaphragm moves the isobaric metering valve to a metering position to limit the rate at which the pressure in the head chamber decreases.

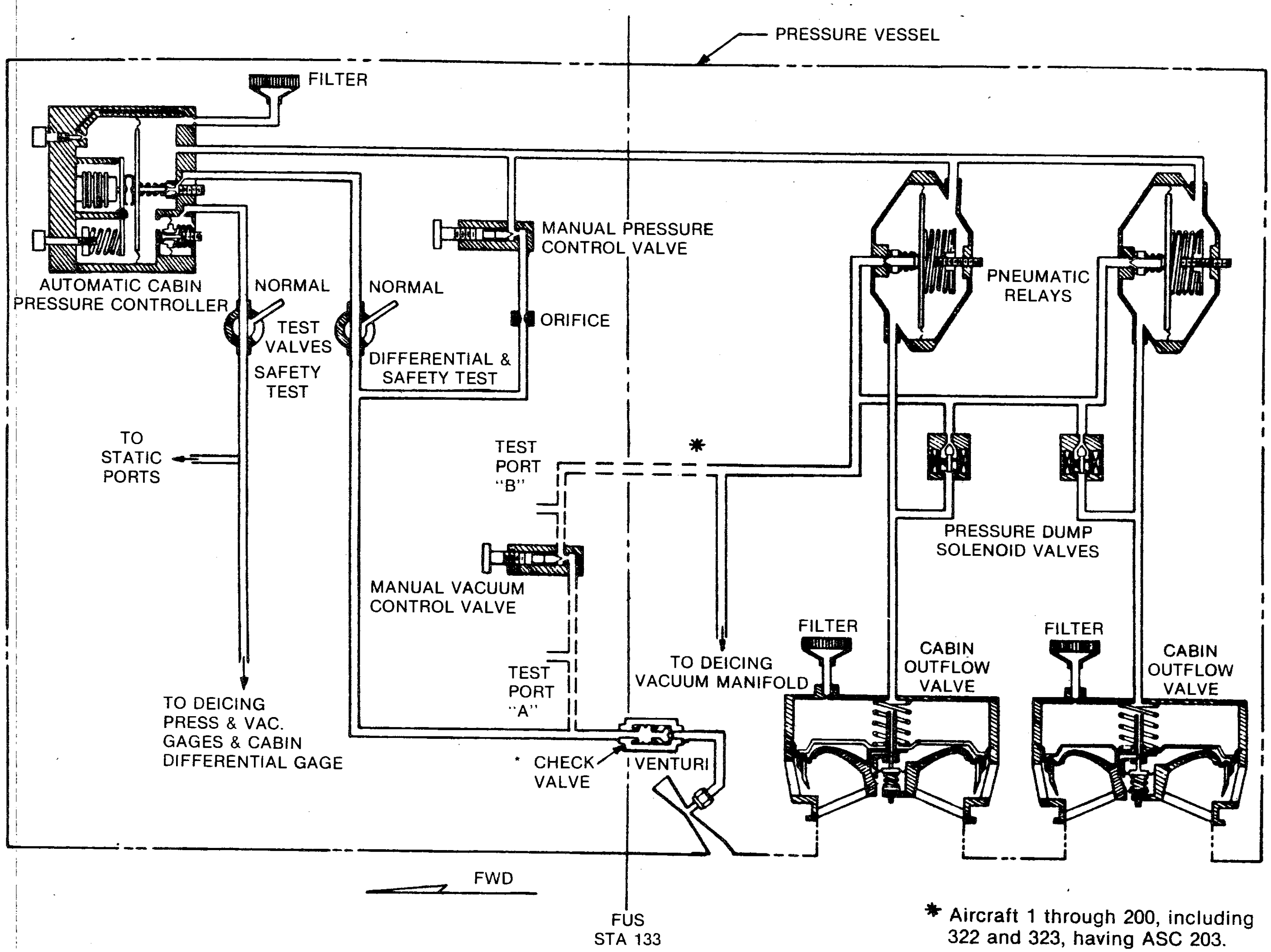
D. Isobaric Range.

As the pressure in the head chamber continues to decrease and the selected isobaric altitude (the cabin cruising altitude) is approached the isobaric bellows expands sufficiently to override the rate control diaphragm and cause the isobaric metering valve to start closing. With the head chamber pressure maintained at essentially a constant value, variations in pressure applied to the outflow valves causes the valves to operate to maintain cabin pressure at a selected value.

E. Normal Maximum Differential Pressure Operation (6.55 ± 0.2 psi).

The normal differential control serves to prevent cabin pressure from exceeding aircraft structural design limits. As the aircraft continues to climb to an altitude where the differential pressure between the cabin and the atmosphere approaches the calibrated differential setting of the control (6.55 ± 0.2 psi) the differential pressure between the head chamber and the atmosphere is sufficient to force the differential diaphragm to move up, opening the differential metering valve and releasing head chamber air to static, which is actually a 6.55 psi vacuum in relation to cabin pressure. As the aircraft climb continues and head chamber pressure continues to decrease, cabin pressure acts to open the outflow valves. The outflow valves operate to maintain the calibrated differential pressure between the cabin and atmosphere as directed by the controller at 6.55 ± 0.2 psi.

As the aircraft descends and atmpospheric pressure increases, the differential metering valve closes and pressure within the head chamber increases. Head chamber air bleeds into the reference pressure chamber at the rate determined by the rate control knob setting. If this increase (this increase refers to head chamber pressure, not reference chamber) exceeds the rate control setting, the rate control diaphragm opens the isobaric metering valve to reduce the rate of cabin pressure increase. As the descent continues into the isobaric range, the increase in pressure in the reference chamber causes the isobaric bellows to become effective, and the action is substantially the reverse of that which occured during the ascent.



* Aircraft 1 through 200, including 322 and 323, having ASC 203.

Cabin Pressure Control System Schematic
Figure 5

CABIN PRESSURIZATION CONTROL SYSTEM — FAULT ISOLATION

Fault	Possible Cause	Correction
Cabin outflow valves do not fully close (isobaric or manual operation).	Break line to controller at pneumatic relay. If valves close, the following are possible causes.	
	Manual pressure control valve is open	Close manual pressure control valve.
	Manual pressure control valve leaks.	Tighten fittings at Manual pressure control valve, or replace valve.
	Incorrect signal from automatic cabin pressure controller.	Replace controller. Replace filter.
	Faulty pneumatic relay.	Replace relay.
	If valves remain open, the following are possible causes:	
	Dump solenoids energized or leaking.	De-energize, replace solenoid.
	Sticky outflow valves.	Replace outflow valves.
	Pneumatic relay open.	Replace relay.
	Clogged filter on outflow valve.	Replace filter.
	Break line from pneumatic relay and outflow valve if valve still remains open.	
	Sticky outflow valve.	Replace outflow valve.
	Clogged filter.	Replace filter.

Fault	Possible Cause	Correction
Cabin outflow valves do not open (isobaric operation) with 1000 feet dialed into controller	Test valves under right console are not set to NORMAL.	Set valves to NORMAL.
	Install test gage between pneumatic relays and controller. Readings should indicate 2 inches or more of water.	
	If reading does not indicate 2 inches or more, controller is not giving proper signal.	Replace controller.
	Insufficient vacuum from pressure control venturi.	Force more air through venturi.
	Incorrect plumbing from pressure control venturi to controller.	Correct plumbing.
	If reading on test gage indicates 2 inches or more of water, remove gage and install it between pneumatic relay and cabin outflow valve. If test gage now indicates 2 inches or more of water the following are possible causes:	
	Outflow valves stuck.	Clean or replace outflow valves.
	If test gage does not indicate 2 inches or more of water, the pneumatic relay is not opening.	Replace pneumatic relay.
During manual operation cabin outflow valves do not open.	Check for vacuum between pneumatic relay and manual pressure control valve. No vacuum at this point indicates that control valve is malfunctioning, that orifice is clogged, or that no air is flowing through pressure control venturi.	Replace Manual Pressure control valve.
		Clean or replace orifice
		Supply air to venturi.
If one outflow valve does not open during manual dump operation, with AIR COND - VENT switch to VENT.	Defective wiring.	Refer to applicable wiring diagram in Wiring Manual.
	Defective pressurization dump Solenoid Valve.	Replace solenoid valve.
	Lack of deicing system vacuum.	Check vacuum at relay
If outflow valve does not close during manual dump - VENT switch to AIR COND.	Defective wiring diagram in Wiring Manual. Defective pressurization dump solenoid valve.	Refer to applicable wiring operation, with AIR COND. Replace solenoid valve.
	Clogged orifice of clogged filter on outflow valve.	Replace orifice. Replace filter.

NOTE: The following steps apply only during 6.55 psi and 7.2 psi ground checkout procedure in maintenance practices in this section.

Fault	Possible Cause	Correction
Cabin pressure does not rise as needle valve is slowly closed during maximum normal differential test.	Defective outflow valve	Replace outflow valve
	Clogged filter	Replace filter
	Leak in system ducting.	Check ducts for leaks.
	Defective GTC load valve.	Replace GTC valve.
Cabin pressure does not stabilize between 6.35 and 6.75 psi or 12.92 and 13.75 in. Hg after closing needle valve during normal differential test.	Static ports clogged.	Clean ports.
	Break in line.	Replace line.
	Leak in lines.	Tighten and check fittings in the system.
	Automatic cabin pressure controller faulty.	Replace controller.

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CABIN PRESSURIZATION CONTROL SYSTEM — MAINTENANCE PRACTICES

1. Cabin Pressure Control — Operational Test

NOTE: Perform prior to Maximum Normal Differential — Operation Test.

A. Remove cabin altimeter pressure warning switch, and all other pressure sensitive devices. Interconnect static hoses normally connected to these units.

CAUTION: ENSURE THERE ARE NO LEAKS IN FLEX HOSE BEFORE CONNECTING TO TEST FITTING IN NOSE WHEEL WELL.

B. Connect flex hose with pressure gage (0 - 10 psi) or manometer (0 - 20 inHg) to test fitting in nose wheel well located on left side Fuselage Station 85.

C. Open the following circuit breakers: (Pilots circuit breaker panel.)

- RAM AIR
- CBN PRESS UNL

D. Depress (close) the CBN/CKP TEMP circuit breaker. (Pilots circuit breaker panel)

E. Turn manual pressure control knob to full INCREASE position.

F. Turn DIFFERENTIAL and SAFETY test valve under copilots console to test position (handle up).

G. Remove line from aft pneumatic relay to aft outflow valve.

H. Cap fittings on outflow valve and pneumatic relay.

2. Maximum Normal Differential — Operation Test

A. Before performing this procedure perform Cabin Pressure Control — Operational Test, this Section.

(1) Remove cabin altimeter pressure warning switch, and all other pressure sensitive devices. Interconnect static hoses normally connected to these units.

CAUTION: ENSURE THERE ARE NO LEAKS IN FLEX HOSE BEFORE CONNECTING TO TEST FITTING IN NOSE WHEEL WELL.

(2) Connect flex hose with pressure gage (0 - 10 psi) or manometer (0 - 20 inHg) to test fitting in nose wheel well located on left side Fuselage Station 85.

(3) Open the following circuit breakers: (Pilots circuit breaker panel.)

- RAM AIR
- CBN PRESS UNL

(4) Depress the CBN/CKP TEMP circuit breaker. (Pilots circuit breaker panel)

(5) Turn manual pressure control knob to full INCREASE position.

(6) Turn DIFFERENTIAL and SAFETY test valve under copilots console to test position (handle up).

(7) Remove line from aft pneumatic relay to aft outflow valve.

(8) Cap fittings on outflow valve and pneumatic relay.

(9) Connect flex hose with needle valve (valve, when operated must be capable of completely unloading APU) to bulkhead fitting APU container (upper forward right side), and open needle valve.

(10) Hold cabin and cockpit temperature toggle switches in DECREASE position for 40 seconds.

(11) Start APU and set AIR switch to ON or to FLIGHT if having CB124.

(12) Close all windows, evacuate aircraft and close doors.

(13) Slowly close down needle valve (raise cabin pressure slowly).

(14) Cabin pressure should rise and stabilize between 6.35 and 6.75 psi or 12.92 and 13.75 inHg.

(15) Open needle valve and depressurize aircraft until gage or manometer reads 0.

WARNING: DO NOT OPEN DOORS UNTIL ALL DIFFERENTIAL IS BLED OFF.

(16) Connect line removed above.

(17) Remove line from forward pneumatic relay to forward outflow valve.

(18) Cap fittings on outflow valve, and pneumatic relay.

(19) Connect flex hose with needle valve (valve, when operated must be capable of completely unloading APU) to bulkhead fitting APU container (upper forward right side), and open needle valve.

(20) Hold cabin and cockpit temperature toggle switches in DECREASE position for 40 seconds.

(21) Start APU and set AIR switch to ON or FLIGHT if having CB124.

(22) Close all windows, evacuate aircraft and close doors.

(23) Slowly close down needle valve (raise cabin pressure slowly).

(24) Cabin pressure should rise and stabilize between 6.35 and 6.75 psi or 12.92 and 13.75 inHg.

(25) Open needle valve and depressurize aircraft until gage or manometer reads 0.

(26) Connect forward outflow valve line removed above.

B. Maximum Normal Differential Operation, using shop air supply, and perform steps as follows:

(1) Remove cabin altimeter pressure warning switch, and all other pressure sensitive devices. Interconnect static hoses normally connected to these units.

CAUTION: ENSURE THERE ARE NO LEAKS IN FLEX HOSE BEFORE CONNECTING TO TEST FITTING IN NOSE WHEEL WELL.

(2) Connect flex hose with pressure gage (0 - 10 psi) or manometer (0 - 20 inHg) to test fitting in nose wheel well located on left side Fuselage Station 85.

(3) Open the following circuit breakers: (Pilots circuit breaker panel.)

- RAM AIR
- CBN PRESS UNL

(4) Depress the CBN/CKP TEMP circuit breaker. (Pilots circuit breaker panel)

(5) Turn manual pressure control knob to full INCREASE position.

(6) Turn DIFFERENTIAL and SAFETY test valve under copilots console to test position (handle up).

(7) Remove line from aft pneumatic relay to aft outflow valve.

(8) Cap fittings on outflow valve and pneumatic relay.

(9) Depress RAM AIR circuit breaker. Set AIR COND - VENT switch to AIR COND position. After 10 seconds, pull RAM AIR circuit breaker (dc power must be supplied in this step).

(10) Connect shop air supply rig, equipped with shutoff valve, to air conditioning ducting on right side of aircraft at rear wing beam.

NOTE: Air supply must be capable of delivering air at the rate of 20 pounds per minute.

(11) Connect source of air pressure to pneumatic deicing system ground test connection in right nacelle. Deicing system PRESSURE gage should read 15 - 18 psi.

(12) Close all windows, evacuate aircraft and close doors.

(13) Open shutoff valve in air supply line, allowing aircraft to pressurize slowly.

(14) Cabin pressure should rise and stabilize between 6.35 and 6.75 psi, or 12.92 and 13.75 inHg.

(15) Depressurize aircraft by closing off shop supply air valve and bleeding off pressure.

(16) Connect line removed above to aft outflow valve.

NOTE: Deice pressure can be brought to zero quickly by deflating seal when differential pressure drops to approximately 1.0 psig.

(17) Remove line from forward pneumatic relay to forward outflow valve.

(18) Cap fittings on outflow valve and relay.

(19) Depress RAM AIR circuit breaker. Set AIR COND - VENT switch to AIR COND position. After 10 seconds, pull RAM AIR circuit breaker (dc power must be supplied in this step).

(20) Connect shop air supply rig, equipped with shutoff valve, to air conditioning ducting on right side of aircraft at rear wing beam.

NOTE: Air supply must be capable of delivering air at the rate of 20 pounds per minute.

(21) Connect source of air pressure to pneumatic deicing system ground test connection in right nacelle. Deicing system PRESSURE gage should read 15 - 18 psi.

(22) Close all windows, evacuate aircraft and close doors.

(23) Open shutoff valve in air supply line, allowing aircraft to pressurize slowly.

(24) Cabin pressure should rise and stabilize between 6.35 and 6.75 psi, or 12.92 and 13.75 inHg.

(25) Depressurize aircraft by closing off shop supply air valve and bleeding off pressure.

(26) Connect forward outflow valve line.

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CABIN PRESSURIZATION CONTROL SYSTEM — MAINTENANCE PRACTICES

1. **Cleaning Outflow Valves**

A. Clean outflow valves, which are located between Fuselage Station 133 and 157 on right side of fuselage under radio racks, as required. In addition, outflow valve filters should be checked and chänged in accordance with Gulfstream American GI Inspection Schedule times.

B. The following method is suggested for cleaning outflow valves:

(1) Place a vacuum hose against outflow valve (filter fittings) on right side of fuselage to lift off its seat. This same operation will remove accumulated dust and dirt.

(2) Wipe off and clean valve seats with mineral spirits conforming to Specification TT-T-291.

(3) When one valve has been cleaned satisfactory, clean other valve in same manner.

2. **Operational Test of the Cabin Pressure Control System Components**

A. Cabin Pressure Control System, Isobaric Operation, proceed as follows:

(1) Apply an external source of 28 volt, dc power and energize main dc bus and essential dc bus (BAT SW — NORM, EXT PWR SW – ON).

(2) Ensure a reading of 3 inHg or more is indicated on the deicer SUCTION gage (vacuum manifold), by performing any one of the following procedures:

(a) Connect source of air pressure to pneumatic deicing system in nacelle.

WARNING: TO PREVENT SERIOUS PERSONAL INJURY AND AIRCRAFT DAMAGE, THE MAIN DOOR OR THE DIRECT VISION WINDOWS MUST BE OPENED IF THE RIGHT ENGINE OR APU IS USED.

(b) Operate either of the main engines.

(c) Operate APU with full air output to cabin and set cabin and cockpit temperature controls for full cooling.

(3) Pull (open) RAM AIR and CBN PRESS UNL circuit breakers. (Pilot's circuit breaker panel.)

(4) Turn manual cabin pressure control needle valve on right console through its full travel in the INCREASE (clockwise) direction.

(5) Force air through venturi in outflow valve compartment.

(6) Turn the CABIN ALT. knob on the automatic cabin pressure controller on right console to adjust needle of the controller to a cabin altitude of 1000 feet below the actual altitude of the test site. The cabin outflow valves, located below the radio rack just aft of Fuselage Station 133, should close fully.

(7) Using the CABIN ALT. knob automatic cabin pressure controller, adjust needle of the controller to cabin altitude of 1000 feet above the pressure altitude of the test site. The cabin outflow valve must open fully.

B. Cabin Pressure Control System, Manual Operation, proceed as follows:

(1) Follow procedure of steps A. (1) through A. (6).

(2) Turn manual cabin pressure control needle valve in the DECREASE (counterclockwise) direction. The cabin outflow valves should open. Turn the needle valve in the INCREASE (clockwise) direction. The cabin outflow valves should close.

C. Cabin Pressure Control System, Manual Dump Operation, proceed as follows:

(1) Follow procedure of steps A.(1) and A.(2)

(2) Depress (close) RAM AIR circuit breaker.

(3) Pull (open) the following circuit breakers: (Pilot's circuit breaker panel)

LG HORN
LG WARN LTS
CBN PRESS UNL

NOTE: Aircraft 1 through 93 including 114 having ASC 108A, Part I incorporated and Aircraft 94 through 200 including 322 and 323, 114, the NUTCRACKER CONT circuit breaker is pulled (open) in place of the LG HORN and LG WARN LTS circuit breakers.

(4) Set AIR COND — VENT switch on right console to AIR COND.

(5) Follow procedure of steps A.(4) and A.(6).

(6) Set AIR COND — VENT switch to VENT. Cabin outflow valves should open fully.

(7) Set AIR COND — VENT switch to AIR COND. Cabin outflow valves should close fully.

D. Cabin Pressure Control System, Automatic Dump Operation, proceed as follows:

(1) Set AIR COND — VENT SWITCH to AIR COND.

(2) Follow procedure of steps A.(1) through A.(6)

(3) Depress (close) the following circuit breakers:

RAM AIR
LG WARN LTS
LG HORN

NOTE: Aircraft 1 through 93 including 114 having ASC 108A, Part I incorporated and Aircraft 94 through 200 including 322 and 323, the NUTCRACKER CONT circuit breaker is pulled (open) in place of the LG HORN and LG WARN LTS circuit breakers.

(4) With aircraft resting on main gear, the outflow valves should close when either main gear nutcracker switch is depressed. Check both right and left gear switches.

E. Prior to Operational Test of the Cabin Pressure Control System, Maximum Normal Differential Operation, proceed as follows:

(1) Remove cabin altimeter pressure warning switch, and all other pressure sensitive devices. Interconnect static hoses normally connected to these units.

CAUTION: ENSURE THERE ARE NO LEAKS IN FLEX HOSE BEFORE CONNECTING TO TEST FITTING IN NOSE WHEEL WELL.

(2) Connect flex hose with pressure gage (0-10 psi) or manometer (0-20 inHg) to test fitting in nose wheel well located on left side Fuselage Station 85.

(3) Open the following circuit breakers: (Pilot's circuit breaker panel.)

RAM AIR
CBN PRESS UNL

(4) Depress (close) the CBN/CKP TEMP circuit breaker. (Pilot's circuit breaker panel)

(5) Turn manual pressure control knob to full INCREASE position.

(6) Turn DIFFERENTIAL & SAFETY test valve under copilot's console to test position (handle up).

(7) Remove line from aft pneumatic relay to aft outflow valve.

(8) Cap fittings on outflow valve and pneumatic relay.

F. Cabin Pressure Control System, maximum normal differential operation. Proceed as follows using the aircraft APU as a pressure source:

(1) Complete steps E. (1) through E. (8).

(2) Connect flex hose with needle valve (valve, when operated must be capable of completely unloading APU) to bulkhead fitting APU container (upper forward right side), and open needle valve.

(3) Hold cabin and cockpit temperature toggle switches in DECREASE position for 40 seconds.

(4) Start APU and set AIR switch to on ("FLIGHT" if CB124 is incorporated).

(5) Close all windows, evacuate aircraft and close doors.

(6) Slowly close down needle valve (raise cabin pressure slowly).

(7) Cabin pressure should rise and stabilize between 6.35 and 6.75 psi or 12.92 and 13.75 inHg.

(8) Open needle valve and depressurize aircraft until gage or manometer reads 0.

WARNING: DO NOT OPEN DOORS UNTIL ALL DIFFERENTIAL IS BLED OFF.

(9) Connect line remove in step E. (7) to aft outflow valve.

(10) Remove line from forward pneumatic relay to forward outflow valve.

(11) Cap fittings on outflow valve, and pneumatic relay.

(12) Perform steps F.(2) through F.(7).

(13) Depressurize aircraft using procedure in step F. (8).

(14) Connect forward outflow valve line removed in step F.(10).

G. Cabin Pressure Control System, maximum normal differential operation, using shop air supply, perform steps E.(1) through E.(8), and proceed as follows:

(1) Depress (close) RAM AIR circuit breaker. Set AIR COND - VENT switch to AIR COND position. After 10 seconds, pull (open) RAM AIR circuit breaker (dc power must be supplied in this step).

(2) Connect shop air supply rig, equipped with shutoff valve, to air conditioning ducting on right side of aircraft at rear wing beam.

NOTE: Air supply must be capable of delivering air at the rate of 20 pounds per minute.

(3) Connect source of air pressure to pneumatic deicing system ground test connection in right nacelle. Deicing system PRESSURE gage should read 15-18 psi.

(4) Close all windows, evacuate aircraft and close doors.

(5) Open shutoff valve in air supply line, allowing aircraft to pressurize slowly.

(6) Cabin pressure should rise and stabilize between 6.35 and 6.75 psi, or 12.92 and 13.75 inHg.

(7) Depressurize aircraft by closing off shop supply air valve and bleeding off pressure.

(8) Connect line removed in Step E. (7) to aft outflow valve.

NOTE: Deice pressure can be brought to zero quickly by deflating seal when differential pressure drops to approximately 1 psig.

(9) Remove line from forward pneumatic relay to forward outflow valve.

(10) Cap fittings on outflow valve and relay.

(11) Perform Steps G.(1) through G.(6).

(12) Depressurize aircraft using procedure in Step G.(7).

(13) Connect forward outflow valve line removed in Step G.(9).

H. Cabin Pressure Control System, safety valve operation, proceed as follows:

(1) Follow steps E. (1) through E. (6).

(2) Turn safety test valve under copilot's console to TEST position (handle up).

(3) Remove line from aft pneumatic relay to aft outflow valve.

(4) Cap fitting on relay. Leave fitting on valve open.

(5) When using APU as pressure source, follow Steps F. (1) through F. (6). When using shop air as pressure source, follow Steps G. (1) through (5).

(6) Cabin pressure must rise and relieve between 7.00 and 7.40 psi, or 14.25 and 15.07 inHg.

(7) Depressurize aircraft by opening needle valve (if APU is used) or closing shop air supply valve (if shop air is used).

WARNING: WAIT UNTIL GAGE OR MANOMETER READS ZERO BEFORE OPENING DOOR.

(8) Connect lines removed in Step H. (3) to aft outflow valve.

(9) Remove lines from forward pneumatic relay to forward outflow valve.

(10) Cap fitting on relay. Leave fitting on valve open.

(11) Follow either procedure in Step H. (5).

(12) Cabin pressure should rise and stabilize between 7.00 and 7.40 psi, or 14.25 and 15.07 inHg.

(13) Depressurize aircraft per Step H. (7).

(14) Connect lines to forward outflow valve removed in Step H. (9).

(15) Return differential and safety test, and safety test valves under copilot's console to NORMAL and safety wire in this position.

(16) Reinstall instruments removed in Step E. (1) above.

(17) Return system to normal configuration depending on whether APU or shop air was used.

(18) Ensure that tight connections are made on all lines disconnected to perform test.

3. Pneumatic Relay Metering Valve — Leakage Test

CAUTION: IN THE EVENT THAT THERE IS ANY DOUBT AS TO THE SATISFACTORY CONDITION OF THE PNEUMATIC RELAY IT IS SUGGESTED THAT THE UNIT NOT BE PLACED IN SERVICE UNTIL FURTHER INFORMATION IS OBTAINED FROM AN AIRESEARCH REPRESENTATIVE OR FROM GULFSTREAM AEROSPACE. THIS TEST IS VALID ONLY FOR THE METERING VALVE PORTION OF THIS UNIT. FOR FURTHER TESTS REFER TO APPLICABLE PUBLICATION.

NOTE: The following test will ascertain whether the pneumatic relay metering valve is properly seated or has been worn beyond acceptable limits. It involves a source of vacuum, a suitable manometer and shutoff valves as shown in diagram of test set-up. (See Figure 201)

A. Remove pneumatic relay from the aircraft.

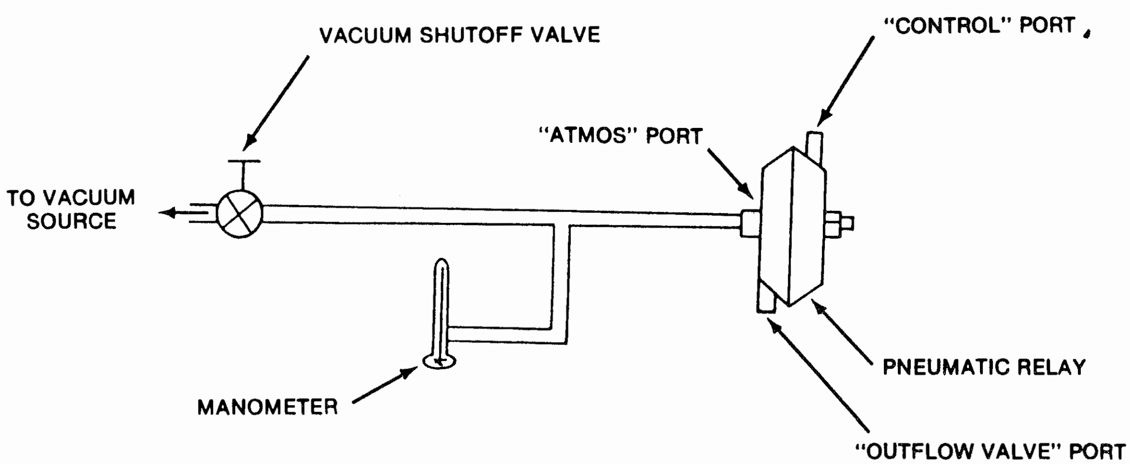
B. Attach source of vacuum to the pneumatic relay port marked "ATMOS". Remaining ports of pneumatic relay marked "CONTROL" and "OUTFLOW VALVE" must be vented to ambient.

C. Open vacuum valve and apply vacuum until measuring device indicates 10.0 ± 0.25 in Hg. Then close vacuum valve.

D. Open vacuum measuring device and note pressure increase in 1 minute. Pressure increase must not exceed 0.6 inHg.

E. If pressure increase exceeds above amount the leakage is excessive.

F. If test is satisfactory release vacuum.



Pneumatic Relay Metering Pin Test Setup
Figure 201

4. **Cabin Pressure Aneroid Switch — Functional Test**

NOTE: Steps A-H below apply to Aircraft 1-118 including 322, 323 not having ASC 259.

A. Pull WARNING LIGHTS circuit breaker on left circuit breaker panel.

B. Disconnect electrical connector from aneroid switch.

C. Connect meter/test lamp across pins B and C of electrical connector.

D. Reset WARNING LIGHTS circuit breaker, power should be indicated.

E. Pull WARNING LIGHTS circuit breaker. Disconnect meter/test lamp.

F. Connect electrical connector to cabin pressure warning aneroid switch. Reset circuit breaker.

G. Flight check aircraft unpressurized to minimum 10,000 feet pressure altitude.

H. Aneroid switch closes on decreasing pressure and should actuate, giving warning at 9250 ± 750 feet.

NOTE: Steps I.-M. below apply to Aircraft 1-118 including 322, 323 having ASC 259 and 119-200 only.

CAUTION: TO AVOID DEPLETING THE BATTERIES PERSONNEL INVOLVED IN THIS FUNCTIONAL TEST SHOULD BE FAMILIAR WITH THE PROCEDURE TO ENSURE A MINIMUM AMOUNT OF TIME IS USED.

I. Remove pneumatic fitting from cabin pressure sensing port and connect a vacuum source using an AN815-40 fitting.

J. Connect aircraft batteries and place battery switch to EMER.

K. Apply vacuum (slowly) to cabin pressure sensing port. CABIN PRESS light should come on at 9250 ± 750 feet altitude.

L. Reduce vacuum. CABIN PRESS light should go off before 8000 feet altitude is reached.

M. Reduce vacuum (slowly) and disconnect vacuum source. Place battery switch to OFF.

PRESSURIZATION SYSTEM — REMOVAL/INSTALLATION

1. **Removal of Automatic Pressurization Controller** (See Figure 401)

CAUTION: BEFORE PERFORMING ANY OF THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

NOTE: Aircraft 1 through 78 including 114 having ASC 105 incorporate and Aircraft 79 through 200 including 322 and 323, the pressurization controller is shock mounted, and flexible lines are provided in lieu of rigid lines.

A. Gain access to automatic controller by removing copilot's seat and access panel on copilot's console.

CAUTION: TAG AND IDENTIFY FLEXIBLE LINES AS THEY ARE DISCONNECTED.

B. Disconnect tubing at five points indicated in Figure 401.

C. Release six fasterners securing controller to console and remove controller.

D. Disconnect short rigid tubing between manual controller T-fitting and outflow valve port on automatic controller; separate controller from manual controller.

E. On shock mounted panel remove screws securing shock mounted panel to controller panel and remove shock mounted panel.

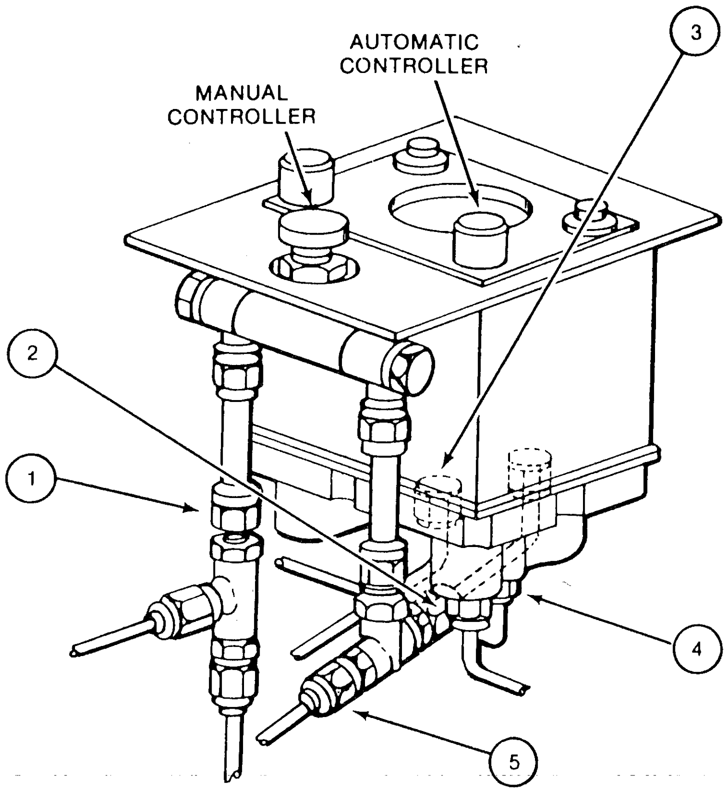


Figure 401
Pressurization Controller

F. Remove automatic controller mounting screws and remove automatic controller.

G. Install protective caps on all lines and ports.

2. **Installation of Automatic Pressurization Controller** (See Figure 401)

NOTE: At the operators' discretion, the replacement of the cockpit pressurization controller filter can be performed at this time.

If replacement controller is to be installed, transfer fittings and tubing form old unit to replacement unit. Orient fittings and tubing in same location and position as was installed on old unit.

A. Install controller and secure with mounting screws.

B. On shock mounted panel install shock mounted panel on controller panel and secure with screws.

C. Remove protective caps from all lines and ports.

D. Mount manual controller on automatic controller; connect short rigid tubing between manual controller tee fitting and outflow valve port on automatic controller.

E. Install controller in console and secure with fasterners.

F. Connect tubing at five points as indicated in Figure 401, remove tags from flexible lines.

G. Install access panel on copilot's console.

H. Install copilot seat.

I. Energize main and essential dc buses.

J. Connect pressurized air to pneumatic deicing system in nacelle.

K. Obtain 3 inHg vacuum in pneumatic deicing system vacuum manifold. Pull (open) RAM AIR and CBN PRESS UNL circuit breakers.

L. Turn manual cabin pressure control needle valve on right console to full increase position (clockwise).

M. Gain access to outflow valves and pressure control venturi (radio rack — lower forward).

N. Force air through venturi in outflow valve compartment.

O. Turn CABIN ALT. control on automatic controller; adjust needle to obtain cabin altitude of 1000 feet below actual altitude of test site. Cabin outflow valve should close.

P. Turn manual control needle valve to decrease (counterclockwise) direction. Cabin outflow valves should open.

Q. Turn manual control needle valve to increase (clockwise) direction. Cabin outflow valves should close.

R. Turn CABIN ALT control on automatic controller; adjust needle to a cabin altitude of 1000 feet above actual altitude of test site. Cabin outflow valves should open fully.

S. Turn off and disconnect air; deenergize main and essential dc buses; return system to its normal configuration.

3. **Removal of Forward and Aft Outflow Valves**

A. Gain access to outflow valves located between Fuselage Stations 133 and 157 under radio rack.

B. Disconnect rigid tubing from outflow valve elbow fitting.

C. Release V-band clamp securing valve to mounting base.

D. Remove outflow valve and gasket. Inspect and replace as required.

E. Install protective caps on all lines and parts.

4. Installation of Forward and Aft Outflow Valves

CAUTION: ENSURE SAFETY PRESSURE RELIEF PORT IN THE THROAT OF THE OUTFLOW VALVE IS UNCAPPED AND OPEN.

NOTE: If a replacement valve is to be installed, transfer filter and fittings to new valve. Use new gaskets under jam nuts are required. Orient fittings in same direction as old valve.

At the operators discretion replacement of the outflow valve filter cartridge(s) can be performed at this time.

A. Remove protective caps.

B. Install gasket and orient outflow valve properly on mount.

C. Tighten V-band clamp securely.

D. Connect rigid tubing to elbow fitting.

E. Energize main and essential dc buses.

F. Connect pressurized air to deicing system in nacelle.

G. Obtain 3 inHg vacuum in pneumatic deicing system vacuum manifold.

H. Pull (open) following circuit breakers: LG HORN LG WARN LTS, and CBN PRESS UNL.

NOTE: Aircraft 1 - 93 including 114 having ASC 108A, part 1 and Aircraft 94 - 200 including 322 and 323, the NUTCRACKER CONT circuit breaker should be pulled (open) in place of the LG HORN and LG WARN LTS circuit breakers.

I. Depress (close) RAM AIR circuit breaker.

J. On cabin pressure controller in cockpit, select MAX rate and set cabin altitude pointer to a setting of 1000 feet below pressure altitude to test site.

K. Turn manual cabin pressure control needle valve on right console to full increase (clockwise) position.

L. Place AIR COND-VENT switch to VENT. Outflow valves should open.

M. Place AIR COND-VENT switch to AIR-COND. Outflow valve should close.

N. Depress (close) LG HORN circuit breaker (NUTCRACKER CONT circuit breaker on Aircraft 1 - 93 including 114 having ASC 108A, part 1 and Aircraft 94 - 200 including 322 and 323), outflow valve should open.

O. Depress right nutcracker switch. Outflow valves should close.

P. Release right nutcracker switch. Outflow valves should open.

Q. Repeat Steps N and O using left nutcracker switch.

R. Turn off and disconnect air; deenergize main and essentail dc buses; return system to its normal configuration.

5. **Removal of Forward and Aft Pneumatic Relays**

   A. Gain access to pneumatic relays, located adjacent to outflow valves between fuselage station 133 and 157 under radio rack.

   B. Disconnect line from relay.

   C. Remove hardware securing relay; remove relay.

   D. Install protective caps on fittings and parts.

6. **Installation of Forward and Aft Pneumatic Relays**

   NOTE: If a replacement relay is to be installed, note position of fittings and transfer fittings to new relay. Use new gaskets as required.

   A. Remove protective caps from fittings and parts.

   B. Install relay using hardware previously removed.

   C. Connect line to relay.

   D. Energize main and essential dc buses.

   E. Connect pressurized air to deicing system in nacelle.

   F. Obtain 3 inHg vacuum in pneumatic deicing system vacuum manifold.

   G. Open (pull) RAM AIR and CABIN PRESSURE UNL circuit breakers.

   H. Turn manual cabin pressure control needle valve on right console to full increase (clockwise) position.

   I. Force air through venturi in outflow valve compartment.

   J. Turn cabin altitude knob automatic controller to adjust needle to a cabin altitude of 1000 feet BELOW actual altitude of test site: cabin outflow valves should close.

   K. Turn manual cabin pressure control needle valve to decrease (counterclockwise) direction; cabin outflow valves should open.

   L. Turn manual cabin pressure control needle valve to increase (clockwise) direction; cabin outflow valves should close.

   M. Turn cabin altitude knob on automatic controller to adjust needle to a cabin altitude of 1000 feet ABOVE pressure altitude of test site; cabin outflow valves should open fully.

   N. Turn off and disconnect air; deenergize main and essential dc buses; return system to its normal configuration.

   O. Close access previously opened.

7. **Removal of Cabin Pressure Aneroid Switch**

   A. Gain access to cabin pressure aneroid switch (located in hydraulic compartment, above hydraulic reservoir).

   B. Disconnect electrical connector; remove hardware securing switch to bracket; remove switch.

   C. Install protective caps on electrical connector and switch receptacle.

8. **Installation of Cabin Pressure Aneroid Switch**

NOTE: If aneroid pressure switch is to be replaced by a like unit remove the modified AN fitting and install it on the replacement switch prior to installation.

A. Remove protective caps from electrical connector and switch receptacle; remove cap from AN fitting (switch).

B. Install switch on bracket using hardware previously removed; connect electrical connector.

C. Close access previously opened.

9. **Removal of Manual Pressurization Controller**

NOTE: Aircraft 1 through 78 including 114 having ASC 105 incorporated and Aircraft 79 through 200 including 322 and 323, the pressurization control is shock mounted and flexible lines are provided in lieu of rigid lines. On aircraft not having ASC 105 incorporated disregard reference made to shock mounted panel.

A. Gain access to manual controller by removing panel directly aft of controller panel.

B. If necessary disconnect deicing panel plug and remove panel.

C. Disconnect short rigid tubing between manual controller to fitting and outflow valve port on automatic controller.

D. On shock mounted panel remove screws securing controller panel to console rail.

E. Support controller and remove shock mount screws.

F. Remove screws securing manual controller to panel and remove controller.

G. Install protective caps on all lines and ports.

10. **Installation of Manual Pressurization Controller**

A. Remove all protective caps.

B. Install manual controller on panel and secure with screws.

C. On shock mounted panel, install controller panel on shock mounted panel, and secure with screws.

D. Connect short rigid tubing to manual controller.

E. Secure panel to console rail with fasterners.

F. Install panel and connect deicing panel plug if removed.

G. Install access panel previously removed.

H. Energize main and essential dc buses.

I. Connect pressurized air to pneumatic deicing system in nacelle.

J. Obtain 3 inHg vacuum in pneumatic deicing system vacuum manifold. Pull (open) RAM AIR and CBN PRESS UNL circuit breakers.

K. Turn manual cabin pressure control needle valve on right console to full increase position (clockwise).

L. Gain access to outflow valves and pressure control venturi (radio rack-lower forward).

M. Force air through venturi in outflow valve compartment.

N. Turn CABIN ALT control on automatic controller; adjust needle to obtain cabin altitude of 1000 feet below actual altitude of test site. Cabin outflow valves should close.

O. Turn manual control needle valve to decrease (counterclockwise) direction. Cabin outflow valves should open.

P. Turn manual control needle valve to increase (clockwise) direction. Cabin outflow valves should close.

Q. Turn CABIN ALT control on automatic controller; adjust needle to obtain a cabin altitude of 1000 feet above actual altitude of test site. Cabin outflow valves should open fully.

R. Turn off and disconnect air; deenergize main and essential dc buses; return system to its normal configuration.

11. **Removal of Cabin Rate of Climb Indicator**

CAUTION: PULL EDGE LIGHTING PANEL STRAIGHT OUT WHEN REMOVING TO AVOID BENDING.

A. Remove edge lighting panel from left side of overhead panel by removing knobs from cabin and cockpit temperature control rheostats and screws which hold lighting panel to main panel.

B. Gain access to cabin rate-of-climb indicator by removing hardware and lowering overhead panel; support overhead panel while lowered.

C. Remove hardware securing indicator to panel (indicator is back mounted); remove indicator.

12. **Installation of Cabin Rate of Climb Indicator**

NOTE: If a replacement indicator is to be installed, ensure port at rear of indicator is open and clear.

A. Install indicator on panel using hardware previously removed.

B. Close overhead panel and secure with hardware previously removed.

C. Install edge lighting panel and secure with hardware previously removed; install cabin and cockpit temperature control rheostat knobs.

D. Energize main dc bus; rotate OVERHEAD EDGE rheostat on overhead panel lighting control section and check that edge lighting panel lights; rotate rheostat to OFF.

E. Deenergize main dc bus.

13. **Removal of Cabin Altimeter — Differential Pressure Gage**

CAUTION: PULL EDGE LIGHTING PANEL STRAIGHT OUT WHEN REMOVING TO AVOID BENDING.

A. Remove edge lighting panel from left side of overhead panel by removing knobs from cabin and cockpit temperature control rheostats and screws which hold lighting panel to main panel.

B. Gain access to cabin altimeter-differential pressure gage by removing hardware and lowering overhead panel; support overhead panel while lowered

C. Disconnect tubing from gage; install protect caps on fittings and ports.

D. Remove hardware securing gage to panel (gage is back mounted); remove gage.

14. **Installation of Cabin Altimeter—Differential Pressure Gage**

NOTE: If a replacement gage is to be installed, note position of fitting and transfer fitting from old gage to new gage.

A. Remove protective caps from fittings and ports.

B. Install gage on panel using hardware previously removed.

C. Connect tubing to gage.

D. Close overhead panel and secure with hardware previously removed.

E. Install edge lighting panel and secure with hardware previously removed; install cabin and cockpit temperature contro rheostat knobs.

F. Energize main dc bus; rotate OVERHEAD EDGE rheostat on overhead panel lighting control section and check that edge lighting panel lights; rotate rheostat to OFF.

G. Deenergize main dc bus.

15. **Removal of Forward and Aft Pressurization Dump Solenoid Valves**

A. Gain access to dump solenoid valves, adjacent to outflow valves between Fusealge stations 133 and 157 under radio rack.

B. Disconnect electrical connector.

C. Disconnect rigid tubing from valve; remove valve.

D. Install protective caps on electrical connector; receptacle, fittings and ports.

16. **Installation of Forward and Aft Pressurization Dump Solenoid Valves**

A. Remove protective caps from electrical connector, receptacle, fittings and ports.

B. Install valve by connecting rigid tubing.

C. Connect electrical connector.

D. Energize main and essential dc buses.

CAUTION: IF APU AIR IS USED, ENSURE DV WINDOW OR A DOOR IS OPEN TO PREVENT PRESSURIZING.

E. Obtain 3 inHg vacuum in deicing system vacuum manifold by operating APU with air on or connecting pressurized air to pneumatic system in nacelle; cabin outflow valves should open (weight on wheels).

F. Open (pull) LG HORN and LG WARN LTS circuit breakers; cabin outflow valves should close.

NOTE: Aircraft 1 through 93 including 114 having ASC 108A, Part 1 incorporated and Aircraft 94 through 200 including 322 and 323, the NUTCRACKER CONT circuit breaker should be open (pulled) in place of the LG HORN and LG WARN LTS circuit breakers.

G. Engage LANDING GEAR WARNING HORN and LANDING GEAR WARNING LIGHTS circuit breakers or NUTCRACKER circuit breaker; cabin outflow valves should open.

H. Turn off and disconnect air (or shut down APU); deenergize main and essential dc buses; return system to normal configuration.

I. Close access previously opened.

17. **Removal of Cabin Air Temperature Bulb**

A. Gain access to cabin air temperature bulb.

B. Remove electrical connector from bulb.

C. Remove jam nut and washer securing bulb to thermostat bracket and remove bulb.

18. **Installation of Cabin Air Temperature Bulb**

A. Install bulb on thermostat bracket and secure with washer and jam nut.

B. Connect electrical connector

C. Energize main and essential dc buses.

19. **Removal of Cabin Air Temperature Indicator**

A. Remove knobs from cabin and cockpit temperature control rheostats.

CAUTION: DO NOT USE FORCE TO REMOVE LIGHTING PANEL AND AVOID TWISTING OR BENDING PANEL.

B. Remove screws that secure right side edge lighting panel to overhead panel and remove panel.

C. Remove screws securing overhead panel to structure and move panel away from structure. Support panel to prevent strain on wiring and lines.

D. Disconnect electrical connector from indicator.

E. Remove screws securing indicator and remove indicator from panel.

20. **Installation of Cabin Air Temperature Indicator**

A. Install indicator in panel and secure with screws.

B. Connect electrical connector to indicator.

C. Install overhead panel on structure and secure with screws.

D. Install edge lighting panel on overhead panel and secure with screws,

E. Install knobs on cabin and cockpit temperature control rheostats.

F. Energize main and essential dc buses.

G. Check cabin temperature indicator for proper indication.

H. Check overhead edge lighting panel for proper operation, use lighting control on overhead panel.

I. Deenergize main and essential dc buses.

ISOBARIC / DIFFERENTIAL / RATE CHANGE — DESCRIPTION / OPERATION

1. **General**

A. Unpressurized, Isobaric, Differential, and Rate of Change Ranges

(1) Isobaric Range

It is important to understand the difference between the unpressurized, isobaric and the differential ranges. To illustrate each, a typical flight is described with the automatic pressurization control system programmed by the crew for a 20,000 foot maximum pressure altitude flight plan. This program enables passengers to fly in a cabin with a pressure of 3000 ft., while the aircraft altitude is actually at 20,000 feet. The following would be the action of the system disregarding rate of change.

After the pressure source (blower) is cut in (right after takeoff), and the aircraft climbs, (for example, starting at a field with a sea-level pressure elevation) between sea level and 3000 ft., the cabin is unpressurized. The outflow valves are fully open and the cabin pressure is equal to the ambient pressure. This is the unpressurized range for this flight.

As the 3000 foot altitude is approached, the automatic controller senses this and starts to close the outflow valves. As a result, outflow air is restricted and the cabin begins to pressurize. This pressurization is intended to hold a 3000 foot cabin altitude as the aircraft continues to climb toward 20,000 feet. As the aircraft pressurizes, the pressure differential builds up, since the cabin pressure is holding at the 3000 foot level, while the ambient pressure is dropping during the climb. Pressurization continues until the aircraft levels off at 20,000 feet. At this point the pressure differential is very close to 6.55 psi.

Thus, a constant 3000 foot cabin pressure is maintained from an ambient altitude of 3000 feet to 20,000 feet. This is the isobaric range for this particular flight.

If it is desired to climb the aircraft to 25,000 feet with the pressurization system programmed for the 3000 foot cabin (20,000 foot flight plan), at approximately the 20,500 foot point in the climb the differential pressure reaches 6.55 psi. Since this cannot be exceeded, the cabin altitude starts going up in direct proportion to the aircraft altitude, with a difference of 6.55 psi between the two. The fixed pressure differential of the system (6.55 psi) does not allow this point to be exceeded. Therefore, the cabin pressure must go up in proportion to the aircraft altitude. The cabin rate of climb is then almost the same as the aircraft rate of climb, since the change is proportional.

Therefore, from 20,000 to 25,000 feet the aircraft rides the 6.55 psi differential, and is consequently in the differential range of operation.

If the aircraft levels off at 25,000 feet, the cabin levels off at approximately 5500 feet, which is the 6.55 psi differential at that altitude.

From this example, you can see that if this system was committed to a fixed isobaric range, plus fixed differentials, it would be a simple operation. In the Gulfstream I this is not the case, since crew and passenger comfort consistent with safety is the primary consideration.

A variable isobaric pressurization system enables the crew to "custom make" their own pressurization schedule for that particular flight. It also enables the crew to change their isobaric range at any time during that flight in the event that their flight plan changes. The object, naturally, is to keep the lowest cabin altitude for the particular flight plan altitude without operating on maximum differential control. The Gulfstream I pressurization system is a variable isobaric system.

(2) Rate of Change

With the high rate of climbs or descents, that modern high performance aircraft are capable of, another facet of pressurization comes into the limelight. This is RATE OF CABIN CHANGE. It is not necessarily the pressure which physically effects a person as long as the pressure is somewhere below 10,000 feet, but it is the changes in pressure which become objectionable to the average

passenger and crew member. Therefore, rapid pressure changes are considered objectionable. A rate of climb pressure change somewhere between 300 to 500 feet per minute is considered acceptable by most individuals, and any range of operation in which the cabin pressure change is close to or equal to aircraft rate of climb or descent (provided they are in excess of 500 feet per minute) can be considered objectionable. The UNPRESSURIZED RANGE and the DIFFERENTIAL RANGE can be considered rapid change ranges. The cabin rate of change function of pressurization can take over, in rapid change range. When the rate of change system is properly employed, it modifies and/or eliminates rapid pressure change ranges. A RATE knob on the automatic controller is utilized by the crew to select an appropriate rate of cabin pressure change commensurate with the particular flight plan. This function with the pressurization system limits cabin pressure changes only as long as it does not require the system to override the normal maximum differentials. The rate of change function, therefore is subordinate to any fixed differentials (6.55 pressure or 0.2 vacuum).

To illustrate the action of the cabin rate of change, consider the previous flight plan for a 20,000 foot altitude with a 3000 foot cabin pressurization. (Takeoff from sea level.)

(a) From takeoff with the system programmed to a 20,000 foot flight plan until the aircraft passed through the 3000 foot altitude, the rate system starts to pressurize the aircraft at a controlled rate less than the rate of climb. This is a "time-rate" problem. If the rate of climb is 2000 ft/min. it takes approximately 10 minutes to get to 20,000 feet. In that 10 minutes, the cabin altitude must go from sea level to 3000 feet. Dividing 3000 by 10 minutes, yields 300 feet per minute. Allowing a margin for error, a rate of cabin pressure change slightly above that selected (500 feet per minute).

(b) This value is set into the controller by positioning the rate knob on the automatic controller until the cabin rate of climb indicator on the overhead panel reads 500 feet per minute during climbout. As the aircraft climbs at 2000 ft/min., the cabin pressure changes at only 500 feet per minute. The object is to allow fast rate of aircraft climb or descent, without rapid cabin pressure changes.

(c) The approximate cabin altitude and pressure, versus aircraft altitude and pressure using a 500 foot per minute setting is shown below with the cabin differential.

NOTE: A setting of the RATE knob will not yield an exactly constant rate of change. The rate of pressure change will vary with the desired correction or change to pressure and the pressure differential.

Time (Minutes)	Cabin Alt (ft)	Aircraft Alt (ft)	Ambient Press. (psia)	Cabin Press. (psia)	Differential (psig)
0	0	0	14.7	14.7	0
1	500	2000	13.7	14.5	0.8
2	1000	4000	12.7	14.2	1.5
3	1500	6000	11.8	13.9	2.1
4	2000	8000	10.9	13.7	2.8
5	2500	10000	10.1	13.4	3.3
6	3000	12000	9.3	13.2	3.9
7	3000	14000	8.6	13.2	4.6
8	3000	16000	7.9	13.2	5.3
9	3000	18000	7.3	13.2	5.9
10	3000	20000	6.7	13.2	6.5

(d) As indicated, from the sixth minute on, the cabin altitude reaches 3000 feet. (the isobaric range) and from then on no change in cabin altitude is in effect. The aircraft rate of climb does not have to be modified in any way to compensate for a rapid cabin pressure change.

(e) This system can also be used to raise or lower the cabin altitude in flight if it is desired to change flight plan altitude as long as the following things are taken into consideration: the

6.55 differential cannot be exceeded, the longer time given the system to change, the slower the change will be (less noticeable), and the more the change in cabin altitude, the longer it takes.

(f) When changing the flight plan the automatic pressurization controller is reprogrammed for a 25,000 ft. flight altitude before the actual change in altitude. This gives an isobaric range of 5500 feet instead of 3000 feet With the cabin rate of change system set for 500 ft/min. rate, pressurization starts bringing the cabin altitude up to the 5500 feet at a rate of 500 ft/min. If clearance is received to go this altitude, say 3 minutes from the time the change is programmed into the system, the cabin is already at 4500 feet.

(g) As the aircraft climbs to 25,000 at 1000 ft/min taking another 5 minutes, the cabin reaches 5500 feet and no further change is made as the aircraft passes through the 22,000 feet altitude. The last 3 minutes of climb, there is no cabin pressure change. Even so, cabin change has only been at a rate of 500 ft/min.

(h) The cabin and aircraft altitudes and pressures and cabin differential for the reprogrammed 25,000 feet flight are shown below.

Time (Minutes)	Cabin Alt (ft)	Aircraft At (ft)	Ambient Press. (psi)	Cabin Press. (psi)	Differential (psi)
0	3000	20,000	6.7	13.2	6.5
1	3500	20,000	6.7	12.9	6.2
2	4000	20,000	6.7	12.7	6.0
3	4500	20,000	6.7	12.5	5.8
4	5000	21,000	6.4	12.3	5.9
5	5500	22,000	6.2	12.0	5.8
6	5500	23,000	5.9	12.0	6.1
7	5500	24,000	5.7	12.0	6.3
8	5500	25,000	5.4	12.0	6.6

Descents from altitude are similar time-rate problems. The rate control is set accordingly, but the big difference is that this should not be done until let down is actually started and the isobaric range must be reprogrammed to the approximate pressure elevation of the landing field. These two actions accomplish the following: shift the isobaric range downward to the field elevation pressure to assure that pressurization is neutralized at touchdown; and allow the pressure change within the cabin to occur at a controlled rate providing that the system does not have to override one of the differentials.

The discussion above for a rate of change system considered a case in which the aircraft flight plan was for a climb path.

Operation of the pressurization system is similar for a descent procedure. Assume that the aircraft is to make a straight descent from an altitude of 25,000 feet ambient to sea level in 10 minutes under cabin conditions of 5500 feet before descent. The sea level isobaric range is immediately established on the CABIN ALT, knob of the automatic controller.

Since the aircraft must go from a cabin altitude of 5500 feet to a sea level (0. feet) altitude in 10 minutes the rate of change is 550 ft/min (5500 feet divided by 10 min). If a 50 fpm margin of error is allowed, the rate of change set in on the RATE control knob is 600 ft/min. With the aircraft descent rate at 2500 ft/min (25,000 feet divided by 10 min) the pressurization system compensates for 1900 feet of altitude in the cabin. This is summarized in the following list.

NOTE: On a descent, to minimize passenger discomfort, try to eliminate the unpressurized range by setting the rate to bring the cabin down to as close to field pressure as possible. Many pilots have their system programmed for a slight amount of pressure in the cabin (approximately 100 feet) on touchdown to avoid being unpressurized for the last part of their landing (rapid pressure change range). The pressurization outflow valves automatically dump this pressure as soon as there is weight on both gears through the nutcracker system.

Time (Minutes)	Cabin Alt (ft)	Aircraft Alt (ft)	Ambient Press. (psi)	Cabin Press. (psi)	Cabin Differential (psi)
0	5500	25,000	5.4	12.0	6.6
1	4900	22,500	6.1	12.3	6.2
2	4300	20,000	6.7	12.6	5.9
3	3700	17,500	7.5	12.9	5.4
4	3100	15,000	8.3	13.2	4.9
5	2500	12,500	9.2	13.4	4.2
6	1900	10,000	10.1	13.7	3.6
7	1300	7500	11.2	14.0	2.8
8	700	5000	12.3	14.3	2.0
9	100	2500	13.4	14.6	1.2
10	0	0	14.7	14.7	0.0

ISOBARIC / DIFFERENTIAL / RATE CHANGE — MAINTENANCE PRACTICES

1. **Cabin Pressure Control, Isobaric — Operational Test**

   A. Apply an external source of 28V, dc power and energize main dc bus and essential dc bus (BATT SW - NORM, EXT PWR SW - ON).

   B. Ensure a reading of 3 inHg or more is indicated on the de-icer SUCTION gage (vacuum manifold), by performing any one of the following procedures:

      (1) Connect source of air pressure to pneumatic deicing system in nacelle.

      WARNING: TO PREVENT SERIOUS PERSONAL INJURY AND AIRCRAFT DAMAGE, THE MAIN DOOR OR THE DIRECT VISION WINDOWS MUST BE OPENED IF THE RIGHT ENGINE OR APU IS USED.

      (2) Operate either of the main engines.

      (3) Operate APU with full air output to cabin and set cabin and cockpit temperature controls for full cooling.

   C. Pull (open) RAM AIR and CBN PRESS UNL circuit breakers. (Pilots circuit breaker panel.)

   D. Turn manual cabin pressure control needle valve on right console through its full travel in the INCREASE (clockwise) direction.

   E. Force air through venturi in outflow valve compartment.

   F. Turn the CABIN ALT. knob on the automatic cabin pressure controller on right console to adjust needle of the controller to a cabin altitude of 1000 feet below the actual altitude of the test site. The cabin outflow valves, located below the radio rack just aft of Fuselage Station 133, should close fully.

   G. Using the CABIN ALT. knob automatic cabin pressure controller, adjust needle of the controller to cabin altitude of 1000 feet above the pressure altitude of the test site. The cabin outflow valve must open fully.

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AUTOMATIC PRESSUIZATION CONTROLER — DESCRIPTION / OPERATION

1. General

(See Figure 1)

The automatic pressurization controller unit is located on the copilots console. By manipulation of knobs located on the unit, a selection can be made of the isobaric and rate of change pressures at which the system will operate during a particular flight.

The controller consists of a dual dial calibrated in thousands of feet, two knobs, and internal mechanisms. The controller contains three separate pressure-controlling systems: isobaric, selected pressure-rate-of-change, and normal maximum pressure differential (6.55 psi). This controller is the sensing or controlling device for the above three functions of the outflow valves. The controller senses or sets up the situation; the outflow valves actually do the work. Three input pressures are fed into the controller.

A. Static pressure, from flush static ports on the forward, right and left sides of the aircraft, near the nose.

B. Cabin pressure, through a filter beneath the controller vented to the pressurized area.

C. Vacuum from the pressure control venturi (which produces vacuum for the use of the controller). This vacuum is utilized by the controller as a portion of the output pneumatic signal, which in turn, through the pneumatic relays is used to control the outflow valves.

Manual controls for the unit are located on each side of the dial. These controls consist of the CABIN ALT. selector knob and the RATE control knob. The CABIN ALT. selector knob, in conjunction with the dial, the dial gear, and the pointer, provides a means of manually selecting the cabin altitude. The dial is calibrated in thousands of feet from — 1000 to + 10,000 feet cabin altitude, and is used in selection of the isobaric cabin altitude. The dial gear is calibrated to indicate the maximum aircraft altitude at which the selected isobaric range of operation can be maintained. The maximum aircraft altitude indication is visible through the window in the dial. For example, if an isobaric cabin altitude of 3000 feet is to be programmed, the CABIN ALT. selector knob is turned until the pointer on the main dial of the controller reads 3000 feet. At the same time, through the gear train, the indication in the small window beneath the dial reads approximately 20,500 feet. This means that the isobaric range of the unit can be maintained at a 3000 foot cabin altitude until an aircraft altitude of 20,500 feet is reached, at which time the 6.55 psi pressure differential takes over. This is merely a reference; it is not an altitude indication.

Programming of this controller for a particular flight involves turning the CABIN ALT. knob until the maximum flight plan altitude appears in the cut in the dial. The hand then indicates the lowest cabin altitude (isobaric) which can be maintained, providing that the aircraft altitude does not exceed the cut in the dial, and that aircraft altitude is above the reading on the handle.

If the controller is not reprogrammed on letdown, the isobaric range holds the cabin altitude until the aircraft passes through ambient altitude with the same value as the programmed altitude. From then on the aircraft is unpressurized, or goes into vacuum relief.

Re-programming to the landing field altitude is necessary to prevent running into the unpressurized range or vacuum differential on letdown. When the final adjustment of the controller is made (on the "in-range check list") it is normally set so that the hand reads landing field pressure altitude minus 100 feet This enables the aircraft to land slightly pressurized for maximum comfort, and avoids the unpressurized range.

The RATE control knob serves to select the rate of cabin pressure increase or decrease within limits established by the cabin pressure differential setting. It indicates only minimum (ccw) and maximum (cw), with a small index arrow dead ahead. Most controllers with the knob arrow dead ahead (aligned with the index arrow on the case) produce a setting of about 500 ft/min rate of change. Turning the knob full CCW produces about a 50 ft/min rate. Full CW rotation will produce about a 2000 ft/min rate of change. This knob is used during changes of pressure in conjunction with the cabin rate of climb indicator located in the upper overhead panel, right side. This instrument is merely a standard, lagged-type rate of climb indicator vented into the cabin, and measures rate of climb or dive of the cabin pressure in thousands of feet per minute. The knob on the controller is adjusted and the rate of climb indicator tells the actual rate of change of cabin altitude.

2. Description

The automatic controller has the functions of isobaric, rate of change, and normal maximum differential control of the pressurization system. The isobaric system consists of a bellows, a rocker arm supported by a fulcrum and a yoke, a compression spring loaded by the cabin altitude selector knob. a cabin altitude pointer, and gearing connection this pointer to the cabin altitude selector knob. (See Figure 2) The bellows is of the evacuated and sealed aneroid type and is connected to the rocker arm. It is called the isobaric bellows. The CABIN ALT. selector knob travelling nut and spring permits selection of the isobaric cabin altitude. The bellows and rocker arm are connected to the rate control diaphragm by means of a leaf spring. The rate control diaphragm (this is the large diaphragm separating the two major chambers of the controller) separates the reference chamber from the head chamber, and actuates a spring loaded isobaric metering valve which is ported to vacuum from the pressure control venturi.

The selected pressure-rate-of-change control consists of a filter, a drilled passage with a capillary tube, a rate metering valve, and a valve seat. Adjustment of the rate metering valve is accomplished by means of the rate controller knob located on the front of the unit.

The normal maximum differential system (6.55 $\pm$ 2 psi) consists of a differential diaphragm, a metering valve and its seat, a diaphragm backing spring, and an adjustment screw. The force of the diaphragm spring at the metering valve opening position is adjusted when the unit is calibrated, and this adjustment remains fixed. The spring loaded side of the differential diaphragm is vented to atmosphere. The opposite side is exposed to head chamber pressure.

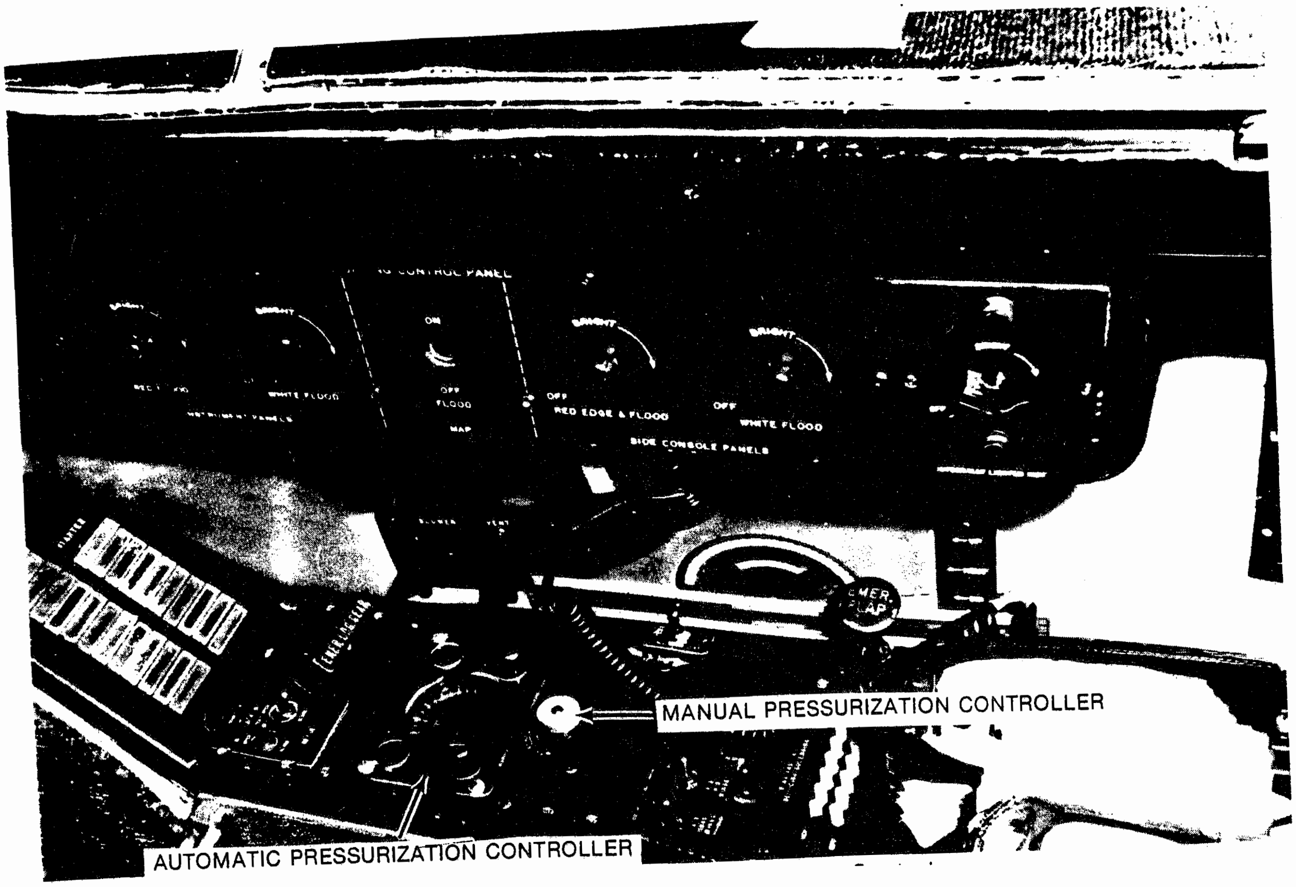
The small drilled passage in the controller has an opening into the reference pressure chamber and airflow between the two chambers is controlled by the rate control knob, which actually is a needle valve. Rapid changes in cabin pressure go directly through the filter into the head chamber, which is on one side of the rate control diaphragm. A small portion of the cabin air passes through the small drilled passage and the rate metering valve, into the reference chamber. Therefore, the pressure in the head control chamber changes almost immediately. Conversely, the rate of change in pressure in the reference chamber is directly proportional to the amount the rate control knob metering pin is opened. The magnitude of the pressure differential thus produced across the rate control diaphragm positions the isobaric metering valve, thereby controlling the rate of pressure change in the head control chamber and therefore the outflow valves. However, the selected rate of pressure change cannot override the maximum normal differential control and vacuum relief control so that the pressure differential of 6.55 psi and the vacuum differential of 0.2 psi are not exceeded.

All controller system connections are located on underside of the unit. The cabin pressure controller is designed to be mounted on a panel in the pressurized area. The controller must be connected to a true atmospheric pressure (static), a source of vacuum (pressure control venturi), cabin pressure (through filter) and to pneumatic relays.

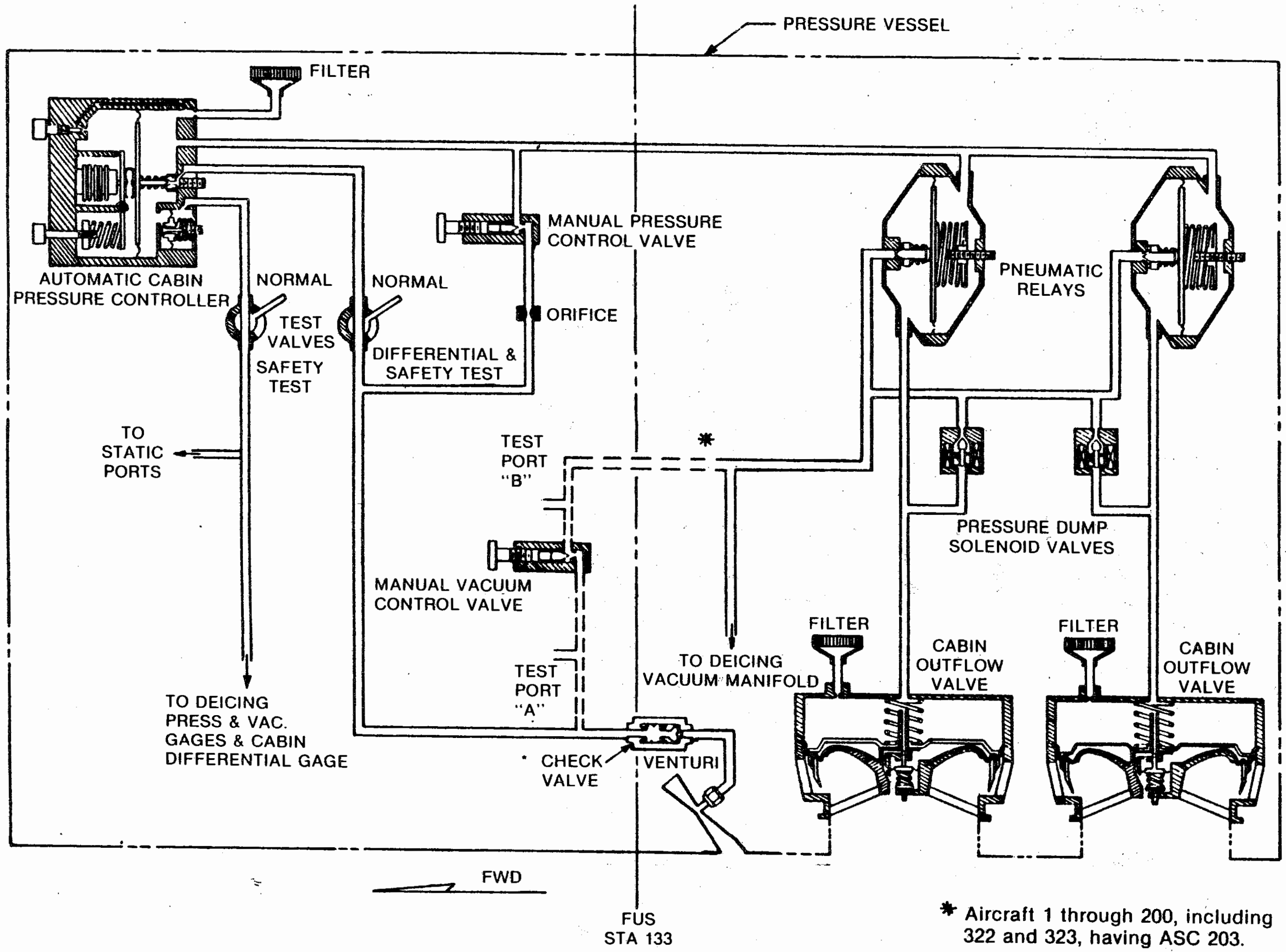
NOTE: This unit is not connected directly to the outflow valves. It must go through the pneumatic relays.

Tubing connections to automatic controller is as follows: (ports are marked)

Controller Port Marking	Connect To
Filter	Filter
ATMOS#1	Static Line
ATMOS#2	Pressure Control Venturi Vacuum
Outflow	Pneumatic Relays



Copilot Console Pressure Control System Controls
Figure 1.



* Aircraft 1 through 200, including 322 and 323, having ASC 203.

Cabin Pressure Control System — Schematic
Figure 2.

AUTOMATIC PRESSUIZATION CONTROLER — MAINTENANCE PRACTICES

1. Automatic Pressurization Controller — Removal / Installation

(See Figure 201)

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

NOTE: Aircraft 1 - 78, 114 having ASC 105 and Aircraft 79 - 200, 322 and 323, the pressurization controller is shock mounted, and flexible lines are provided in lieu of rigid lines.

A. Removal

(1) Gain access to automatic controller by removing copilots seat and access panel on copilots console.

CAUTION: TAG AND IDENTIFY FLEXIBLE LINES AS THEY ARE DISCONNECTED.

(2) Disconnect tubing at five points indicated in Figure 201.

(3) Release fasteners securing controller to console and remove controller.

(4) Disconnect short rigid tubing between manual controller T-fitting and outflow valve port on automatic controller; separate controller from manual controller.

(5) On shock mounted panel remove screws securing shock mounted panel to controller panel and remove shock mounted panel.

(6) Remove automatic controller mounting screws and remove automatic controller.

(7) Install protective caps on all lines and ports.

B. Installation

NOTE: At the operators discretion, the replacement of the cockpit pressurization controller filter can be performed at this time.

If replacement controller is to be installed, transfer fittings and tubing form old unit to replacement unit. Orient fittings and tubing in same location and position as was installed on old unit.

(1) Install controller and secure with mounting screws.

(2) On shock mounted panel install shock mounted panel on controller panel and secure with screws.

(3) Remove protective caps from all lines and ports.

(4) Mount manual controller on automatic controller; connect short rigid tubing between manual controller tee fitting and outflow valve port on automatic controller.

(5) Install controller in console and secure with fasteners.

(6) Connect tubing at five points as indicated in Figure 201 remove tags from flexible lines.

(7) Install access panel on copilots console.

(8) Install copilot seat.

(9) Energize main and essential dc buses.

(10) Connect pressurized air to pneumatic deicing system in nacelle.

(11) Obtain 3 inHg vacuum in pneumatic deicing system vacuum manifold. Pull (open) RAM AIR and CBN PRESS UNL circuit breakers.

(12) Turn manual cabin pressure control needle valve on right console to full increase position (clockwise).

(13) Gain access to outflow valves and pressure control venturi (radio rack - lower forward).

(14) Force air through venturi in outflow valve compartment.

(15) Turn CABIN ALT. control on automatic controller; adjust needle to obtain cabin altitude of 1000 feet below actual altitude of test site. Cabin outflow valve should close.

(16) Turn manual control needle valve to decrease (counterclockwise) direction. Cabin outflow valves should open.

(17) Turn manual control needle valve to increase (clockwise) direction. Cabin outflow valves should close.

(18) Turn CABIN ALT control on automatic controller; adjust needle to a cabin altitude of 1000 feet above actual altitude of test site. Cabin outflow valves should open fully.

(19) Turn off and disconnect air; de-energize main and essential dc buses; return system to its normal configuration.

2. Automatic Cabin Pressure Control Dump — Operational Test

A. Set AIR COND - VENT SWITCH to AIR COND.

B. Apply an external source of 28V, dc power and energize main dc bus and essential dc bus (BATT SW - NORM, EXT PWR SW - ON).

C. Ensure a reading of 3 inHg or more is indicated on the de-icer SUCTION gage (vacuum manifold), by performing any one of the following procedures:

(1) Connect source of air pressure to pneumatic deicing system in nacelle.

WARNING: TO PREVENT SERIOUS PERSONAL INJURY AND AIRCRAFT DAMAGE, THE MAIN DOOR OR THE DIRECT VISION WINDOWS MUST BE OPENED IF THE RIGHT ENGINE OR APU IS USED.

(2) Operate either of the main engines.

(3) Operate APU with full air output to cabin and set cabin and cockpit temperature controls for full cooling.

D. Pull (open) RAM AIR and CBN PRESS UNL circuit breakers. (Pilots circuit breaker panel.)

E. Turn manual cabin pressure control needle valve on right console through its full travel in the INCREASE (clockwise) direction. (See Figure 202)

F. Force air through venturi in outflow valve compartment.

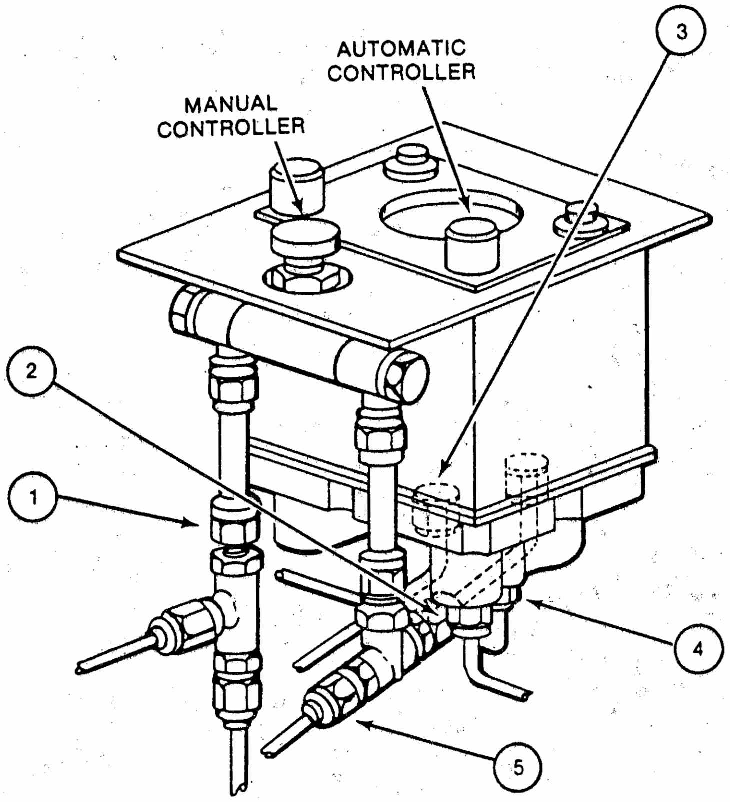
G. Turn the CABIN ALT. knob on the automatic cabin pressure controller on right console to adjust needle of the controller to a cabin altitude of 1000 feet below the actual altitude of the test site. The cabin outflow valves, located below the radio rack just aft of Fuselage Station 133, should close fully.

H. Depress (close) the following circuit breakers:

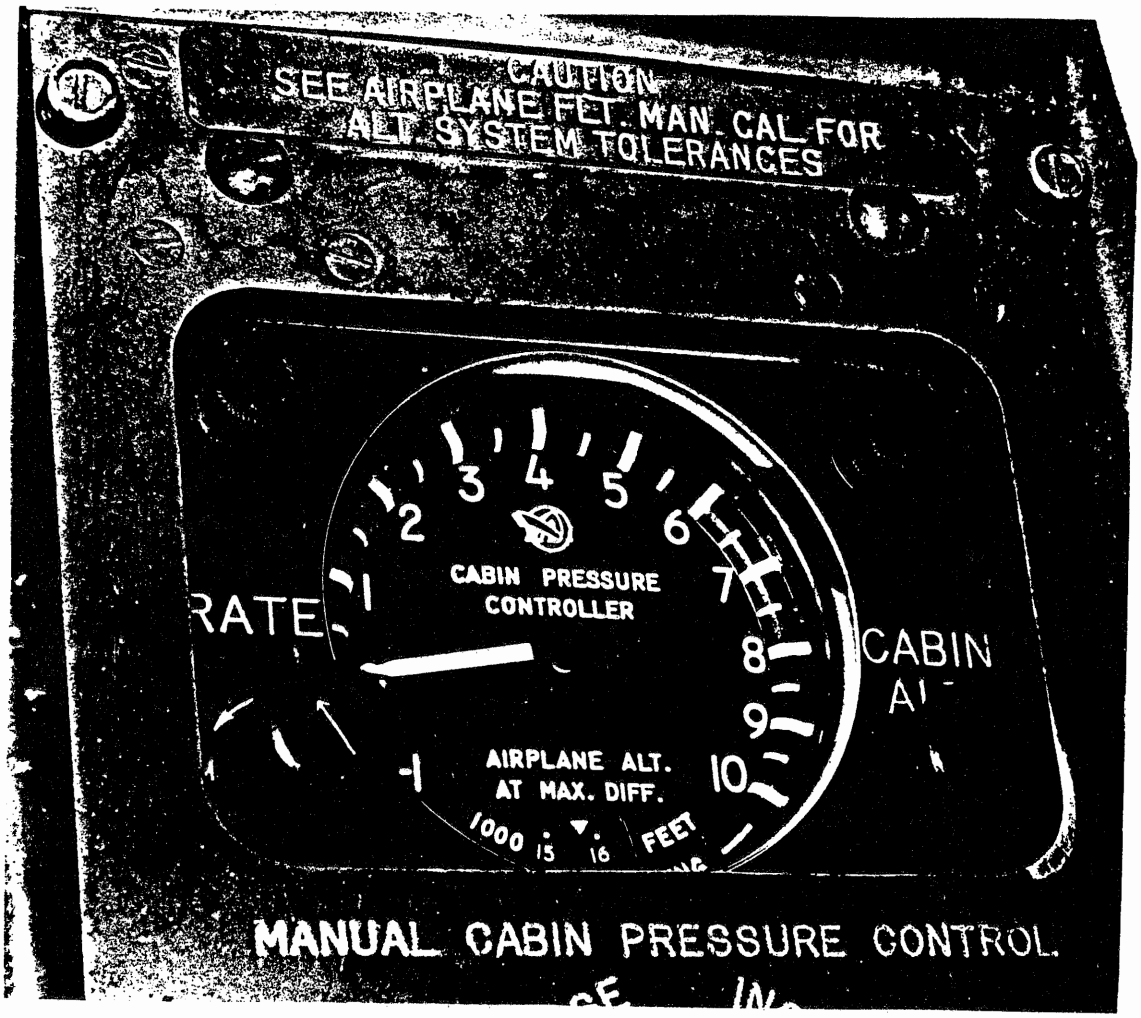
- RAM AIR
- LG WARN LTS
- LG HORN

NOTE: Aircraft 1 - 93, 114 having ASC 108A, Part 1 and Aircraft 94 - 200, 322 and 323, the NUTCRACKER CONT circuit breaker is pulled (open) in place of the LG HORN and LG WARN LTS circuit breakers.

I. With aircraft resting on main gear, the outflow valves should close when either main gear nutcracker switch is depressed. Check both right and left gear switches.



Pressurization Controller
Figure 201.



Automatic Controller
Figure 202.

MANUAL PRESSURIZATION CONTROLLER — DESCRIPTION / OPERATION

3.General

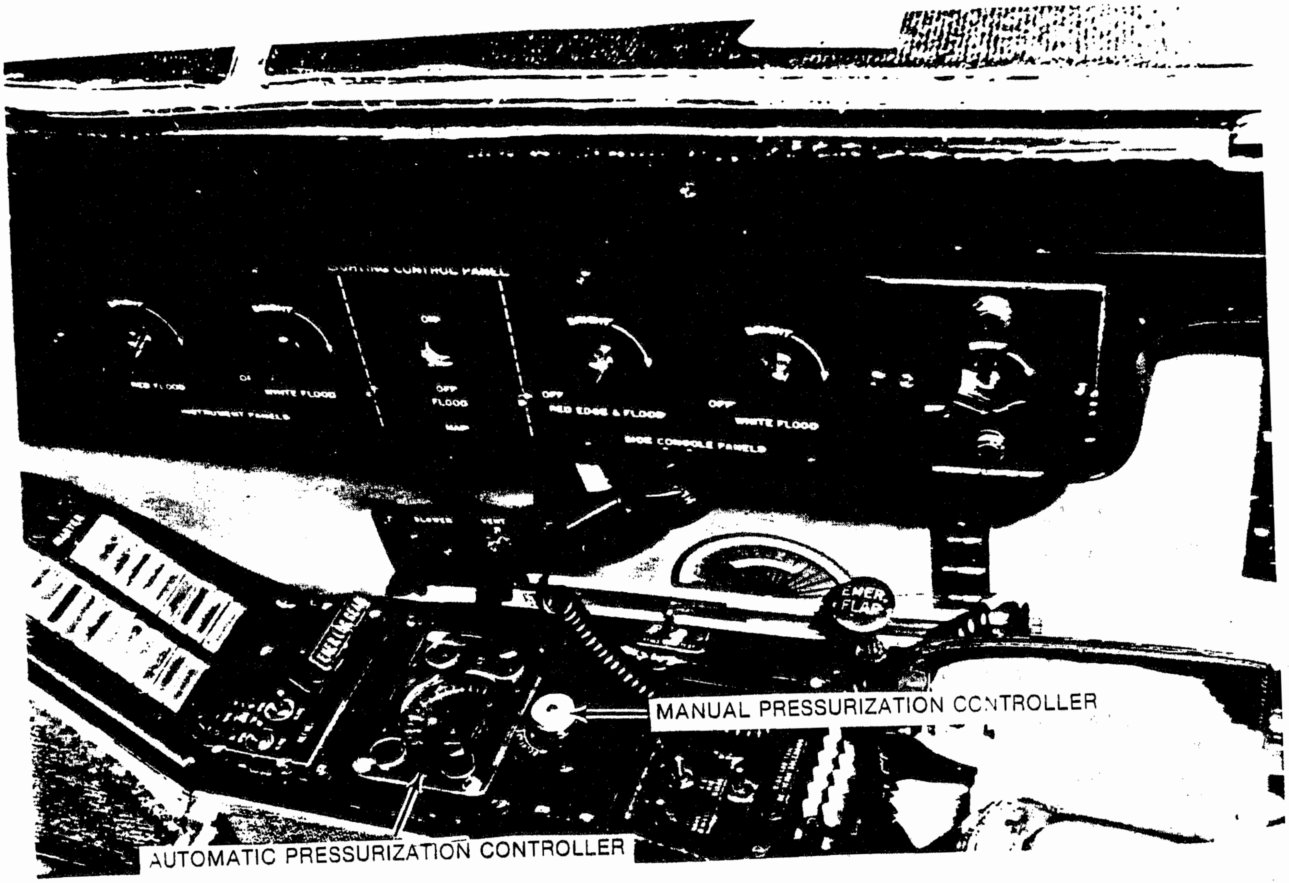
The manual pressurization controller enables the crew to depressurize the aircraft by controlled manual means without resorting to an absolute, instantaneous dump, which would be the case if the pressurization dump solenoid were used. The valve consists of a housing within an inlet connection and an outlet connection, an adjustment knob, and a valve shaft. The valve shaft is turned by the knob to seat the needle valve. The needle valve is spring loaded inside the bottom end of the shaft. As the valve is opened, vacuum from the pressure control venturi is applied at the inlet connection side of the valve housing and flows through the valve seat opening and out the opposite connection. The valve is operated by turning the knob on the top of the valve counterclockwise to open the valve. To open the valve fully from the closed position requires approximately seven complete turns. A pointer on the top of the knob gives an indication of internal valve position. The pointer is 90 to 105 degrees counterclockwise from the inlet connection when the air begins to flow through the unit. Clockwise rotation of the valve shaft after the valve is seated merely compresses the spring in the end of the valve shaft, therefore it is considered a non-jamming type valve. When opened, vacuum is ported through the valve to the primary control pressure line to the pneumatic relays, thus manually controlling the outflow valves. It bypasses the automatic controller, and since it is always kept in the closed position, unless used for depressurizing, it can only reduce differential.

Completely open, the pressurization is fully dumped.

1. **Operation**

CAUTION: KEEP THE MANUAL CONTROLLER IN THE FULL CLOSED POSITION (FULL CW) AT ALL TIMES UNLESS THE AUTOMATIC CONTROLLER IS TO BE BYPASSED TO DECREASE PRESSURIZATION.

The manual cabin pressure controller is a small needle valve installed in the copilots console aft of the automatic controller. (See Figure 1) It bypasses the pressure control venturi vacuum around the automatic controller to the pneumatic relays for the manual control of the system whenever the automatic controller fails. When fully open, the needle valve depressurizes the aircraft. This control can only decrease the cabin pressure being maintained by the automatic cabin pressure controller.



Copilot Console Pressure Control System Controls
Figure 1.

MANUAL PRESSURIZATION CONTROLLER — MAINTENANCE PRACTICES

1. **Manual Pressurization Controller — Removal / Installation**

CAUTION: BEFORE PERFORMING THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

NOTE: Aircraft 1 - 78, 114 having ASC 105 and Aircraft 79 - 200, 322 and 323, the pressurization control is shock mounted and flexible lines are provided in lieu of rigid lines. On aircraft not having ASC 105 disregard reference made to shock mounted panel.

(1) Gain access to manual controller by removing panel directly aft of controller panel.

(2) If necessary disconnect deicing panel plug and remove panel.

(3) Disconnect short rigid tubing between manual controller to fitting and outflow valve port on automatic controller.

(4) On shock mounted panel remove screws securing controller panel to console rail.

(5) Support controller and remove shock mount screws.

(6) Remove screws securing manual controller to panel and remove controller.

(7) Install protective caps on all lines and ports.

B. Installation

(1) Remove all protective caps.

(2) Install manual controller on panel and secure with screws.

(3) On shock mounted panel, install controller panel on shock mounted panel, and secure with screws.

(4) Connect short rigid tubing to manual controller.

(5) Secure panel to console rail with fasteners.

(6) Install panel and connect deicing panel plug if removed.

(7) Inspect area for foreign objects, security of all attachments.

(8) Install access panel previously removed.

(9) Perform Manual Cabin Pressure Control — Operational Test, this Section.

2. **Manual Cabin Pressure Control — Operational Test**

A. Apply an external source of 28V, dc power and energize main dc bus and essential dc bus (BATT SW - NORM, EXT PWR SW - ON).

B. Ensure a reading of 3 inHg or more is indicated on the de-icer SUCTION gage (vacuum manifold), by performing any one of the following procedures:

(1) Connect source of air pressure to pneumatic deicing system in nacelle.

WARNING: TO PREVENT SERIOUS PERSONAL INJURY AND AIRCRAFT DAMAGE, THE MAIN DOOR OR THE DIRECT VISION WINDOWS MUST BE OPENED IF THE RIGHT ENGINE OR APU IS USED.

(2) Operate either of the main engines.

(3) Operate APU with full air output to cabin and set cabin and cockpit temperature controls for full cooling.

C. Pull (open) RAM AIR and CBN PRESS UNL circuit breakers. (Pilots circuit breaker panel.)

D. Turn manual cabin pressure control needle valve on right console through its full travel in the INCREASE (clockwise) direction.

E. Force air through venturi in outflow valve compartment.

F. Turn the CABIN ALT. knob on the automatic cabin pressure controller on right console to adjust needle of the controller to a cabin altitude of 1000 feet below the actual altitude of the test site. The cabin outflow valves, located below the radio rack just aft of Fuselage Station 133, should close fully.

G. Turn manual cabin pressure control needle valve in the DECREASE (counterclockwise) direction. The cabin outflow valves should open. Turn the needle valve in the INCREASE (clockwise) direction. The cabin outflow valves should close.

3. Manual Cabin Pressure Control Dump — Operational Test

A. Apply an external source of 28V, dc power and energize main dc bus and essential dc bus (BATT SW - NORM, EXT PWR SW - ON).

B. Ensure a reading of 3 inHg or more is indicated on the de-icer SUCTION gage (vacuum manifold), by performing any one of the following procedures:

(1) Connect source of air pressure to pneumatic deicing system in nacelle.

WARNING: TO PREVENT SERIOUS PERSONAL INJURY AND AIRCRAFT DAMAGE, THE MAIN DOOR OR THE DIRECT VISION WINDOWS MUST BE OPENED IF THE RIGHT ENGINE OR APU IS USED.

(2) Operate either of the main engines.

(3) Operate APU with full air output to cabin and set cabin and cockpit temperature controls for full cooling.

C. Depress (close) RAM AIR circuit breaker.

D. Pull (open) the following circuit breakers: (Pilots circuit breaker panel)

- LG HORN
- LG WARN LTS
- CBN PRESS UNL

NOTE: Aircraft 1 - 93, 114 having ASC 108A, Part 1 and Aircraft 94 - 200, 322 and 323, 114, the NUTCRACKER CONT circuit breaker is pulled (open) in place of the LG HORN and LG WARN LTS circuit breakers.

E. Set AIR COND - VENT switch on right console to AIR COND.

F. Turn manual cabin pressure control needle valve on right console through its full travel in the INCREASE (clockwise) direction.

G. Force air through venturi in outflow valve compartment.

H. Turn the CABIN ALT. knob on the automatic cabin pressure controller on right console to adjust needle of the controller to a cabin altitude of 1000 feet below the actual altitude of the test site. The cabin outflow valves, located below the radio rack just aft of Fuselage Station 133, should close fully.

I. Set AIR COND - VENT switch to VENT. Cabin outflow valves should open fully.

J. Set AIR COND - VENT switch to AIR COND. Cabin outflow valves should close fully.

4. Cabin Pressurization Controller Filter — Replacement

A. Gain access to filter by removing copilots seat and console kick plate below pressurization controller.

B. Locate filter assembly.

C. Roll rubber retainer away from filter assembly; remove and discard filter element.

D. Install new filter element in filter assembly.

E. Roll rubber retainer over filter assembly.

F. Install console kick plate and copilots seat.

PRESSURIZATION DUMP SOLENOID VALVES — DESCRIPTION / OPERATION

1. General

The pressurization dump solenoid valves, located near the outflow valves and pneumatic relays, are electrically operated, normally closed, ball type solenoid valves. When energized, they depressurize the cabin by routing the deicing system vacuum manifold directly to the outflow valves opening them fully. All pressurization is dumped immediately. (This can be extremely uncomfortable, and in some cases dangerous if ambient altitude is high, and oxygen masks are not used.)

Placing the AIR COND-VENT switch on the copilots console in the VENT position, or by weight on both main gears through the nutcracker system will energize the pressurization dump solenoids, thereby, causing depressurization of the cabin.

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PRESSURIZATION DUMP SOLENOID VALVES — MAINTENANCE PRACTICES

1. Pressurization Dump Solenoid Valves — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Gain access to solenoid valves, adjacent to outflow valves between Fuselage Stations 139 and 150 under radio rack.

(3) Disconnect electrical connector.

(4) Disconnect rigid tubing from valve, remove valve.

(5) Install protective caps and electrical connector, receptacle, fittings and parts.

B. Installation

(1) Remove protective caps.

(2) Install valve by connecting rigid tubing.

(3) Connect electrical connector.

(4) Energize main and essential dc buses.

CAUTION: IF APU AIR IS USED, ENSURE DV WINDOW OR A DOOR IS OPEN TO PREVENT PRESSURIZATION.

(5) Obtain three inches Hg vacuum in deicing system vacuum manifold by operating APU with air on or connecting pressurized air to pneumatic system in nacelle; cabin outflow valves should open.

NOTE: On Aircraft 1 - 93 and 114, having ASC 108A Part 1, and Aircraft 94 - 200, NUTCRACKER CONT circuit breaker should be pulled in place of LG WARN HORN and LG WARN LTS circuit breakers.

(6) Pull either LG WARN HORN and LG WARN LTS circuit breaker or NUTCRACKER circuit breaker; cabin outflow valves should close. Depress circuit breaker(s) pulled in Step 6; cabin outflow valves should open. Turn off and disconnect air (or shutdown APU); deenergize main and essential dc buses; return system to normal configuration.

(7) Inspect area for presence of foreign objects then close access panel.

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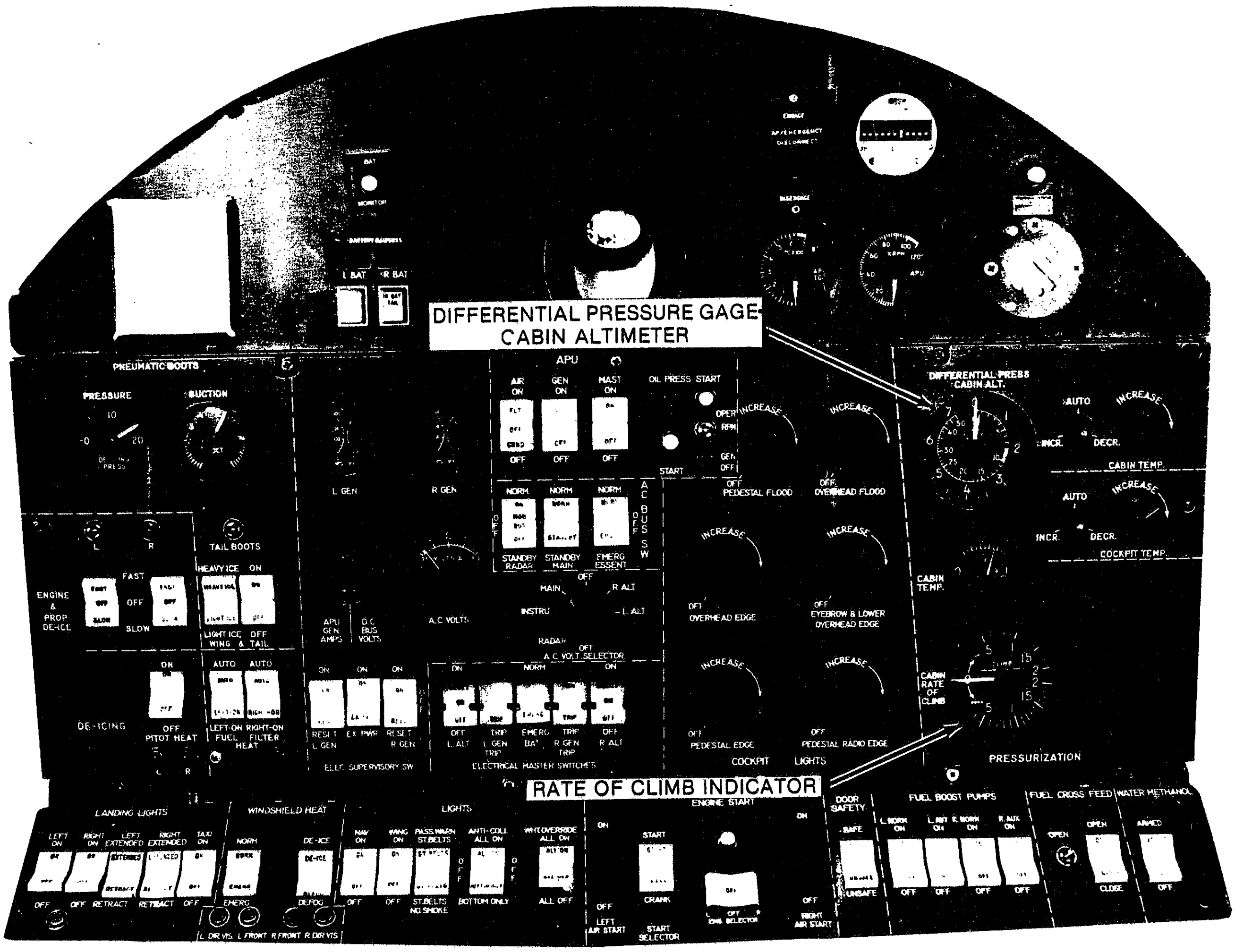
CABIN ALTIMETER / DEFERENTIAL PRESSURE GAGE — DESCRIPTION / OPERATION

1. **General**

   A dual indicator is installed in the upper overhead panel, directly adjacent to the air conditioning control switches. (See Figure 1) One needle indicates cabin altitude and the other indicates cabin to ambient pressure differential as a reference for the effectiveness of the pressurization system.



Upper Overhead Panel Pressure Control System Controls
Figure 1.

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CABIN ALTIMETER / DIFFERENTIAL PRESSURE GAGE — MAINTENANCE PRACTICES

1. **Cabin Altimeter / Differential Pressure Gage — Removal / Installation**

A. Removal

(1) Remove all power from aircraft.

CAUTION: PULL EDGE LIGHTING PANEL STRAIGHT OUT REMOVING TO AVOID BENDING.

(2) Remove edge lighting panel from left side of overhead panel by removing knobs from cabin and cockpit temperature control rheostats and screws that hold lighting panel to main panel.

(3) Gain access to pressure gage by lowering overhead panel; support panel while lowering.

(4) Disconnect tubing from gage; install protective caps on fittings and ports.

(5) Remove gage (gage is back mounted).

B. Installation

NOTE: If a replacement gage is to be installed, note position of transfer fitting to new gage.

(1) Remove protective caps from fittings and ports.

(2) Install gage.

(3) Connect tubing to gage.

(4) close overhead panel and secure.

(5) install edge lighting panel and secure. Install cabin and cockpit temperature control rheostat knobs.

(6) Energize main dc bus. Rotate OVERHEAD EDGE rheostat and check that edge lighting panel lights come on; rotate rheostat off.

(7) De-energize main dc bus.

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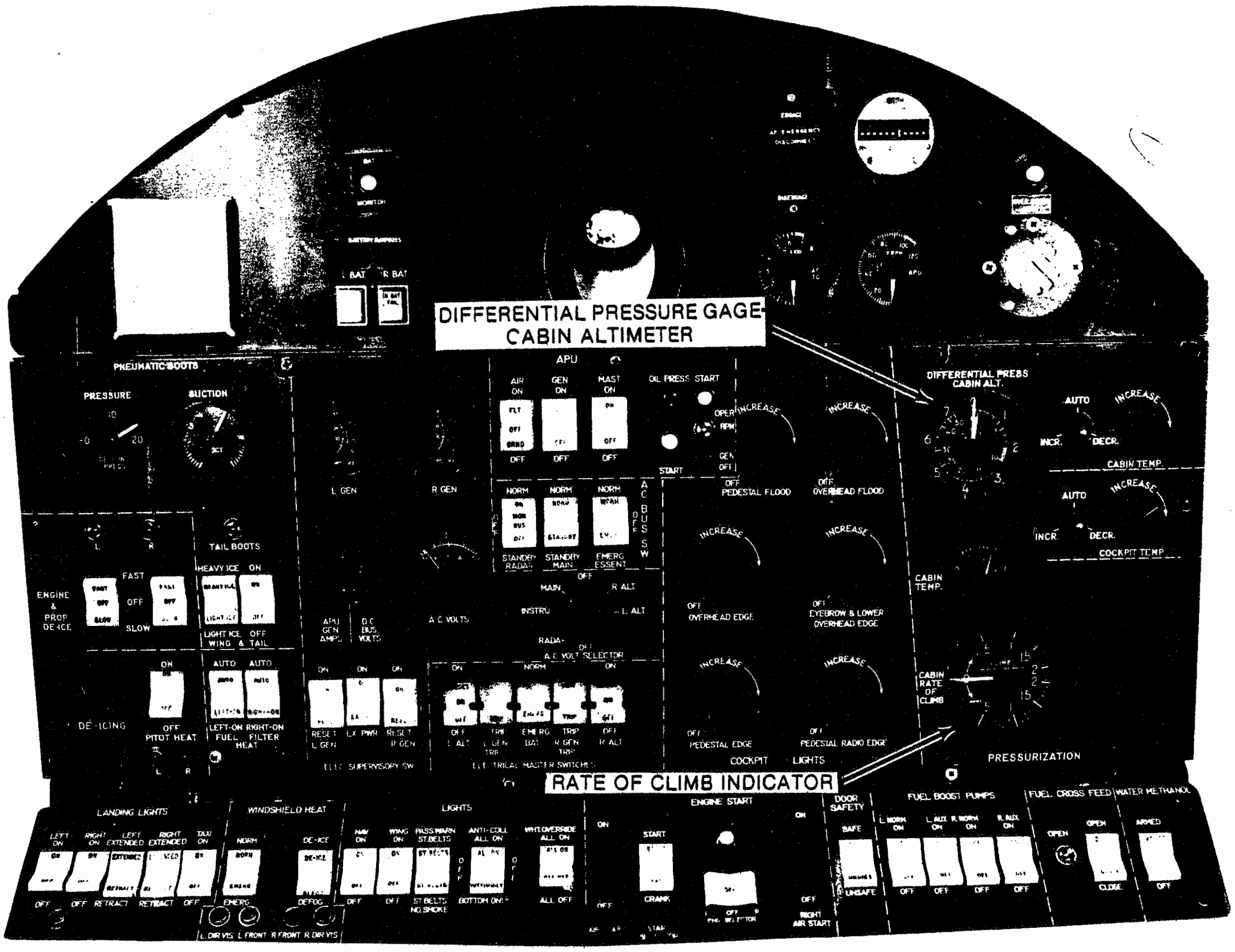
CABIN RATE OF CLIMB INDICATOR — DESCRIPTION / OPERATION

1. **General**

   Installed in the upper overhead panel under the differential pressure gage cabin altimeter (See Figure 1) is a standard lagged type rate of climb indicator vented to the cabin. This instrument gives a visual indication in thousands of feet of the rate of change in cabin pressurization within the aircraft.



Upper Overhead Panel Pressure Control System Controls
Figure 1.

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CABIN RATE OF CLIMB INDICATOR — MAINTENANCE PRACTICES

1. **Cabin Rate of Climb Indicator — Removal / Installation**

   A. Removal

   CAUTION: PULL EDGE LIGHTING PANEL STRAIGHT OUT WHEN REMOVING TO AVOID BENDING.

   (1) Remove all power from aircraft.

   (2) Remove edge lighting panel from left of overhead panel by removing knobs from cabin and cockpit temperature control rheostats and screws that hold lighting panel to main panel.

   (3) Lower overhead panel; support panel while lowering.

   (4) Remove indicator (indicator is back mounted).

   B. Installation

   NOTE: If replacement indicator is to be installed, ensure port at rear of indicator is open and clear.

   (1) Install indicator on panel.

   (2) Close overhead panel and secure.

   (3) Install edge lighting panel and secure. Install cabin and cockpit temperature control rheostat knobs.

   (4) Connect 28V dc power to aircraft.

   (5) Energize main dc bus; rotate OVERHEAD EDGE rheostat and check that edge lighting panel lights come on; rotate rheostat to off.

   (6) De-energize main dc bus

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CABIN PRESSURE WARNING SYSTEM — DESCRIPTION / OPERATION

1. **General**

An aneroid type pressure switch is located at the top of the hydraulic reservoir compartment. This switch is normally open if the cabin altitude is below 9250 $\pm$ 750 feet. If the pressurization fails to maintain the cabin below this altitude, the switch is actuated and a CABIN PRESS light is illuminated on the master warning system, indicating a possible pressurization failure. The switch is set so that it will open on increasing pressure before 8000 feet cabin altitude is reached.

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CABIN PRESSURE WARNING SYSTEM — MAINTENANCE PRACTICES

1. **Cabin Pressure Warning — Functional Test**

A. Steps below apply to Aircraft 1 - 118, not having ASC 259.

(1) Pull WARNING LIGHTS circuit breaker on left circuit breaker panel.

(2) Disconnect electrical connector from aneroid switch.

(3) Connect meter/test lamp across pins B and C of electrical connector.

(4) Reset WARNING LIGHTS circuit breaker, power should be indicated.

(5) Pull WARNING LIGHTS circuit breaker. Disconnect meter/test lamp.

(6) Connect electrical connector to cabin pressure warning aneroid switch. Reset circuit breaker.

(7) Flight check aircraft unpressurized to minimum 10,000 feet pressure altitude.

(8) Aneroid switch closes on decreasing pressure and should actuate, giving warning at 9250 ± 750 feet.

B. Steps below apply to Aircraft 1 - 118, having ASC 259 and Aircraft 119 - 200.

CAUTION: TO AVOID DEPLETING THE BATTERIES PERSONNEL INVOLVED IN THIS FUNCTIONAL TEST SHOULD BE FAMILIAR WITH THE PROCEDURE TO ENSURE A MINIMUM AMOUNT OF TIME IS USED.

(1) Remove pneumatic fitting from cabin pressure sensing port and connect a vacuum source using an AN815-40 fitting.

(2) Connect aircraft batteries and place battery switch to EMER.

(3) Apply vacuum (slowly) to cabin pressure sensing port. CABIN PRESS light should come on at 9250 ± 750 feet altitude.

(4) Reduce vacuum. CABIN PRESS light should go off before 8000 feet altitude is reached.

(5) Reduce vacuum (slowly) and disconnect vacuum source. Place battery switch to OFF.

2. **Cabin Pressure Control Safety Valve — Operational Test**

A. Remove cabin altimeter pressure warning switch, and all other pressure sensitive devices. Interconnect static hoses normally connected to these units.

CAUTION: ENSURE THERE ARE NO LEAKS IN FLEX HOSE BEFORE CONNECTING TO TEST FITTING IN NOSE WHEEL WELL.

(1) Connect flex hose with pressure gage (0 - 10 psi) or manometer (0 - 20 inHg) to test fitting in nose wheel well located on left side Fuselage Station 85.

(2) Open the following circuit breakers: (Pilots circuit breaker panel.)

- RAM AIR
- CBN PRESS UNL

(3) Depress (close) the CBN/CKP TEMP circuit breaker. (Pilots circuit breaker panel)

(4) Turn manual pressure control knob to full INCREASE position.

(5) Turn DIFFERENTIAL and SAFETY test valve under copilots console to test position (handle up).

(6) Turn safety test valve under copilots console to TEST position (handle up).

(7) Remove line from aft pneumatic relay to aft outflow valve.

(8) Cap fitting on relay. Leave fitting on valve open.

B. When using APU as pressure source, use following Steps.

(1) Connect flex hose with needle valve (valve, when operated must be capable of completely unloading APU) to bulkhead fitting APU container (upper forward right side), and open needle valve.

(2) Hold cabin and cockpit temperature toggle switches in DECREASE position for 40 seconds.

(3) Start APU and set AIR switch to ON or FLIGHT if having CB124.

(4) Close all windows, evacuate aircraft and close doors.

(5) Slowly close down needle valve (raise cabin pressure slowly).

(6) Cabin pressure must rise and relieve between 7.00 and 7.40 psi, or 14.25 and 15.07 inHg.

(7) Depressurize aircraft by opening needle valve.

WARNING: WAIT UNTIL GAGE OR MANOMETER READS ZERO BEFORE OPENING DOOR.

(8) Connect lines removed to aft outflow valve.

C. When using SHOP AIR as pressure source, use following Steps.

(1) Depress (close) RAM AIR circuit breaker. Set AIR COND - VENT switch to AIR COND position. After 10 seconds, pull (open) RAM AIR circuit breaker (dc power must be supplied in this step).

(2) Connect shop air supply rig, equipped with shutoff valve, to air conditioning ducting on right side of aircraft at rear wing beam.

NOTE: Air supply must be capable of delivering air at the rate of 20 pounds per minute.

(3) Connect source of air pressure to pneumatic deicing system ground test connection in right nacelle. Deicing system PRESSURE gage should read 15 - 18 psi.

(4) Close all windows, evacuate aircraft and close doors.

(5) Open shutoff valve in air supply line, allowing aircraft to pressurize slowly.

(6) Cabin pressure must rise and relieve between 7.00 and 7.40 psi, or 14.25 and 15.07 inHg.

(7) Depressurize aircraft by opening needle valve (if APU is used) or closing shop air supply valve (if shop air is used).

WARNING: WAIT UNTIL GAGE OR MANOMETER READS ZERO BEFORE OPENING DOOR.

(8) Connect lines removed to aft outflow valve.

(9) Remove lines from forward pneumatic relay to forward outflow valve.

(10) Cap fitting on relay. Leave fitting on valve open.

D. Following either procedure above: (B or C)

(1) Cabin pressure should rise and stabilize between 7.00 and 7.40 psi, or 14.25 and 15.07 inHg.

(2) Depressurize aircraft by opening needle valve (if APU is used) or closing shop air supply valve (if shop air is used).

(3) Connect lines to forward outflow valve.

(4) Return differential and safety test, and safety test valves under copilots console to NORMAL and safety wire in this position.

(5) Reinstall cabin altimeter pressure warning switch, and all other pressure sensitive devices. Interconnect static hoses normally connected to these units.

(6) Return system to normal configuration depending on whether APU or shop air was used.

(7) Ensure that tight connections are made on all lines disconnected to perform test.

3. Cabin Pressure Aneroid Switch — Removal / Installation

CAUTION: BEFORE PERFORMING THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to cabin pressure aneroid switch (located in hydraulic compartment, above hydraulic reservoir).

(2) Disconnect electrical connector; remove hardware securing switch to bracket; remove switch.

(3) Install protective caps on electrical connector and switch receptacle.

B. Installation

NOTE: If aneroid pressure switch is to be replaced by a like unit remove the modified AN fitting and install it on the replacement switch prior to installation.

(1) Remove protective caps from electrical connector and switch receptacle; remove cap from AN fitting (switch).

(2) Install switch on bracket using hardware previously removed; connect electrical connector.

(3) Inspect area for foreign objects, security of all attachments.

(4) Close access previously opened.

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PNEUMATIC RELAYS — DESCRIPTION / OPERATION

1. General

Each outflow valve is controlled by a pneumatic signal from the controller. This signal is matched by two pneumatic relays located adjacent to the outflow valves which by using the pneumatic deicing system vacuum can produce a high flow from and thus rapid response of the outflow valves. One relay serves each outflow valve but both receive the same signal from the controller.

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PNEUMATIC RELAYS — MAINTENANCE PRACTICES

1. **Pneumatic Relays / Forward and Aft — Removal / Installation**

CAUTION: BEFORE PERFORMING THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to pneumatic relays, located adjacent to outflow valves between Fuselage Station 133 and 157 under radio rack.

(2) Disconnect line from relay.

(3) Remove hardware securing relay; remove relay.

(4) Install protective caps on fittings and parts.

B. Installation

NOTE: If a replacement relay is to be installed, note position of fittings and transfer fittings to new relay. Use new gaskets as required.

(1) Remove protective caps from fittings and parts.

(2) Install relay using hardware previously removed.

(3) Connect line to relay.

(4) Perform Pneumatic Relay Forward/Aft — Operational Test, this Section.

2. **Pneumatic Relays / Forward / Aft — Operational Test**

A. Energize main and essential dc buses.

B. Connect pressurized air to deicing system in nacelle.

C. Obtain 3 inHg vacuum in pneumatic deicing system vacuum manifold.

D. Pull RAM AIR and CABIN PRESSURE UNL circuit breakers.

E. Turn manual cabin pressure control needle valve on right console to full increase (clockwise) position.

F. Force air through venturi in outflow valve compartment.

G. Turn cabin altitude knob automatic controller to adjust needle to a cabin altitude of 1000 feet BELOW actual altitude of test site: cabin outflow valves should close.

H. Turn manual cabin pressure control needle valve to decrease (counterclockwise) direction; cabin outflow valves should open.

I. Turn manual cabin pressure control needle valve to increase (clockwise) direction; cabin outflow valves should close.

J. Turn cabin altitude knob on automatic controller to adjust needle to a cabin altitude of 1000 feet ABOVE pressure altitude of test site; cabin outflow valves should open fully.

K. Turn off and disconnect air; de-energize main and essential dc buses return system to its normal configuration.

L. Inspect area for foreign objects, security of all attachments.

M. Close access previously opened.

3. **Pneumatic Relay Metering Valve — Leakage Test**

CAUTION: IN THE EVENT THAT THERE IS ANY DOUBT AS TO THE SATISFACTORY CONDITION OF THE PNEUMATIC RELAY IT IS SUGGESTED THAT THE UNIT NOT BE PLACED IN SERVICE UNTIL FURTHER INFORMATION IS OBTAINED FROM AN AIRESEARCH REPRESENTATIVE OR FROM GULFSTREAM AEROSPACE. THIS TEST IS VALID ONLY FOR THE METERING VALVE PORTION OF THIS UNIT. FOR FURTHER TESTS REFER TO APPLICABLE PUBLICATION.

NOTE: The following test will ascertain whether the pneumatic relay metering valve is properly seated or has been worn beyond acceptable limits. It involves a source of vacuum, a suitable manometer and shutoff valves as shown in diagram of test setup. (See Figure 201)

A. Remove pneumatic relay from the aircraft.

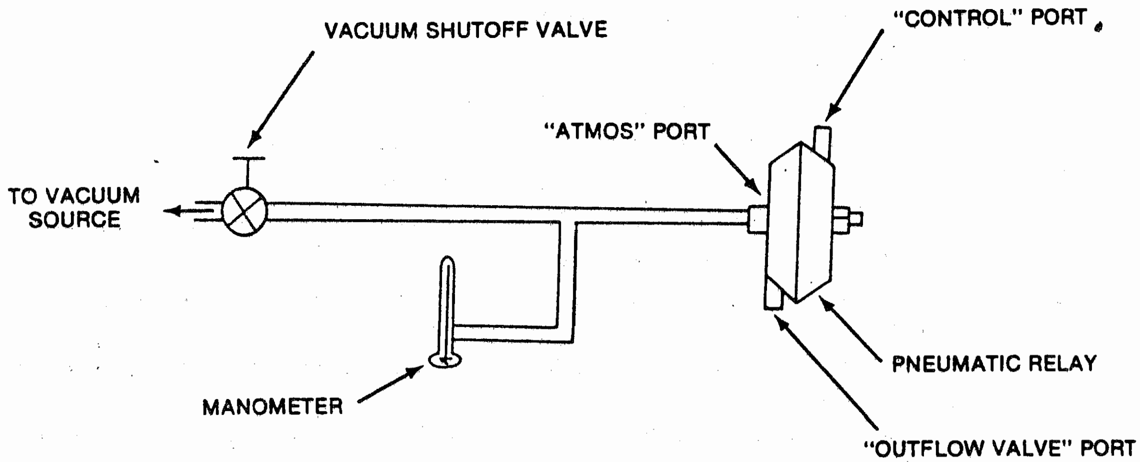
B. Attach source of vacuum to the pneumatic relay port marked "ATMOS". Remaining ports of pneumatic relay marked "CONTROL" and "OUTFLOW VALVE" must be vented to ambient.

C. Open vacuum valve and apply vacuum until measuring device indicates 10.0 $\pm$ 0.25 in Hg. Then close vacuum valve.

D. Open vacuum measuring device and note pressure increase in 1 minute. Pressure increase must not exceed 0.6 inHg.

E. If pressure increase exceeds above amount the leakage is excessive.

F. If test is satisfactory release vacuum.



Pneumatic Relay Metering Pin Test Setup
Figure 201.

OUTFLOW VALVES — DESCRIPTION / OPERATION

1. General

Two outflow valves, installed aft of the copilots bulkhead from Fuselage Station 133 to approximately Fuselage Station 157, are mounted on the right skin of the aircraft under the radio rack. The valves exhaust overboard. Both valves are identical and operate in parallel for safety reasons.They are pneumatically-operated diaphragm devices which are controlled by a pneumatic signal from the automatic controller, manual controller, or a dump signal from the pressurization dump solenoid valves. The outflow valves will allow the cabin air to exhaust from the pressurized area, or will close down, restricting outflow air to maintain pressurization and allow outflow air to exhaust at specific rates.

2. Detailed

The outflow valve have the function of creating an outflow air path, safety pressure relief, and vacuum differential control of the pressurization system. The base of the outflow valve, which is a magnesium casting, contains openings through which air flows between the cabin and the atmosphere. The flow of air is controlled by a large flow valve which seats against the base. This valve is secured to a coverplate which slides on a teflon shaft when the valve moves. Air from the cabin can flow between the valve and the coverplate. The control diaphragm functions, in effect, as two separate diaphragms.

The portion of the diaphragm between the valve and the base operates on a differential pressure between the cabin and the control chamber. The other portion of the diaphragm between the base and coverplate operates on a differential pressure between the cabin and the atmosphere. This control diaphragm, and a small pressure relief diaphragm cover and cover assembly, form two chambers; a cabin pressure chamber and a control chamber. The pressure relief diaphragm separates the cabin pressure chamber and a recess in the base, thus forming a third chamber, the atmosphere sensing chamber.

The control chamber of the valve contains a spring which loads the valve towards the closed position. This is the large spring directly beneath the port from the pneumatic relay. The control chamber is vented to the cabin through a filter. The cabin pressure chamber contains a pilot shaft. This shaft guides the cover which is attached to the flow valve by means of the teflon shaft and contains an air passage for venting the control chamber to the atmosphere through the atmosphere sensing chamber. This passage which is used in conjunction with the safety pressure relief mechanism, is operated by a ball type pressure relief valve. This relief valve is connected to the safety pressure relief diaphragm. The atmosphere sensing chamber contains a spring which loads the safety pressure relief diaphragm holding the ball valve normally closed. The spring is the very small one in the bottom center of the valve. Note there are actually three connections to the outflow valve: one from the pneumatic relay into the control chamber; one from the filter, which allows cabin air to enter through the small orifice into the control chamber; and a third in the bottom, which allows static air pressure to enter the atmosphere sensing chamber. The outflow valve is mounted on the pressurized side of the cabin. The cabin air flows through the orifice in the filtered cabin air inlet port to the control chamber. This flow results in a pressure in the control chamber; this is cabin pressure minus whatever pressure loss occurs across the orifice. Cabin pressure exists in the cabin air pressure chamber through the small holes in the circumference of the flow valve. Atmosphere pressure exists in the atmosphere sensing chamber through the static port at the bottom.

In normal operation, vacuum from the pneumatic relays, controlled by the automatic or manual controller signal pressure, is ported into the control chamber. This creates a low pressure area in the outflow valve control chamber which is directly proportional to the controller demands. The cabin air pressure in the cabin air pressure chamber on the other side of the control diaphragm thus is greater than the reduced pressure in the control chamber causing the diaphragm to move the flow valve toward open. The amount will depend on the amount of vacuum applied to the control chamber. The more vacuum, the greater the opening. The less the restriction, the less pressurization is attained.

Safety pressure relief occurs when the pressure differential between the cabin pressure and the atmospheric pressure is sufficient to cause the safety pressure relief diaphragm to move against the force of its backing spring. Movement of the diaphragm unseats the ball valve. When this occurs, air escapes from the control chamber to the atmosphere, reducing the pressure in the control chamber. Cabin pressure exerted against

control diaphragm from within the cabin pressure chamber forces the control diaphragm to open. This allows cabin air to flow to the ambient air, maintaining the cabin differential pressure at a safety limit of 7.2 $\pm$ 0.2 psi in case the normal maximum differential pressure control (6.55 psi) in the controller fails.

Vacuum relief occurs whenever ambient pressure exceeds cabin pressure by 0.20 psi. When this condition exists, pressure on the atmosphere side of the vacuum relief section of the control diaphragm forces the diaphragm against the coverplate and lifts the flow valve off of its seat. This allows air at atmospheric pressure to flow into the cabin.

Dump operation occurs whenever the cabin pressurization dump solenoid valves are energized electrically. This permits vacuum to decrease pressure in the outflow valve control chamber, permitting cabin pressure to push the valve open.

The solenoid dump valves can be energized to dump the system by either of the following means:

A. Weight on main landing gear through nutcracker system.

B. Placing AIR COND - VENT switch on copilots console in VENT position. (This also accomplishes other functions in air conditioning system.)

In-flight controlled depressurization can be accomplished by opening the manual cabin pressure control valve on the copilots console. This permits venturi vacuum to decrease pressure in the control chamber of the controller and in the signal pressure line to the pneumatic relays. The pneumatic relays are displaced, compressing the springs and opening the needle valve on the output side of the relays. As a result deicing system vacuum decreases pressure in the outflow valve control chamber and cabin pressure opens the outflow valves.

CAUTION: IT IS TO BE NOTED THAT THE OUTFLOW VALVE PORTS ARE NOT MARKED ON UNIT, THEREFORE THE FOLLOWING MUST BE OBSERVED.

CENTER PORT OF THE TWO ON THE DOME (CABIN) SIDE OF VALVE ALWAYS GOES TO IT'S PNEUMATIC RELAY AND PRESSURIZATION DUMP SOLENOID.

THE PORT OFF CENTER ON THE DOME (CABIN) SIDE OF RELAY ALWAYS GOES TO ITS FILTER.

THE SINGLE PORT ON THE AMBIENT (OUTFLOW) SIDE OF VALVE MUST BE OPEN TO ALLOW STATIC AIR TO ENTER. IF CAPPED, ENSURE CAP IS REMOVED BEFORE INSTALLING VALVE IN AIRCRAFT. NO TUBING CONNECTION IS MADE. PORT IS LEFT OPEN.

OUTFLOW VALVES — MAINTENANCE PRACTICES

1. Outflow Valve — Removal / Installation

CAUTION: BEFORE PERFORMING FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to outflow valves located between Fuselage Stations 133 and 157 under radio rack.

(2) Disconnect rigid tubing from outflow valve elbow fitting.

(3) Release V-band clamp securing valve to mounting base.

(4) Remove outflow valve and gasket. Inspect and replace as required.

(5) Install protective caps on all lines and parts.

B. Installation

CAUTION: ENSURE SAFETY PRESSURE RELIEF PORT IN THE THROAT OF THE OUTFLOW VALVE IS UNCAPPED AND OPEN.

NOTE: If a replacement valve is to be installed, transfer filter and fittings to new valve. Use new gaskets under jam nuts are required. Orient fittings in same direction as old valve.

At the operators discretion replacement of the outflow valve filter cartridge(s) can be performed at this time.

(1) Remove protective caps.

(2) Install gasket and orient outflow valve properly on mount.

(3) Tighten V-band clamp securely.

(4) Connect rigid tubing to elbow fitting.

(5) Energize main and essential dc buses.

(6) Connect pressurized air to deicing system in nacelle.

(7) Obtain 3 inHg vacuum in pneumatic deicing system vacuum manifold.

(8) Pull following circuit breakers: LG HORN LG WARN LTS, and CBN PRESS UNL.

NOTE: Aircraft 1 - 93, 114 having ASC 108A, part 1 and Aircraft 94 - 200, 322 and 323, the NUTCRACKER CONT circuit breaker should be pulled (open) in place of the LG HORN and LG WARN LTS circuit breakers.

(9) Depress RAM AIR circuit breaker.

(10) On cabin pressure controller in cockpit, select MAX rate and set cabin altitude pointer to a setting of 1000 feet below pressure altitude to test site.

(11) Turn manual cabin pressure control needle valve on right console to full increase (clockwise) position.

(12) Place AIR COND-VENT switch to VENT. Outflow valves should open.

(13) Place AIR COND-VENT switch to AIR.COND. Outflow valve should close.

(14) Depress LG HORN circuit breaker (NUTCRACKER CONT circuit breaker on Aircraft 1 - 93, 114 having ASC 108A, part 1 and 94 - 200, 322 and 323), outflow valve should open.

(15) Depress right nutcracker switch. Outflow valves should close.

(16) Release right nutcracker switch. Outflow valves should open.

(17) Turn off and disconnect air; de-energize main and essential dc bus; return system to its normal configuration.

2. **Outflow Valve — Inspection**

A. Clean outflow valves, which are located between Fuselage Station 139 and 150 on right side of fuselage under radio racks, as required. In addition, outflow valve filters should be checked and changed in accordance with Chapter 5.

B. The following method is suggested for cleaning outflow valves:

(1) Inspect for condition and presence of foreign objects.

(2) Place a vacuum hose against outflow valve (filter fittings) on right side of fuselage to lift off its seat. This same operation will remove accumulated dust and dirt.

(3) Wipe off and clean valve seats with mineral spirits conforming to Specification TT-T.291.

(4) When one valve has been cleaned satisfactory, clean other valve in same manner.

3. **Outflow Valve Filter Forward / Aft — Service / Replacement**

A. Gain access to outflow valve filters, located on each outflow valve between Fuselage Stations 139 and under radio rack.

B. Roll rubber retainer away from filter assembly; remove and discard filter cartridge.

C. Install new filter cartridge in filter assembly.

D. Roll rubber retainer over filter assembly.

E. Inspect for presence of foreign objects then close access panel.

COLLINS RAD[illegible] [illegible]MPANY • CEDAR RAPIDS

COLLINS

NOT REVISED REFERENCE ONLY

Cedar Rapids, Iowa 52406 Area Code 319 395-1000 Cable: COLINRAD

October 1, 1971

REVISION NO 3

AP-103F SERVICE INFORMATION LETTER 14-66

Attached are revised pages of AP-103F Service Information Letter 14-66, titled "Installation of the AP-103F Flight Control System in the Grumman Gulfstream G-159 Aircraft," originally dated 4-6-66, revised 1-6-67 and 4-28-67.

This revision results from an aircraft system requirement to change the 562A-5G roll channel radio trip point resistor R16 from 75 kΩ to 110 kΩ. The value change will appear in paragraph 6.1.J. and in figure 4. Revision bars in the margin show where the changes are made.

Replace pages 1, 2, 3, 4, 9, and 10 in your issue with the attached.

INFORMATION BULLETIN

COLLINS RADIO COMPANY

SUBJECT SERVICE INFORMATION LETTER 14-66 FOR INSTALLATION OF THE AP-103F FLIGHT CONTROL SYSTEM

DATE 4-6-66
Page 1 of 13
Revised: 1-6-67
Revised: 4-28-67
Revised: 10-1-71

FROM COLLINS RADIO COMPANY, PUBLICATIONS ENGINEERING DEPARTMENT, CEDAR RAPIDS, IOWA, USA

AIRCRAFT AFFECTED: Grumman Gulfstream G-159

This service information letter contains the strapping and installation information necessary to install an AP-103F Flight Control System with FD-108 or FD-109G instrumentation in the Grumman Gulfstream G-159 Aircraft. Detailed information for installation, strapping, and checkout procedures are contained in the Collins AP-103F Flight Control System Maintenance Manual.

INSTALLATION DATA

1. AIRCRAFT TYPE: Grumman Gulfstream G-159

2. EQUIPMENT LIST:

2.1 AP-103F INSTRUMENTATION

TYPE	NOMENCLATURE	LOCATION
121A-2	Emergency Disconnect	Overhead Panel
327D-1W	Trim Indicator	Pilot's Panel
329B-7A	Flight Director Indicator (006 Status)	Pilot's Panel
331A-6A	Course Indicator	Pilot's Panel
332D-11	Vertical Reference	Rear Galley Floor
332G-2	Yaw Rate Gyro	Rear Galley Floor
334C-3/351B-3	Aileron Primary Servo (000 Status)	Below Radio Rack
334C-3/351B-3	Elevator Primary Servo (000 Status)	Below Pilot's Seat
334C-3/351B-3	Rudder Primary Servo (000 Status)	Below Co-Pilot's Seat
334D-2-33E	Trim Servo	Pedestal
334C-1D	Instrument Amplifier	Radio Rack
345A-4B	Sensing Unit	Under forward passageway Floor
562A-5G	Flight Computer	Radio Rack
562C-4	Autopilot Amplifier	Radio Rack
590A-3A	Altitude Controller	Behind Rear Pressure Bulkhead
590B-2	Air Speed Sensor	Radio Rack
614E-2F/2G	Pedestal Controller (002 Status)	Pedestal

2.2 FD-108 INSTRUMENTATION

TYPE	NOMENCLATURE	LOCATION
329B-7A	Flight Director Indicator	Pilots Instrument Panel

TYPE	NOMENCLATURE	LOCATION
331A-6A	Course Indicator	Pilots Instrument Panel
344C-1D	Instrument Amplifier	Nose Compartment

2.3 FD-109G INSTRUMENTATION

TYPE	NOMENCLATURE	LOCATION
329B-8G	Flight Director Indicator	Pilots Instrument Panel
331A-6A	Course Indicator	Pilots Instrument Panel
344C-1F	Instrument Amplifier	Nose Compartment
161E-1	Mode Coupler	Nose Compartment

3. SPECIAL DATA:

3.1 The analogs listed in this SIL are predicated on the use of the 332D-11 and MC-102 compass system.

NOTE For optimum operation, individual equipment may require HDG, CD, BANK; and pitch analog changes. Refer to SIL 302-66 for method of determining strapping.

3.2 In installations having FD-108 or FD-109G system installed on co-pilot's side using 562A-5G Flight Computer, strapping listed in this SIL is recommended.

3.3 After completion of installation of strapping mark GULFSTREAM G159 on the aircraft type label.

4. SERVO DATA:

4.1 SERVO TORQUES:	NOMINAL
4.1.1 Aileron	116 in. lbs.
4.1.2 Elevator	116 in. lbs.
4.1.3 Rudder	116 in. lbs.

4.2 Maximum Trim Tab Speed: 6.5 Degrees per Minute

5. 562C-4 GAIN POSITIONS

5.1 AILERON BOARD: (Figure 1)

5.1.A Roll Cmd Rate 1 5 degrees/second

5.1.B Ail Cmd Gain 11

5.1.C Roll Rate Gain	6	
5.1.D Turn Knob Limit	2	30 Degree/bank

5.2 ELEVATOR BOARD: (Figure 2)

5.2.A El Cmd Gain	13
5.2.B Alt Int Rate	9
5.2.C Pit Rate Gain	10
5.2.D Up El Gain	8
5.2.E Alt Gain	15

5.3 RUDDER BOARD: (Figure 3)

5.3.A Adverse Yaw	0
5.3.B Coordination	10
5.3.C Rud Cmd Gain	4

6. 562A-5G STRAPPING:

6.1 Roll Channel: (TB4, Figure 4 or 4A)

6.1.A Bank Analog		200	
6.1.B Hdg Analog		300	
6.1.F Bank Limits			
6.1.F.1 Hdg		25°	
6.1.F.2 VOR/LOC		25°	
6.1.F.3 APPR		15°	
6.1.G CD Rate	LOC	2.2	
	VOR	1.7	
6.1.H Bank Rate		0.3	
6.1.I Radio Rate		1[illegible].0	
6.1.J Radio Trip Point	001 Status		
	R14	42.2K	(705-0662-000)
	R15	46.4K	(705-0663-000)
	R16	110K	(705-0672-000)
	Strap	2 to 41	

003 Status

	CR12	1N754A	(353-2716-000)
	R14	34.8K	(705-0660-000)
	R15	34.8K	(705-0660-000)
	R16	147K	(705-0675-000)
	R17	75K	(705-0668-000)
	Strap	2 to 41	
6.1.K CD Analog		300	

6.2 Pitch Channel: (TB5, Figure 5, 5A, or 5B).

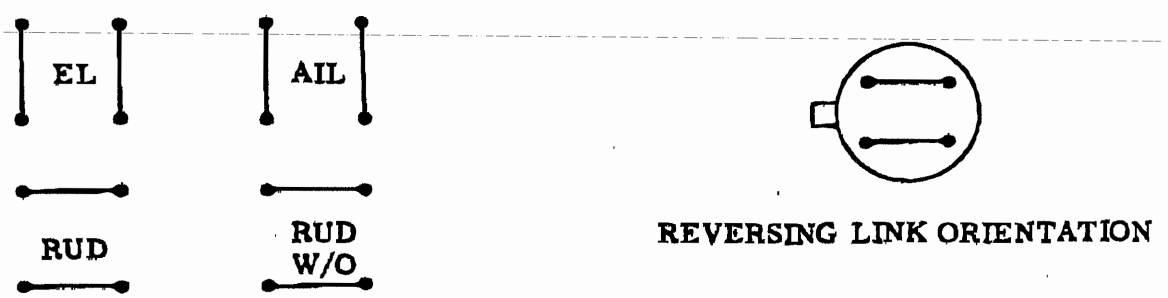
6.2.A Alt Rate		0	
6.2.B Alt Analog		10	
6.2.C Pitch Analog		200	
6.2.F Pitch Limit		12	degrees
		4.87K	
6.2.G Fixed Pitch Up		6	degrees
	R35	11.5K	(724-0119-000)
6.2.H Alt Gain		25	feet
6.2.I Pitch Down		1.0	degrees

7. LOGIC AND SPECIAL:

7.1 Short C27, 140 uf, on the rear board of the Aileron module.

7.2 Added diodes on 562A-5G TB5 are 1N645 (353-2607-000).

8. REVERSING LINKS:



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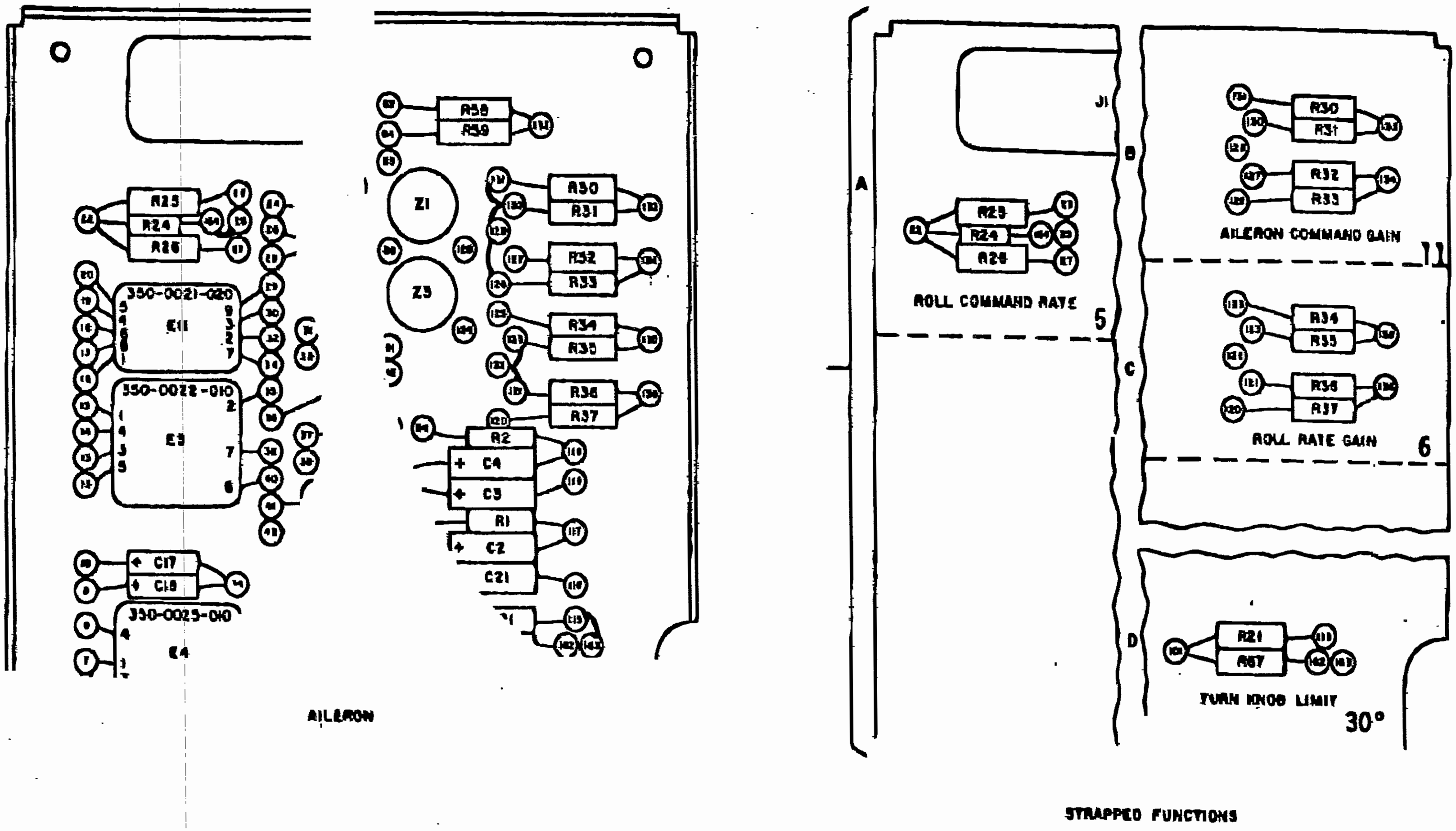
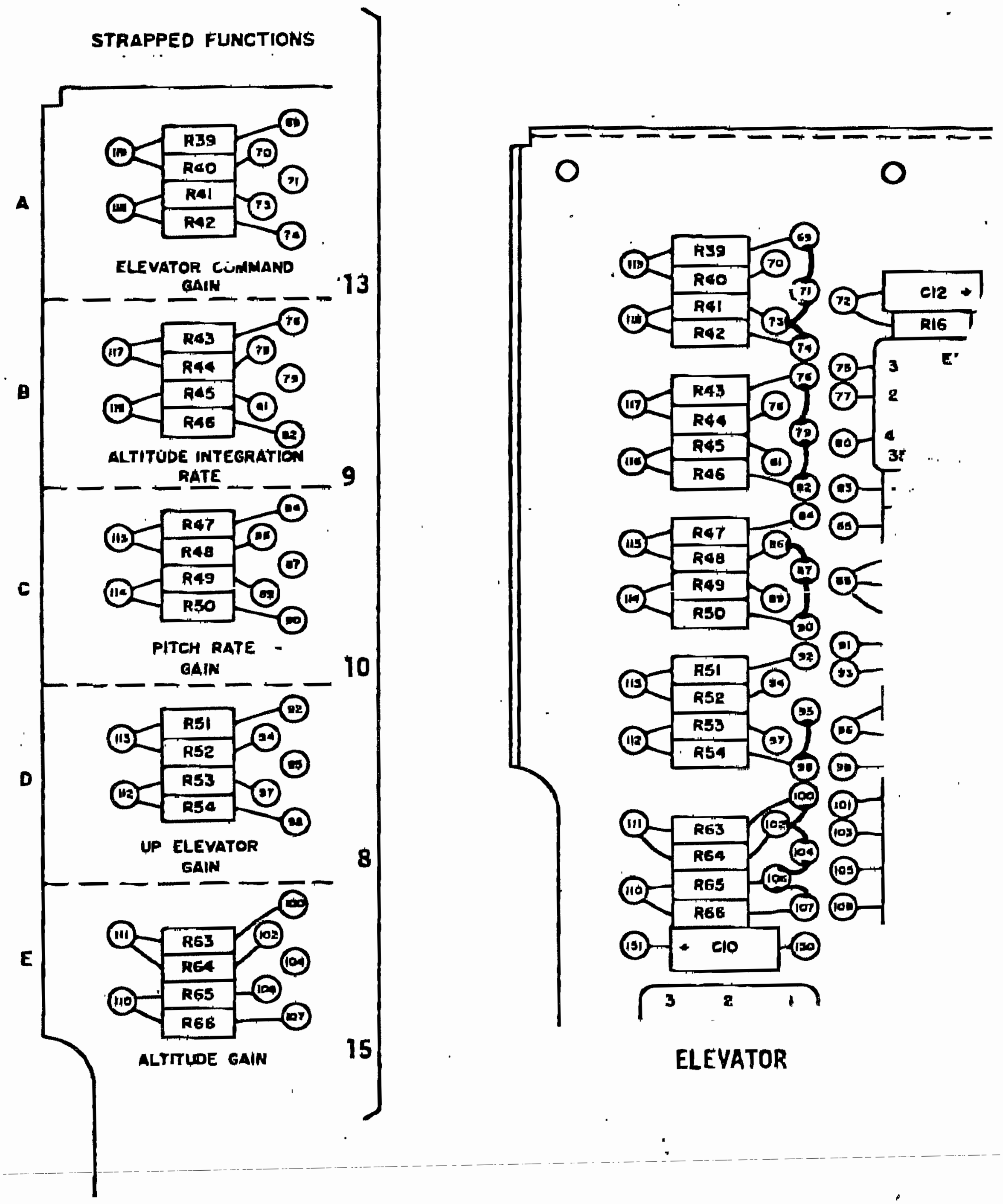


NOTE Short C27, 140-uf, on the rear board of the Aileron module

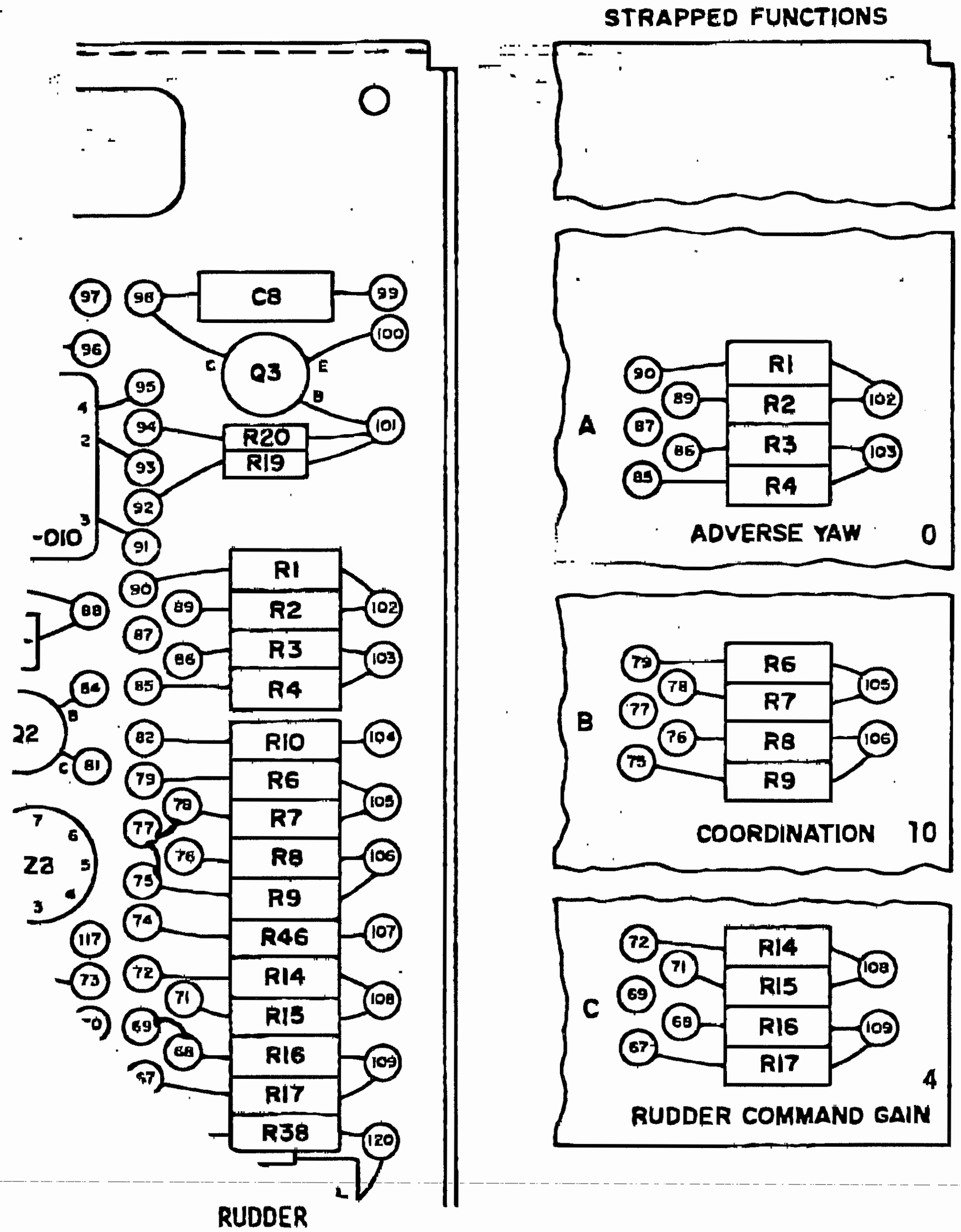
562C-4 Autopilot Amplifier, Aileron Board Strapped Functions

Figure 1



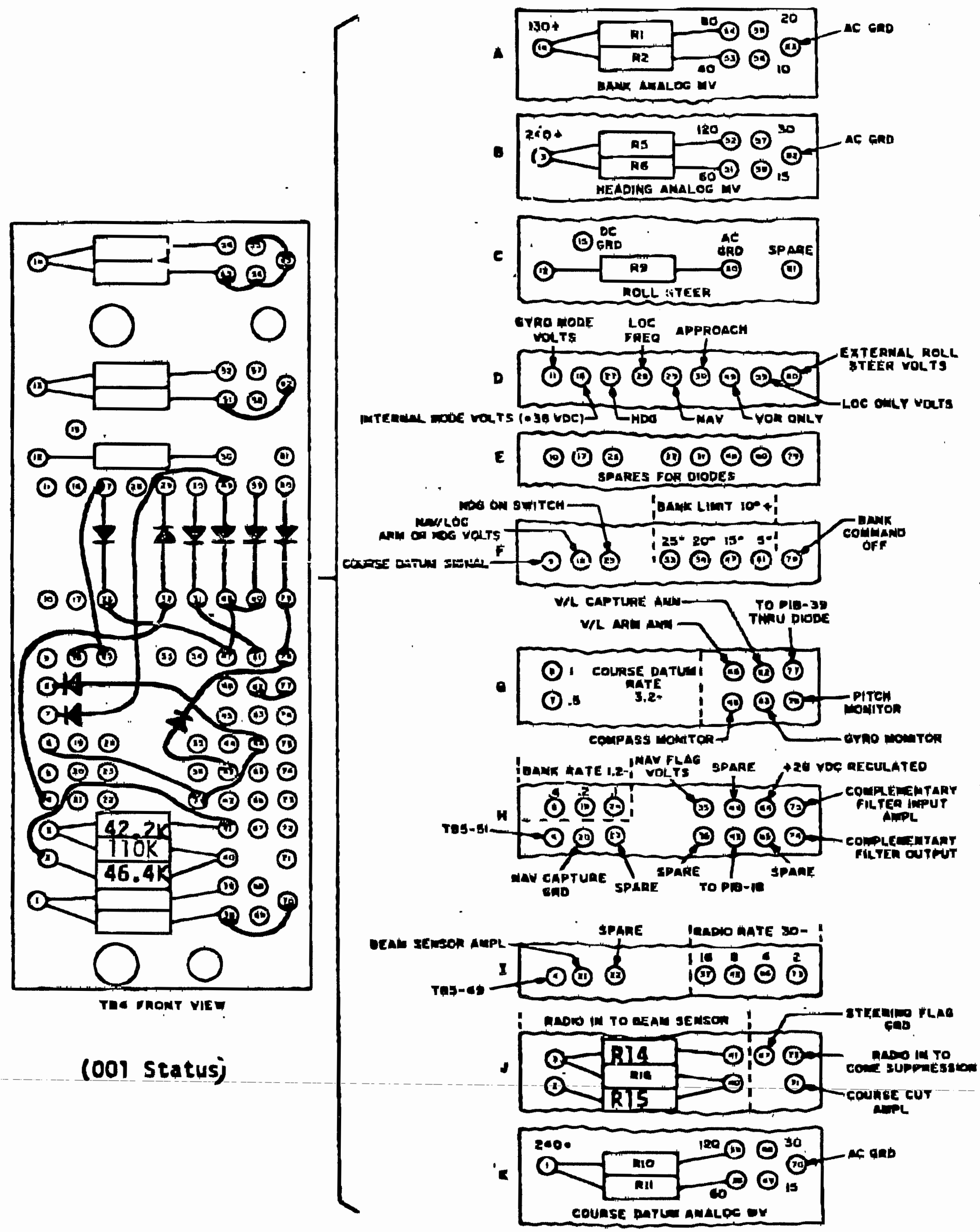
562C-4 Autopilot Amplifier, Elevator Board Strapped Functions

Figure 2



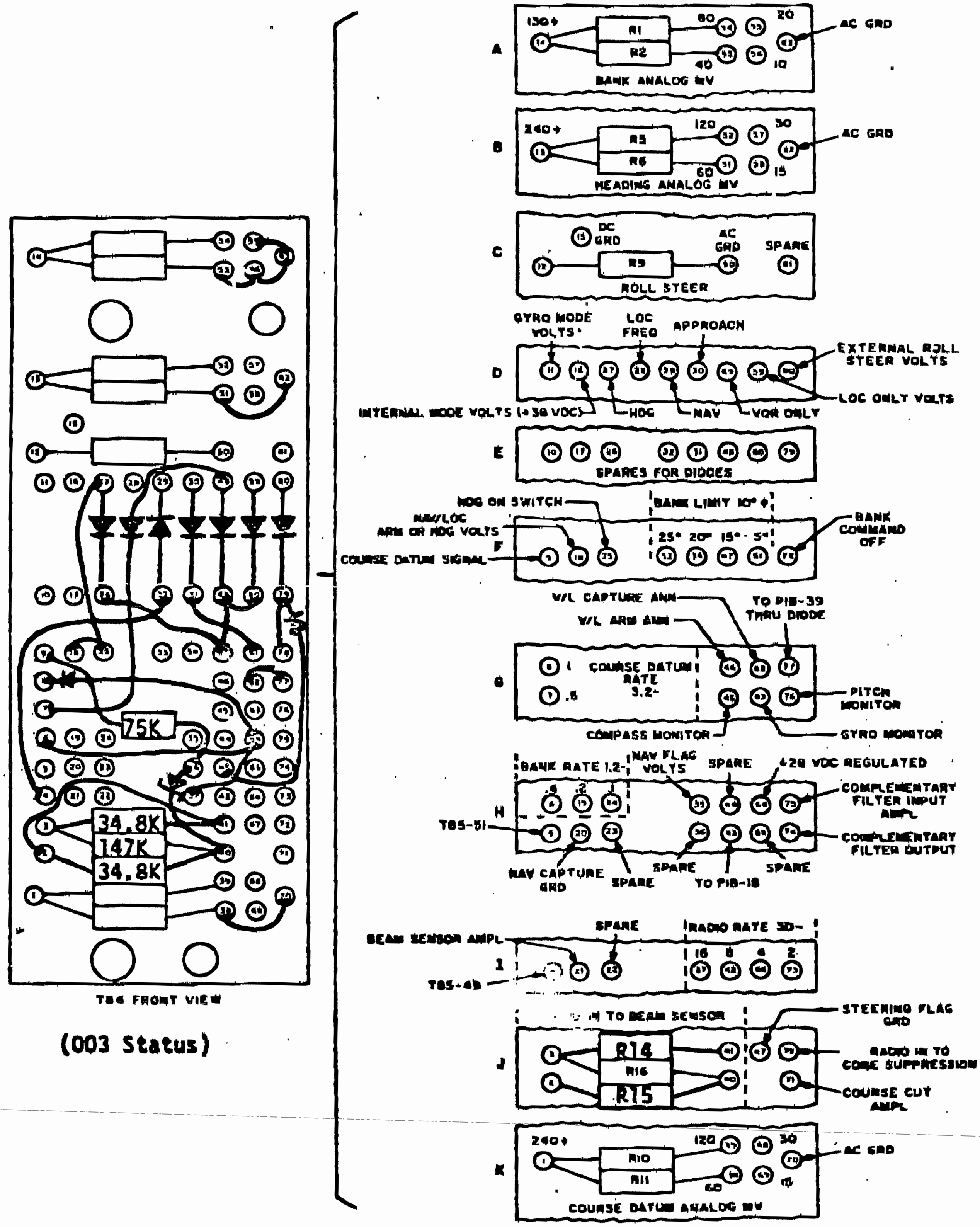
562C-4 Autopilot Amplifier, Rudder Board Strapped Functions

Figure 3



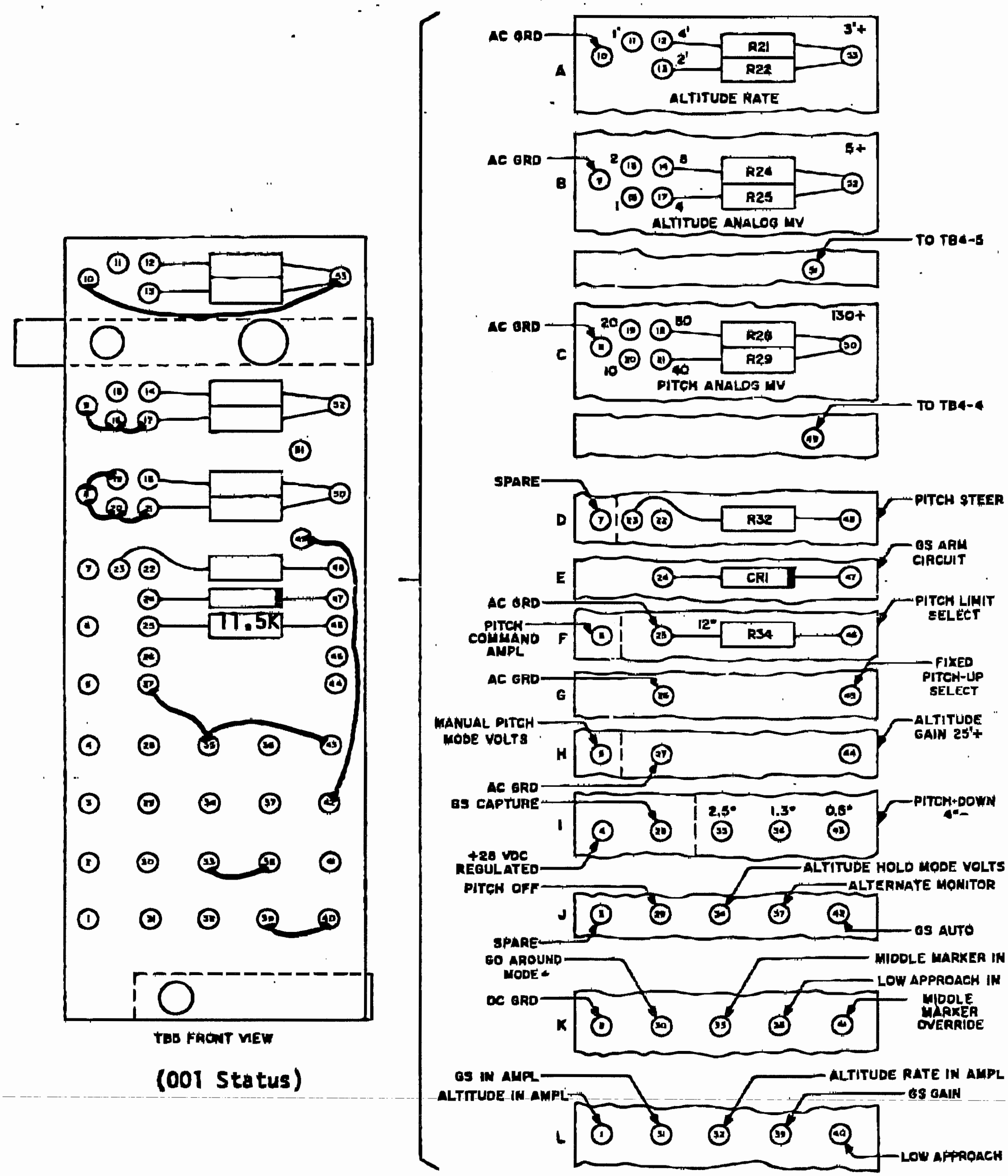
(001 Status)

562A-5G Flight Computer, Logic Board TB4 Functions
Figure 4



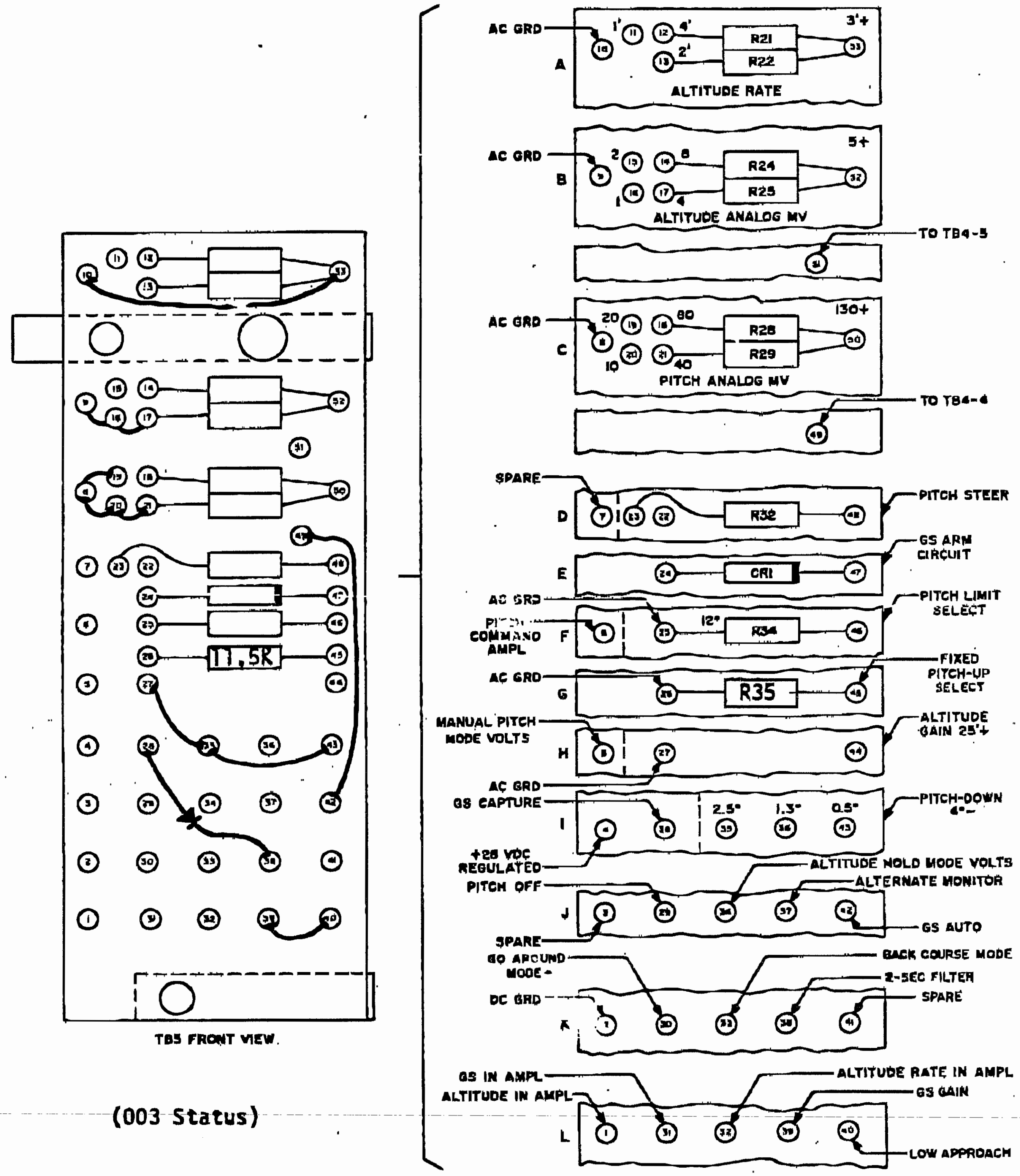
562A-5G Flight Computer, Logic Board TB4 Functions

Figure 4A



562A-5G Flight Computer, Logic Board TB5 Functions

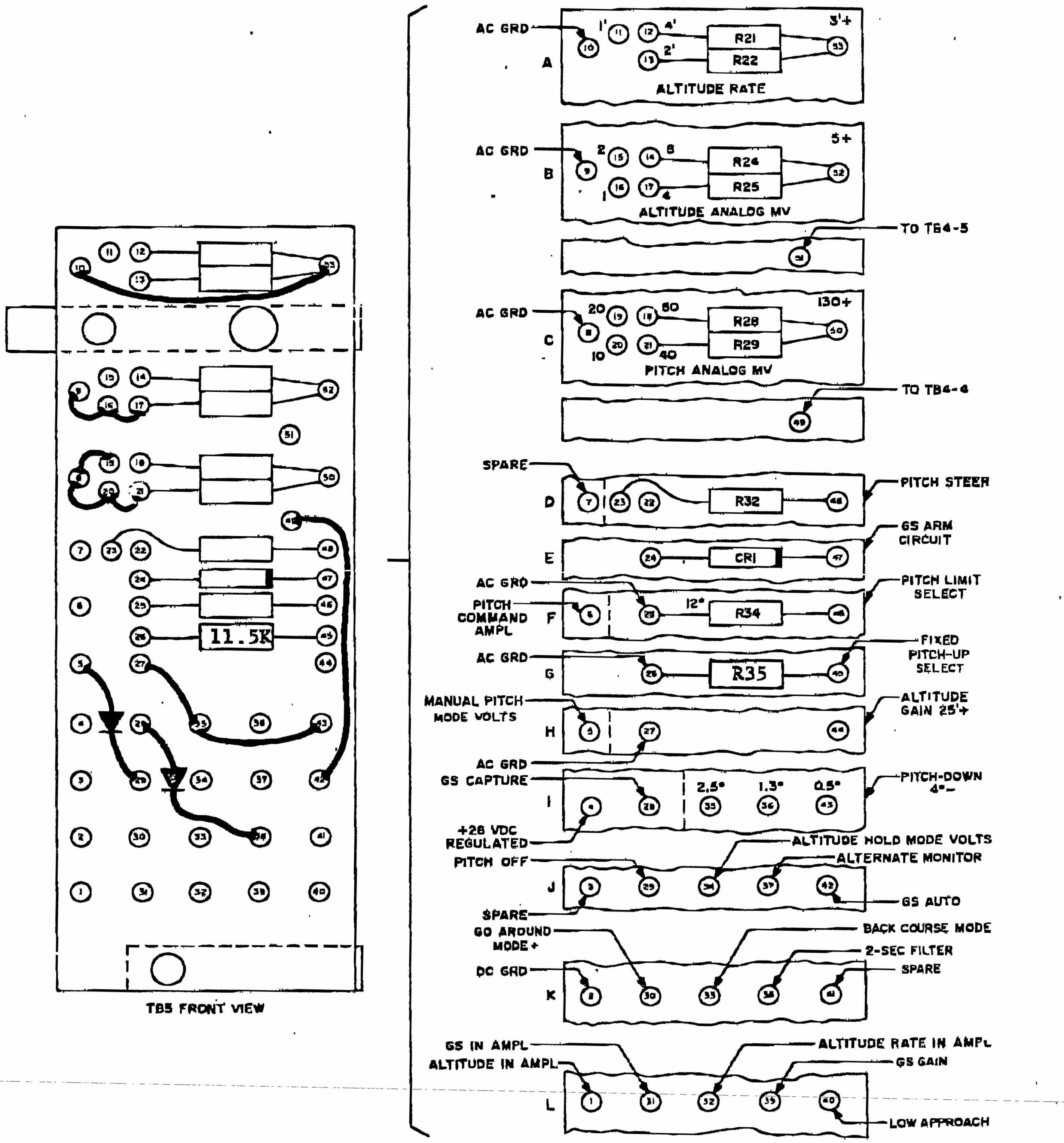
Figure 5



(003 Status)

562A-5G Flight Computer, Logic Board TB5 Functions, FD-108 Instrumentation

Figure 5A



562A-5G Flight Computer, Logic Board TB5 Functions
FD-109G Instrumentation

Figure 5B

Rockwell Avionics

Collins

400 Collins Road NE
Cedar Rapids, IA 52498

NOT REVISED REFERENCE ONLY

Facsimile

To:	Skyservice Attn: Ivan Marcheterre Quebec, Canada FAX:514-636-7009 ☎ 514-636-3300	**From**	Douglas W. Jessen Rockwell-Collins Collins Application Center FAX: (319) 295-0337 ☎ (319) 295-0618
Date:	October, 17, 1997	**Pages:**	1 (including cover page)

Message-

Subject: STC SA1-498 INSTALL COLLINS MODEL AP-103, AP-103D OR AP-103E OR AP-103F AUTOMATIC FLIGHT CONTROL SYSTEM

Hello Ivan,

The rigging tension is 45 lbs (+/- 5 lbs) (ALL Primary Servo's)

Best regards,

Douglas Jessen
Collins Avionics Applications Center
MS 164-100

Skyservice

October 17, 1997

COLLINS RADIO CO.
Cedar Rapids, IA

Attention: Mr. John Bohn

Dear John,

Following our telephone conversation, would you please supply me with the following information:

AP-103F, Auto Pilot Servos cable tensions for a Gulfstream I (G159), serial number 148. The installation was made as per STC SA1-498.

Please forward the information to my attention, by fax at 514-636-7009.

If you need additional information, I can be reached at 514-636-3300.

Thank you for your assistance.

Yvan Marcheterre
Avionics Shop Supervisor

9785 Ryan Avenue, Dorval, Quebec, Canada H9P 1A2 Telephone: (514) 636-3300, Fax: (514) 636-4855

LIST OF EFFECTIVE PAGES

CHAPTER 22

AUTOPILOT

SECTION	PAGE	DATE
*22	i	May 8/92
*22	ii	May 8/92
*22	1	May 8/92
*22	2	May 8/92
*22-0	1	May 8/92

* Pages revised by Revision 46
** Pages added by Revision 46
*** Pages deleted by Revision 46

TABLE OF CONTENTS

CHAPTER 22

AUTOPILOT

AUTOPILOT — GENERAL

Information for this Chapter to be supplied when equipment is installed

LIST OF EFFECTIVE PAGES

CHAPTER 23

COMMUNICATION

* Pages revised by Revision 49
** Pages added by Revision 49
*** Pages deleted by Revision 49

TABLE OF CONTENTS

CHAPTER 23

COMMUNICATION

VOICE RECORDERS — DESCRIPTION / OPERATION

1. General

The voice recorder system consists of a voice recorder unit installed in the tail compartment. A control unit typically installed in the copilots side console in the cockpit, a remote microphone installed in the cockpit and a bulk erase switch typically located at the radio rack area.

The system, which receives power from the essential bus, is a thirty-minute continuous loop voice recording system with test and bulk erase capabilities.

2. Voice Recorder Underwater Locator Beacon

The locator beacon is a battery powered unit which can send a pulsed acoustic signal into the surrounding water when activated by a water-sensitive switch. The unit consist of a self-contained battery, electronic module and a transducer. The battery is shock mounted and is separated from the electronic module by a built-in bulkhead in the case. Access to the battery is gained by removing a sealed end cap. The opposite end of the unit contains a teflon insulated water-sensitive switch. This switch must be free of dirt or moisture to prevent leakage currents across the switch. Clean switch a minimum of once a month by wiping clean with freon solvent.

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VOICE RECORDERS — MAINTENANCE PRACTICES

1. Voice Recorder — Functional Test

A. Ensure nutcracker system is in weight-on-wheels configuration.

B. Apply electrical power to aircraft and depress all appropriate system circuit breakers.

C. Ensure all audio select switches are OFF on all three audio control panels (if installed).

D. Insert a 600 ohm headset into jack provided in cockpit CVR test panel.

E. At test panel, press and hold TEST switch. Verify the following:

(1) Meter deflects at least four times within green band (one for each of the four channels).

NOTE: All four channels will be continuously tested as long as TEST switch is depressed.

(2) A continuous 400 Hz tone is audible.

F. Release TEST switch.

G. Provide audible signal into cockpit area microphone. Verify that in approximately 1 second an output from the area microphone will be audible through headphones.

H. Simultaneously press and hold for approximately 6 seconds the BULK ERASE switch, typically located at radio rack and the ERASE switch, located at CVR test panel. Verify a 400 cycle tone is present.

I. Disable cockpit area microphone by disconnecting cockpit CVR test panel, and install a headset jack adapter (see Figure 201).

J. Select all radios and NAVs ON and tune them to receive local signals (where available).

K. At pilot flight station, verify following systems are audible (1 second delay) at CVR test panel jack adapter:

(1) Boom mic.

(2) Oxygen mask mics.

(3) Hand mics.

(4) Cockpit interphone (if installed).

(5) Received COMM audio.

(6) Received NAV audio.

L. Repeat Step K. above at copilot station.

M. Repeat Step K. above at jump-seat station.

N. Using galley handset (if installed), verify the following is audible at CVR test panel jack adapter:

(1) Audio to crew stations.

(2) Cabin page.

O. Using each cabin handset (if installed), verify any audio directed to the crew is audible at CVR test panel jack adapter.

P. Remove headphone jack adapter and reconnect CVR test panel.

Q. In tail compartment, connect a 600 ohm headset to CVR and monitor.

R. On CVRs equipped with test buttons, press and hold TEST switch of each channel. Verify approximately 1 second after pressing each TEST switch, a 400 Hz tone is audible and meter (if installed) shows a 3/4 of full scale deflection.

S. Dismount CVR impact switch from equipment shelf.

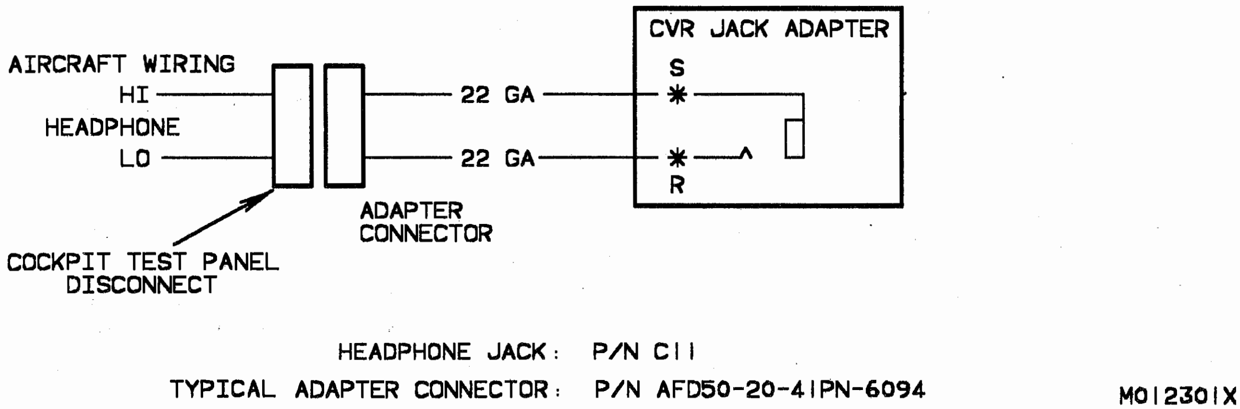
CAUTION: DO NOT STRIKE IMPACT SWITCH WITH ANYTHING OTHER THAN YOUR HANDS. DO NOT STRIKE TOP OF IMPACT SWITCH AS THE TOP CAN EASILY BE PUSHED DOWN ENOUGH TO DAMAGE IMPACT SWITCH.

T. While supplying an audio input into cockpit area mic, strike the base of impact switch sharply to close G-switch. Verify activation of red indicator on impact switch and the shutdown of cockpit voice recorder through the immediate loss of audio return.

U. Reset impact switch by pressing reset button adjacent to tripped indicator. Verify indicator light goes off.

V. Secure impact switch on equipment shelf.

W. Remove headset and return aircraft to original status.



NOTE: Installation May Vary. Verify part numbers by referring to Outfitters Drawings and/or Vendors Manuals.

CVR Jack Adapter
Figure 201.

2. Voice Recorder Underwater Locator Beacon Battery — Removal / Installation

Refer to Manufactures Maintenance Manual.

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CHAPTER 24

ELECTRICAL POWER

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CHAPTER 24

ELECTRICAL POWER

ELECTRICAL POWER — DESCRIPTION / OPERATION

1. **General**

(See Figure 1)

NOTE: This chapter describes the components and circuitry related to electrical power supply and distribution. Schematic diagrams included in this chapter supplement the descriptive text. For information concerning the actual wiring of the power supply and distribution systems (as well as the circuitry of other systems), refer to the Gulfstream I Wiring Diagram Manual.

Basic electrical power for the aircraft is provided by the dc power system consisting of two 27.7 volt (regulated), 300 ampere generators connected in parallel, provisions for applying external power, a tie and distribution network, control and indicating components, and two 24 volt, 34 ampere-hour nickel-cadmium batteries. On Aircraft 1 - 200, 322 and 323 having ASC 194, a battery failure monitor system is installed. This system will provide a battery current indicating monitor system that will provide charge and discharge readings and monitor increasing charging rates or fixed charge values.

Ground engine starting power is supplied by the two batteries, external power or both. An auxiliary source of dc power is provided by the auxiliary power unit driven-generator. This generator may be either a 50 or 200 ampere rated unit, set at a voltage slightly lower than the engine driven generators (27 volts). This unit is designed for battery charging and limited system power on the ground. It also provides emergency dc power for flight use. The generator is not designed to be paralleled with the engine driven generators, nor to be used during engine starts.

The Gulfstream I also contains a fixed frequency and a variable frequency ac system for ac requirements. The fixed frequency system consists of three or four inverters and associated control, protective and indicating components. This system provides single phase ac power for frequency sensitive ac equipment. The variable frequency system consists of two alternators, one per engine with associated control, protective and indicating components. These units provide three phase ac power for resistive heating purposes on propeller, engine duct, and windshields.

The dc power supply and distribution system, and the fixed frequency (115 volt, 400 cycle, single-phase) ac power supply and distribution system are of single wire design and use the aircraft structure as ground return. The variable frequency (212 volt, 11.0 KVA, 206 to 413 cycle), three phase ac power supply and distribution systems are three wire systems with line-to-line voltage only, and no ground return.

Aircraft 1 - 200, 322 and 323 having ASC 231, have circuit breakers installed in the left and right nacelle, to protect the wiring and electrical equipment located within the nacelle area from electrical overload.

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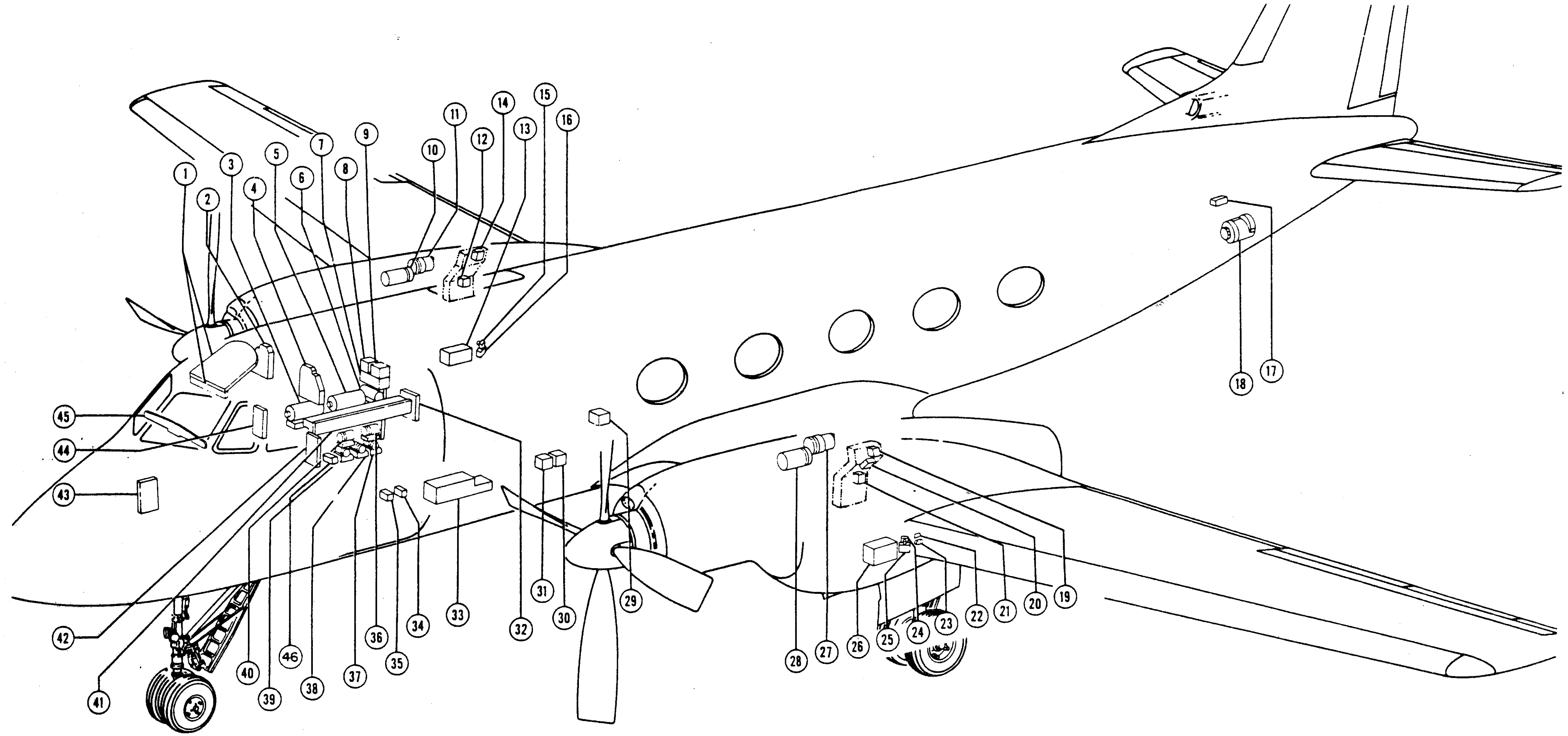


1. Overhead Panels	11. Right Alternator	23. External Power Receptacle	35. R. Gen. Overvoltage Relay
2. Right Circuit Breaker Panel	12. R. Gen. Reverse Current Cutout	24. L. Bat. Emergency Relay	36. R. Alt. Voltage Regulator
3. Inverter B	13. Right Battery	25. L. Bat. Normal Contactor	37. APU Gen. Voltage Regulator
4. Left Circuit Breaker Panel	14. R. Alt. Current Balance Relay	26. Left Battery	38. R. Gen. Voltage Regulator
5. Inverter A	15. R. Bat. Emergency Relay	27. Left Alternator	39. L. Gen. Voltage Regulator
6. Inverter E	16. R. Bat. Normal Contactor	28. Left Generator	40. L. Alt. Voltage Regulator
7. Alternator Relay Box	17. APU Relay Box	29. Main Power Junction Box	41. Inverter Relay Box
8. R. Alt. Overvoltage Sensing Transformer-Rectifier	18. APU Generator	30. L. Alt. Field Transformer-Rectifier	42. Left Sta. 133 Relay Box
9. L. Alt. Overvoltage Sensing Transformer-Rectifier	19. L. Alt. Current Balance Relay	31. R. Alt. Field Transformer-Rectifier	43. Pedestal Terminal Panel
10. Right Generator	20. External Power Relay	32. Inverter Fuse Panel	44. Right Sta. 133 Relay Box
	21. L. Gen. Reverse Current Cutout	33. Mid Relay Box	45. Eyebrow Panel
	22. External Battery Switch	34. L. Gen. Overvoltage Relay	46. APU Gen. Overvoltage Relay

Location of Electrical Power Equipment
Figure 1.

GENERAL ELECTRICAL INFORMATION — DESCRIPTION / OPERATION

1. General

This section covers certain general aspects of design and construction common to all Gulfstream I electrical systems. Details of actual systems are discussed in their appropriate section of this manual. The following information is intended to lay the ground work for a basic understanding of the overall electrical systems design so that maintenance personnel can better Fault isolate those systems causing difficulty.

2. Wiring

Wire in the Gulfstream I is generally of two types:

A. Stranded copper with Teflon and Fiberglass insulation conforming to MIL-W-5086.

B. Stranded aluminum with Teflon and Fiberglass insulation. This wire usage is held to a minimum. It is used only in heavy wire gages, on long runs where a weight advantage can be gained. This wire conforms to MIL-W-7072.

NOTE: Several other types of wire are used, such as "hi-temp" wire, but its usage is limited. "Hook up" wire conforming to specification MIL-W-7076 or MIL-W-16878 is used in protected locations for ease in routing. This wire lacks an abrasive proof outer jacket, and is thus unsuited for general use.

3. Wire Identification

All wiring in the aircraft which is installed by Gulfstream Aerospace is identified in accordance with MIL-W-5088 wiring identification code. This section deals only with the Gulfstream original factory installation. Wiring which is a part of the Rolls-Royce supplied engine harness, will not be discussed. Refer to appropriate DART manual for details on these assembly. Each wire in the aircraft has a number hot-stamped on it in several places near each junction end, and ever 12 to 18 inches along its entire length. This identification number is not used for any other wire in the aircraft.

NOTE: The term wire as used herein, refers to one segment of wire, not a circuit made up of several lengths. A circuit may be made up of one or more segments or pieces of wire. Each piece of wire has its own identity.

For the convenience of those who have never utilized the MIL-W-5088 wiring identification code, it will be quoted in part in this section. Basic facts about this system of wire markings are as follows:

A wire will be marked with a letter, followed by a number, another letter and another number. If the wire is a ground or is part of an ac carrying circuit, it will also have a letter at the end. Examples: Q105B12, Q5F20N. .

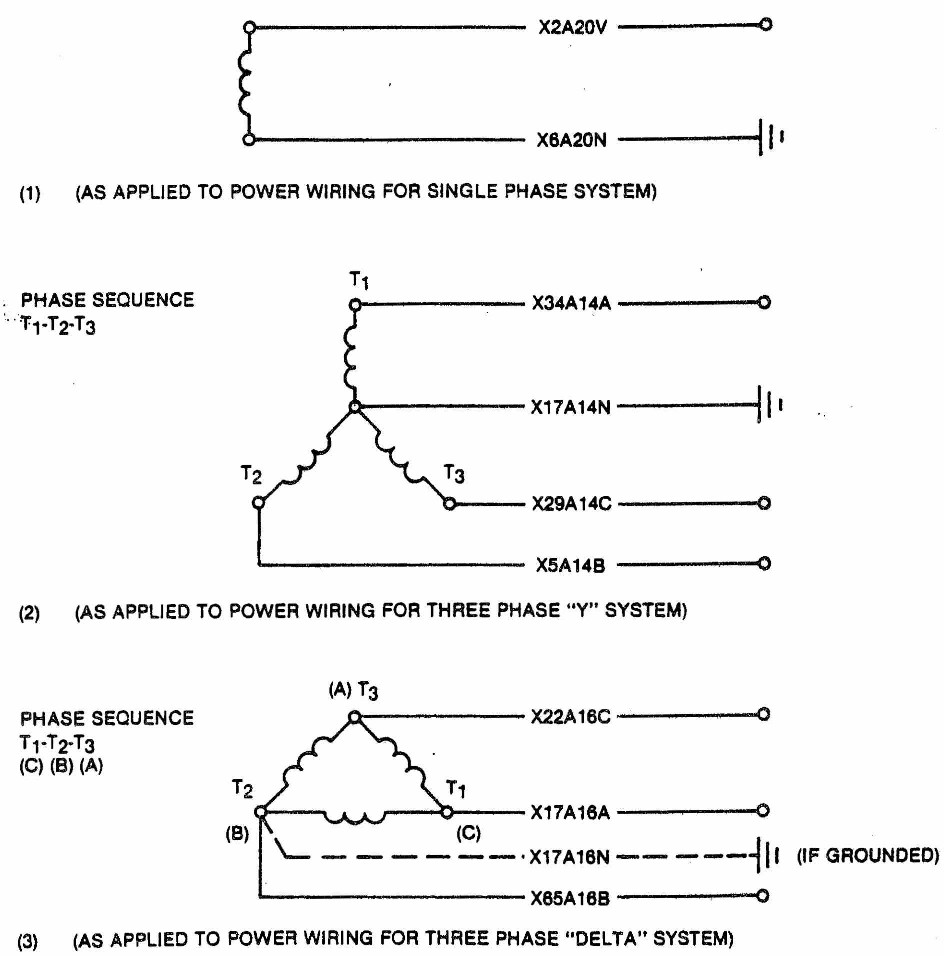
These wire numbers can be identified as follows by using the code:

```
Q    105   B     12
|     |     |     |__ WIRE SIZE IN AWG
|     |     |__ SEGMENT LETTER
|     |__ WIRE IDENTITY NUMBER
|__ CIRCUIT FUNCTION LETTER
```

```
Q     5     F     20     N
|     |     |     |      |__ GROUND
|     |     |     |__ WIRE SIZE IN AWG
|     |     |__ SEGMENT LETTER
|     |__ WIRE IDENTITY NUMBER
|__ CIRCUIT FUNCTION LETTER
```

Utilizing the MIL-W-5088 specification, we find that a prefix letter Q on a wire indicates that its function is in a fuel or oil system.

The wire identity number is selected by the design engineer who designs the circuit and serves to identify that wire as a part of a certain circuit. Wire identity numbers only take on real meaning when referring to the wiring diagram of that circuit.



Wire Identification Application
Figure 1.

The segment letter is utilized to identify that piece of wire or segment of a circuit. There may be many wires in a circuit coded Q105 but there is only one Q105 wire. Usually, the segment letter starts with A at one end of a circuit. As a circuit progresses, the A segment will terminate at a plug or terminal strip. The succeeding wire segment of the same circuit will then be coded B segment, and so on through the alphabet until the circuit is completed. Parallel circuitry may be branched off from this circuit in one of two ways:

- Utilizing the letter following the last segment of that circuit.
- Utilizing an entirely different wire identity number. This can only be ascertained by using the wiring diagram for that system.

In the second example given, note that there is a letter following the wire size. This letter N appearing at the end of a wire number signifies that that wire is a ground. Other variations used in MIL-W-5088 are:

- "A" phase A of a three phase AC circuit
- "B" phase B of a three phase AC circuit
- "C" phase C of a three phase AC circuit
- "V" ungrounded phase of a single phase AC circuit

Other pertinent facts about the MIL-W-5088 Spec are:

- All AC power circuits between their generating device and their distribution point are coded with the circuit function letter C.
- After these ac circuits pass through their individual circuit breakers and receive an identity as part of a particular system, they then assume the circuit function letter of their using system, e.g., from inverter output to the distribution point a wire is coded X, since it is only a part of the distribution system. From the distribution point where it picks up a circuit breaker, it then might assume the Q identity as part of a system.
- All DC control wires concerned with AC generating devices, such as field circuits, relay circuits, etc., are coded with the circuit function letter V.

NOTE: V used as a circuit function letter is a DC control circuit for AC generating systems. V used as an end character signifies the ungrounded phase of a single phase ac circuit.

Some of the listings under MIL-W-5088 are not applicable to the Gulfstream I, such as mine laying gear, etc. They are deleted from the following excerpts from the specifications:

CIRCUIT FUNCTION LETTER	CIRCUITS
C	Control Surfaces
D	Instrument (other than flight or engine)
E	Engine Instruments
F	Flight Instruments
G	Landing Gear
H	Heating, Ventilating and De-icing
J	Ignition
K	Engine, Propeller
L	Lighting
M	Miscellaneous (used for door control)
P	DC Power
Q	Fuel, Oil and Hydraulic
R	Radio
SN	Radar
T	Special Electronics
V	Inverters, Alternators
W	Warning and Emergency
X	AC Power
Z	Spare Wires (as Utilized by Gulfstream)

4. Spare Wires and Terminals

Original Gulfstream I design considerations made provisions for future electrical alterations and additions. Toward this end, 216 spare wires and 160 spare terminals were installed throughout 12 junction boxes in the aircraft. (These quantities do not include the radio terminal panel where 1232 additional terminals were originally provided for use during the installation of electronic packages.)

Periodically, Aircraft Service Changes (ASC) are issued for specific circuit changes. These change publications include schematic diagrams for theory of operation, wiring diagrams to provide connection details, and step-by-step instructions on how to accomplish the change. Both the system wiring diagram and the step-by- step instructions illustrate the complete new wiring of each point (terminal) touched by the system. Always follow the instructions given in an ASC exactly as prescribed. Failure to do so can result in crossed circuits, circuit failure, and/or aircraft failure.

5. Terminal Strip and Junction Box Identification

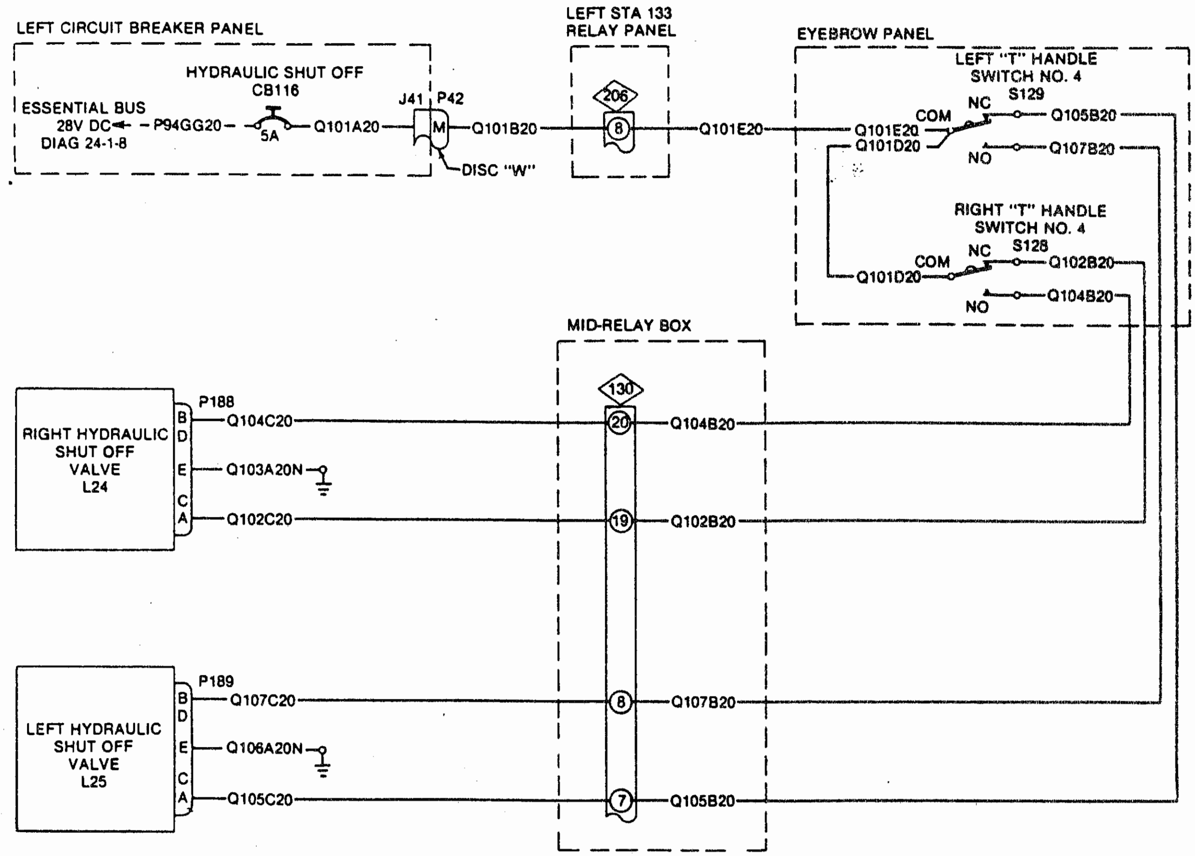
(See Figure 2)

All junction boxes are identified by a name. This name, where space allows or when it is not perfectly obvious, appears on the cover of the box.

All terminal strips in all boxes are identified by a number. This number appears in a diamond symbol at one end of the terminal strip. No two terminal strips in the aircraft have the same number. Every terminal on each terminal strip also is numbered by a white micarta identification plate adjacent to the terminal.

All aircraft wiring diagrams indicate the following:

- Junction box in which a circuit is terminated.
- Terminal strip number on which segments are terminated.
- Terminal number of that terminal strip on which that segment is terminated.
- Cable assembly or box assembly of any and all parts of that system.
- Part numbers, manufacturers, etc., of all vendor supplied major components in that drawing.
- Plug and receptacle part numbers and pin letters of those connectors through which that circuit passes.
- Applicable specification or processes involved in the manufacture of assembly of this system, as required.



Typical Wiring Diagram
Figure 2.

6. Potting of Connectors

With very few exceptions, the soldered end of all plugs and mating receptacles in the aircraft are filled with a synthetic rubber compound commonly known as potting.

Those plugs which are not in flexible conduit installations do not contain any backshell or backshell clamp. The potting compound provides the support and waterproofing function previously accomplished by a backshell and backshell clamp. Some flexible conduit plugs have backshells filled with potting compound.

Vinyl tubing (spaghetti) is not used at the soldered pins of any potted plug or receptacle. The potting supplies this insulation.

Some wire runs pass through a type of bulkhead fitting without a break. This is a feedthrough fitting. It is potted on both sides to provide pressurization integrity of the pressure vessel. The three large aluminum cross-tie cables running between nacelles are examples of this type of potting.

With very few exceptions, those potted plugs and receptacles with unused pins are fitted with the largest size wire that the pin will accept. This is usually a short length and is considered a spare wire. If a circuit is open in that plug to a failed solder connection in the potted section, or an additional circuit is needed to pass through this plug and receptacle, these spare wires may be utilized.

Most spare wires from a potted plug and receptacle combination are laced into the bundle of circuits passing through the plugs and receptacles, and can be found within 8 to 12 inches from their plug. They have a small rubber boot over their dead ends. Picking up these circuits for use merely involves stripping off the boot, stripping the wire, and utilizing an insulated permanent, solderless splice to tie additional wire lengths into the circuits. The same is done on the mating pin spare wire on the receptacle end. Certain relays and switches also are potted where connections terminate. This is for insulation and isolation purposes.

To repair a failed circuit in a potted plug the following procedure is used if no spare wires are available.

- Locate the bad circuit pin in the plug or receptacle as to physical position in the pin group.
- With a very sharp, thin bladed knife or razor blade, carefully cut away the potting around the bad pin until the soldered part is exposed and free.
- Carefully, with a pencil iron or other small tipped soldering iron, clean out all broken wires from the pin.
- Re-solder the old wire or a new length into the pin. Take care not to touch any of the other pins or potting left on the plug with the hot iron.

WARNING: CARBON TETRACHLORIDE IS NOT TO BE USED ON ANY ELECTRICAL CONNECTION OR PLUG.

CAUTION: INSPECT THE SOLDERING VERY CAREFULLY FOR DROPS OF METAL, ETC., BEFORE REPOTTING.

AFTER WASHING DO NOT TOUCH WASHED AREA WITH THE FINGERS, AS THEY WILL LEAVE AN OILY RESIDUE.

- Using an approved cleaning agent, clean the rosin and debris from the pin in question and wash approximately two to three inches of the wire with the solvent. This is done to ensure proper adhesion of the potting to the wire and the pin.
- Prepare some potting compound as per instructions on the container.
- Application can be done in one of two ways.

  Using a small stick or brush, gradually build up a sufficient amount of potting compound on the wire and the pin to fill in the cavity which was cut out.

  Using masking tape, cellophane tape or even smooth face card board, form a well around the cut out portion of the plug, and fill the well with potting compound.

- Work the compound well down into the soldered section, taking care that no air pockets are present. This can be done by packing it down with a small stick, and using compound which is rather fluid in consistency. A cone made from stiff paper formed like a funnel can be helpful if a lot of compound is needed.
- Let the plugs, wires, and potting set for a least 8 hours before moving it. Requires 24 hours for complete cure. Heat lamp will reduce cure time.
- Strip off the masking tape or card board, if used, and check to make sure that there is no cavitation or larger air bubbles in the newly potted section, which would be a path for moisture into the plug. Touch up with additional compound if necessary. Plug is then ready for use.

CAUTION: KEEP HYDRAULIC FLUID AWAY FROM POTTING WHETHER PRIMED OR NOT.

Hydraulic fluid will attack and completely deteriorate potting compound. Therefore, most potted plugs are painted with hydraulic fluid resistant epoxy resin primer.

Handling potted plugs and receptacles is no different than regular backshell and clamp type plugs. Observe normal precautions when handling or installing these units. Do not put excessive tension or bend radius on wires close to the plugs or receptacles, since this will cause eventual failure of the soldered connections.

Potting is not a cure-all, but properly handled and with the normal precautions exercised for any plug or receptacle, will give long service. The advantages of water proofing and pressurization sealing of plug and receptacles using this method far outweigh its disadvantages.

All plugs which are potted also contain a special Wiring installed between the shell and the cup nut to prevent moisture from entering through the threaded portions. It also serves as a locking device similar to a lock washer.

NOTE: A W-ring is an O-ring with a square cross-section area.

7. Circuit Protective Devices

A. With the exception of eight small cartridge fuses in the wing and tail deicer timer, no cartridge fuses are used in the electrical installation as delivered.

B. Current limiters are large protective devices such as slow-blow and stud mounted fuses. These are used for large loads.

C. Trip free circuit breakers are used in all other circuits requiring protection in the Gulfstream I.

D. The circuit breakers used are of two basic types.

(1) The common pull-type, that is, the type which can be manually pulled to break the circuit involved.

(2) Non-pull type, i.e., a small amperage breaker which cannot be pulled manually without breaking the head. Non-pull type breakers can be identified as follows:

- They are very small of fractional amperage.
- They have a red head.
- The head is about half the diameter and half the height of the common pull type.
- The head is not ridged for a finger hold.

E. All circuit breakers which are fed from the essential dc bus are painted Dayglo orange on their heads for fast identification.

F. Most protective devices are located in one of four places.

- Large fuses are located in the inverter fuse panel, on the aft side of Fuselage Station 193 bulkhead, on the right side of the cabin.
- Pilots circuit breaker panel, behind the pilots head in the cockpit.
- Copilots circuit breaker panel, behind the copilots head in the cockpit.
- Inverter relay circuit breaker panel, electronics compartment Fuselage Station 133 to 193, on the right side of the aircraft.

G. Some circuit protective devices are located in isolated areas which can be difficult to locate. They are as follows:

Unit	No. Per A/C	Location
L & R DC GEN. VOLT-AMM. Circuit Breaker	2	Mid-relay box, FLR-8
R. ALT. VOLTMETER Circuit Breaker	2	Right nacelle relay box
L. ALT. VOLTMETER Circuit Breaker	2	Left nacelle relay box
L. GEN. WARN - Circuit Breaker (Aircraft having ASC 231)	1	Left nacelle relay box
R. GEN. WARN - Circuit Breaker (Aircraft having ASC 231)	1	Right nacelle relay box
L. GEN VR SENS - Circuit Breaker (Aircraft having ASC 231)	1	Left nacelle relay box
R. GEN VR SENS - Circuit Breaker (Aircraft having ASC 231)	1	Right nacelle relay box
L. GEN. VR - Circuit Breaker (Aircraft having 231)	1	Left nacelle relay box
R. GEN. VR - Circuit Breaker (Aircraft having ASC 231)	1	Right nacelle relay box
L. GEN. OVR - Circuit Breaker (Aircraft having ASC 231)	1	Left nacelle relay box
R. GEN. OVR - Circuit Breaker (Aircraft having ASC 231)	1	Right nacelle relay box
EXT PWR CONT - Circuit Breaker (Aircraft having ASC 262)	1	Left Nacelle Battery Enclosure.
GEN OUTPUT SENSE/ FIELD - Circuit Breaker (Aircraft having ASC 262)	1	APU RELAY BOX
Outside Battery Switch Circuit Breaker	1	Battery compartment, left nacelle
Windshield Heat Power Detecting Relays Circuit Breaker	8	FLR-2, inside windshield heat relay box
Pneumatic De-icing Timer Output Fuses	8	Inside timer, FLR-7
Left Landing Light Current Limiter	1	Left nacelle relay box
RIGHT Landing Light Current Limiter	1	Right nacelle relay box
APU Overload Control (175A) For Generator (200A)	1	Inside APU relay box
Circuit Limiter - (Aircraft 1 - 60 having ASC 262)	1	APU relay box.

8. General Electrical Diagram Practices

Unless otherwise stated on the drawings:

- All switches are always shown in the off position if there is one.
- All plunger type microswitches are always shown in the released, plunger not depressed, position.
- All relays are always shown in the relaxed, de-energized position.

9. Bolted Electrical Connections — Torque Values

Bolted (or screw thread) electrical connections are rugged and usually considered foolproof. However, when a joint is not sufficiently tightened it will run hot, while on the other hand, an overtightened terminal may be fractured and be prone to snap off.

A practice of always tightening to specific torque values will prevent either type of undesirable effect. Suitable torque values are:

- 1/4 inch bolt 40 to 60 inch pounds
- 5/16 inch bolt 65 to 95 inch pounds
- 3/8 inch bolt 95 to 110 inch pounds

On a connection utilizing any form of stop nut (elastic or otherwise) amend the torque value as follows:

- Run nut on stud until it does not quite make contact with terminal pile-up.
- Measure torque value required to turn nut at this point.
- Add this torque to that listed above and tighten connection to the total value.

The size of the washer used is another point to consider in bolted connections. It is necessary to use a washer large enough to exert squeeze over most of the terminal tongue area. In the case of aluminum terminals, the washer must squeeze the full diameter of the tongue to prevent cold flow from occurring. Proper connections should not run significantly hotter than their surrounding wires. Any hot spot denotes trouble.

10. Miscellaneous Nomenclature

Nutcracker System

This is a system of strut switches which sense whether the aircraft has weight on the landing gear or not. These switches actuate relays whose contacts control certain circuits which need "on ground" or "in flight" information for various purposes.

Emergency Gang Bar

The emergency gang bar is a large red, spring loaded lever in the center overhead panel. It is used to place the electrical system into emergency configuration in the event of certain flight emergencies. Pulling the bar down trips both engine-driven dc generators, both engine-driven alternators, and places the master battery switch into emergency, thereby eliminating all non-essential electrical loads.

Fire Pull T-Handle

There is one fire T-handle for each engine, mounted on the cockpit eyebrow panel. Each provides for isolation of all electrical and fluid systems for its particular engine. Each T-handle is actually a linear operated bank of electrical switches. Pulling the handle automatically shuts off that engine's alternator and dc generator, and shuts off the fuel, water methanol, and hydraulic fluid for that engine. This action also exposes a fire extinguisher switch for that engine. Each handle lights up when the associated engine's fire detection senses an alarm configuration.

11. Location Diagrams

(See Figure 3 thru Figure 20.

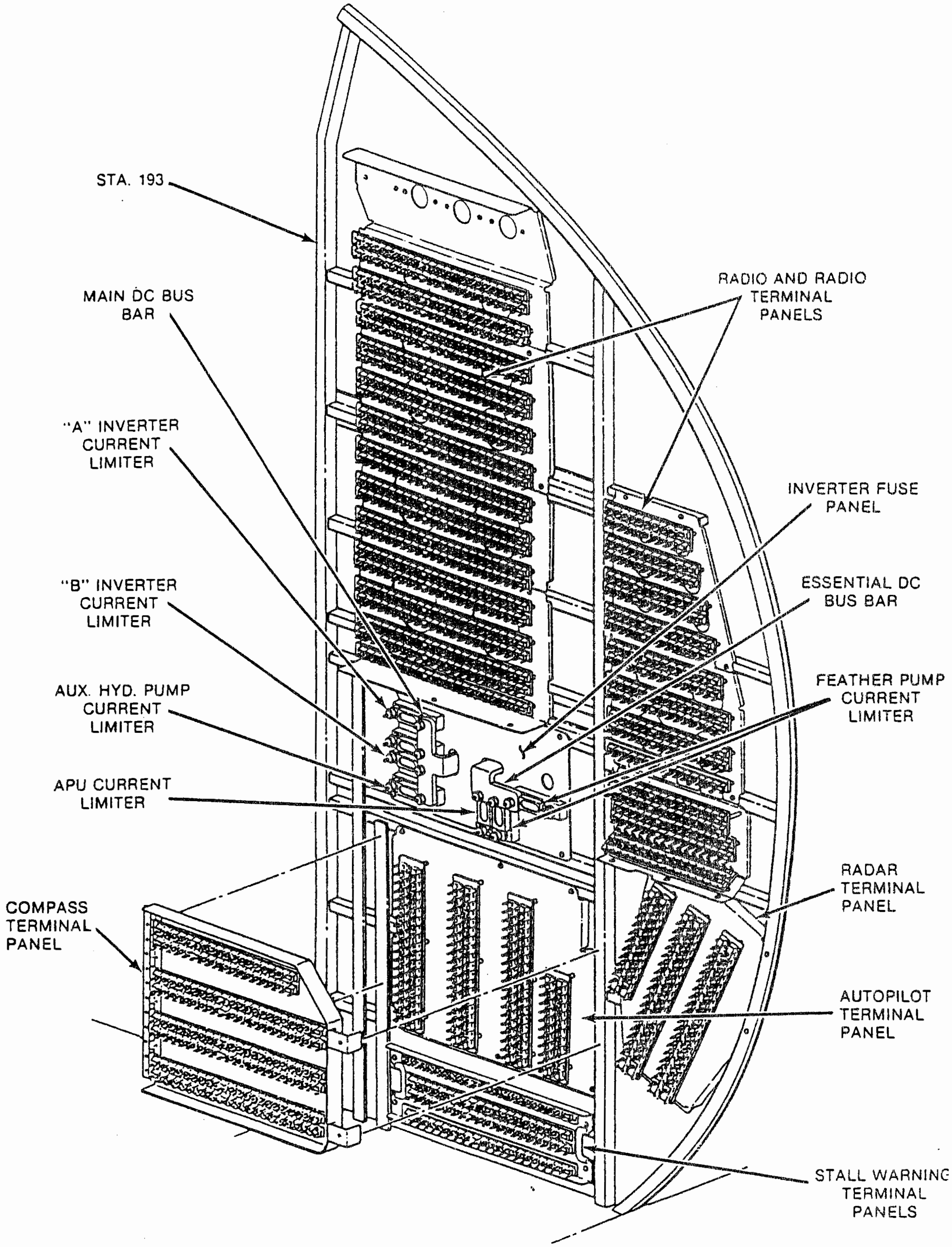
All cabin floorboards are numbered. Major components located under each floorboard are located on the indicated figures. Details of major relay panels with locations of individual units are included.

Unit	Mfg. & Unit Part No.	Brush Part No.	Quantity Per Unit	New Length	Minimum Length
DC Gen., Engine 300 Amp	Jack & Heintz 30007-003	23014-1035	4	1 21/64 in.	11/16 in.
DC Gen., Engine 300 Amp	Bendix	C-1530453	8		
DC Gen. APU		—		Checked at APU Overhaul See Chapter 5.	
Alternator	Rotax BA.1001	N.146713			0.5 in.
Engine Starter	Rotax C.5104	N131858			0.718 in.
Inverter "E" 250VA	Leland SE-16-3	AC (Straight) SA-10215 AC (Angle) SA-10214 DC A-10007	2 2 4		5/16 in. * 5/16 in. * 7/16 in. *
Inverter "A" or "B", 1500VA	Leland SE-20-2	AC (Straight) SA-9238 AC (Angle) SA-9236 DC A-10303	2 2 4		7/16 in. 7/16 in. * 9/16 in. *
Inverter "A" or "B", 2500 VA	Leland MGE-23-2	AC (Straight) SA-9238 AC (Angle) SA-9236 DC A-10372	2 2 4		7/16 in. * 7/16 in. * 9/16 in. *
Prop. Hub Sw	Rotol	RA.63813	3		0.050 from bottom of "U"
Prop. Deicing	Rotol	RA.47387/I	15		0.25 from face of brush block

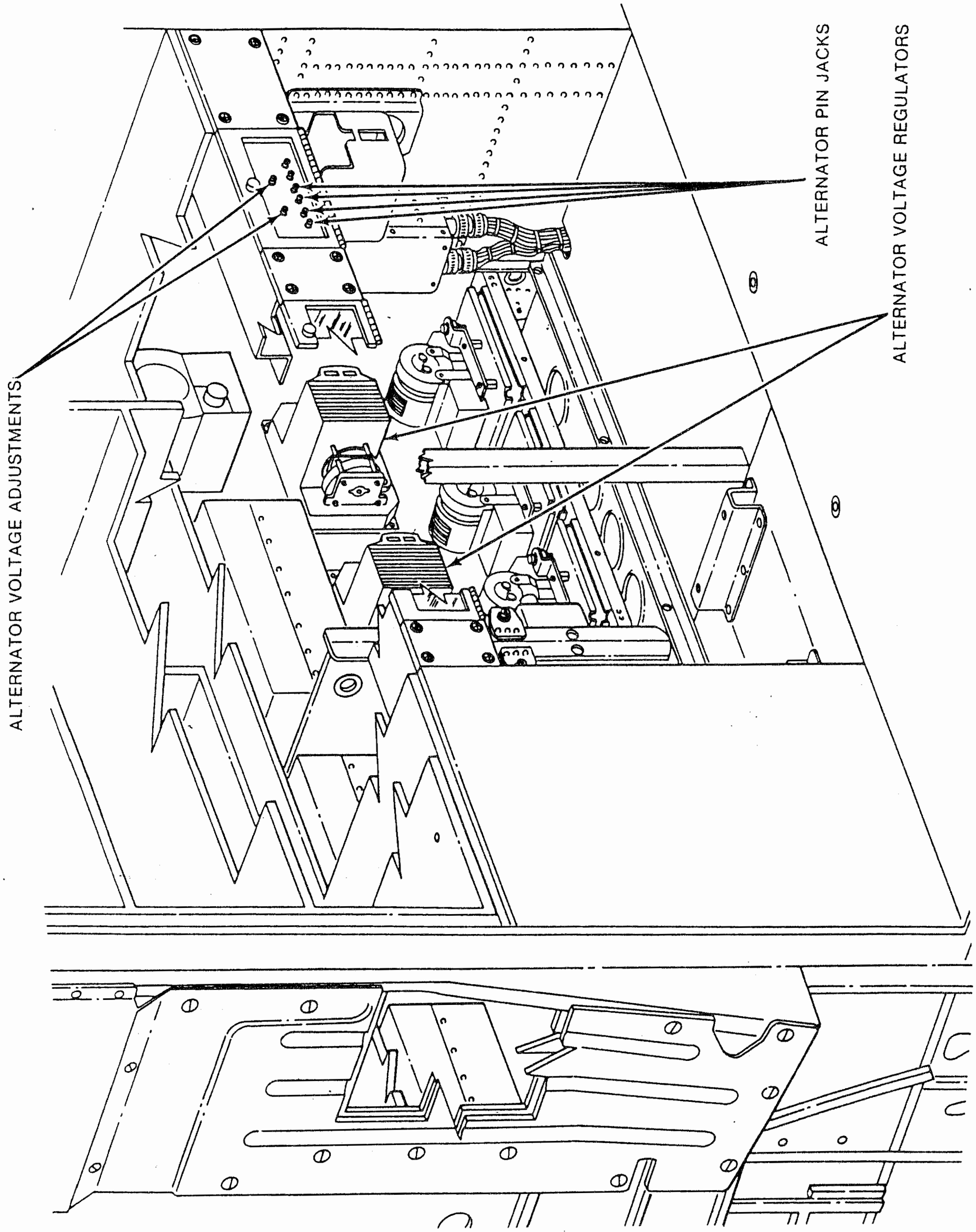
NOTE: Items marked with * are provided with wear marks. Dimensions given should agree with wear marks as to point of replacement.

The data presented herein is a summary of available information concerning brushes in the Gulfstream electrical equipment.

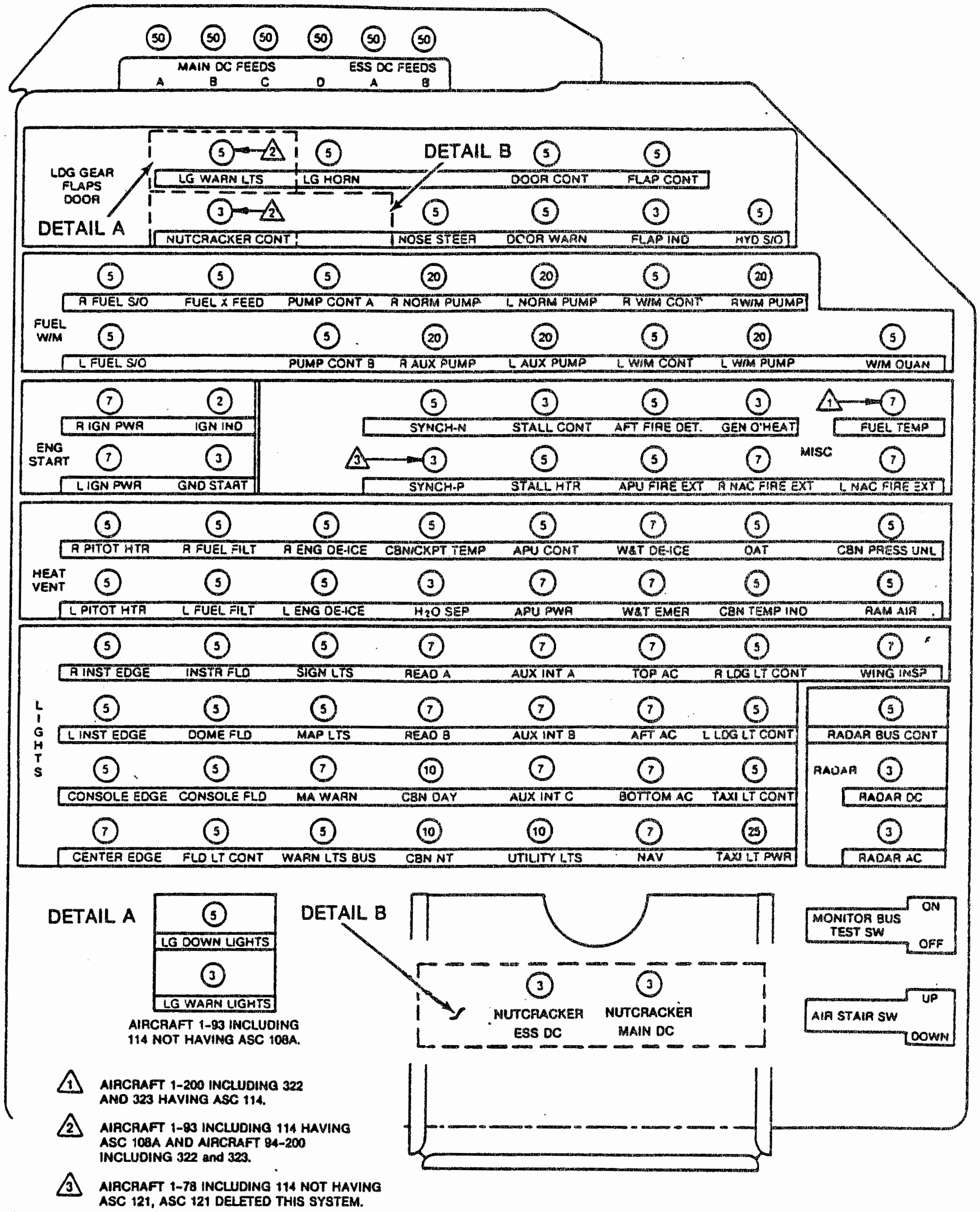
This data is not intended to supersede manufacturer recommendations, but is simply a means of collecting and publishing such facts as are available.



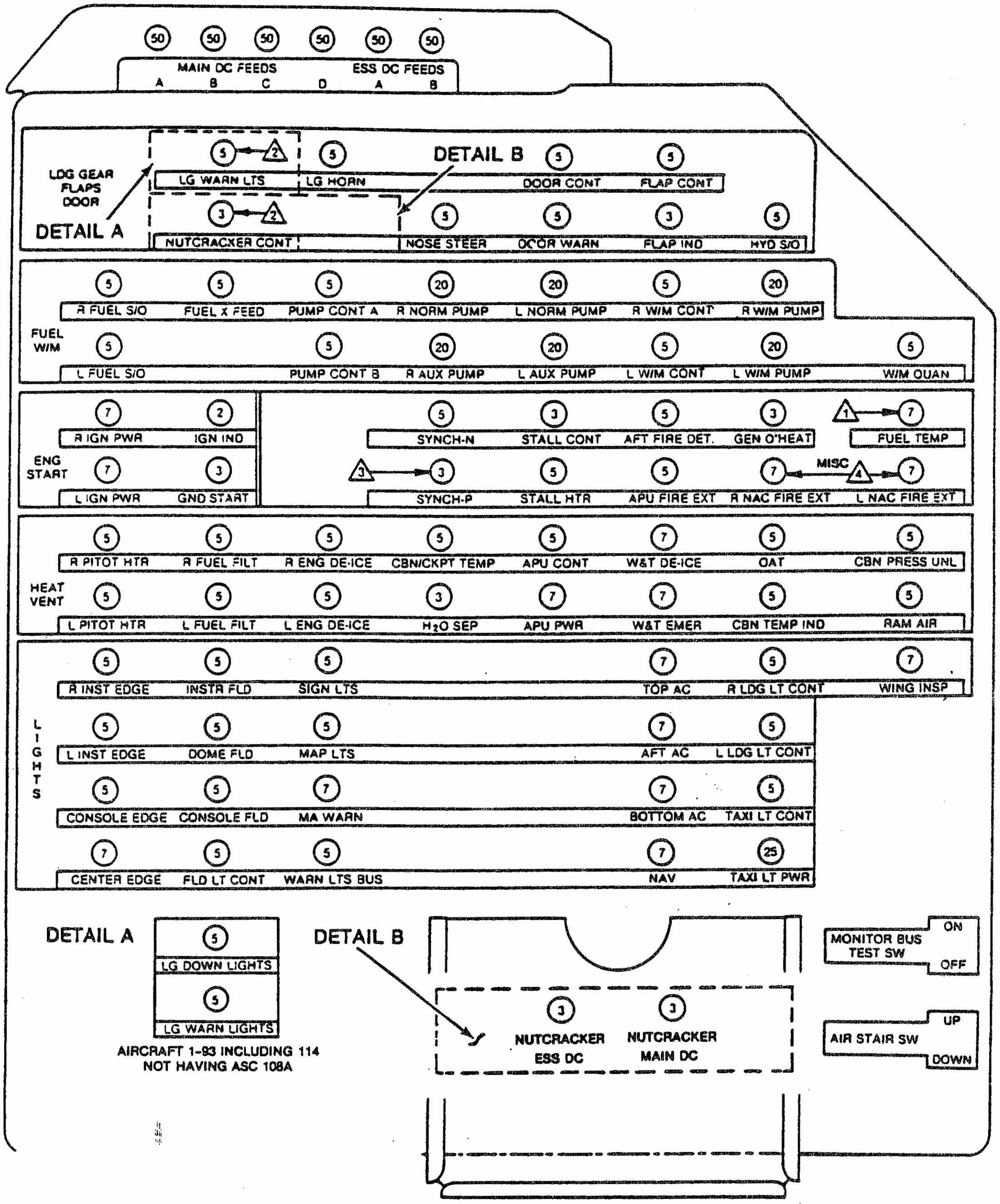
Equipment Location — Fuselage Station 193 Terminal Panel
Figure 3.



Inverter Relay Panel — Main Electrical Power Distribution
Figure 4.

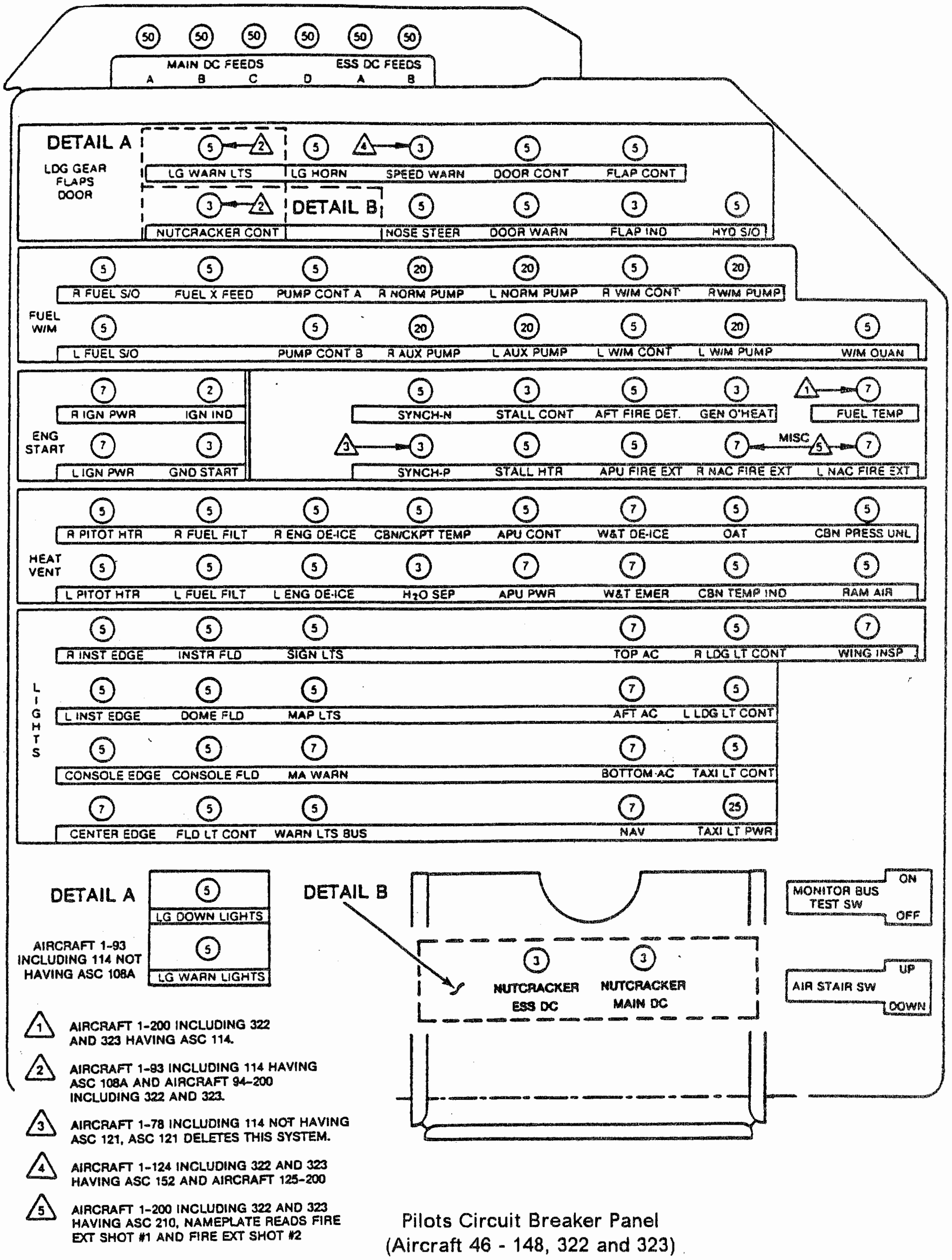


Pilots Circuit Breaker Panel
(Aircraft 1, 2, 3, and 9)
Figure 5.

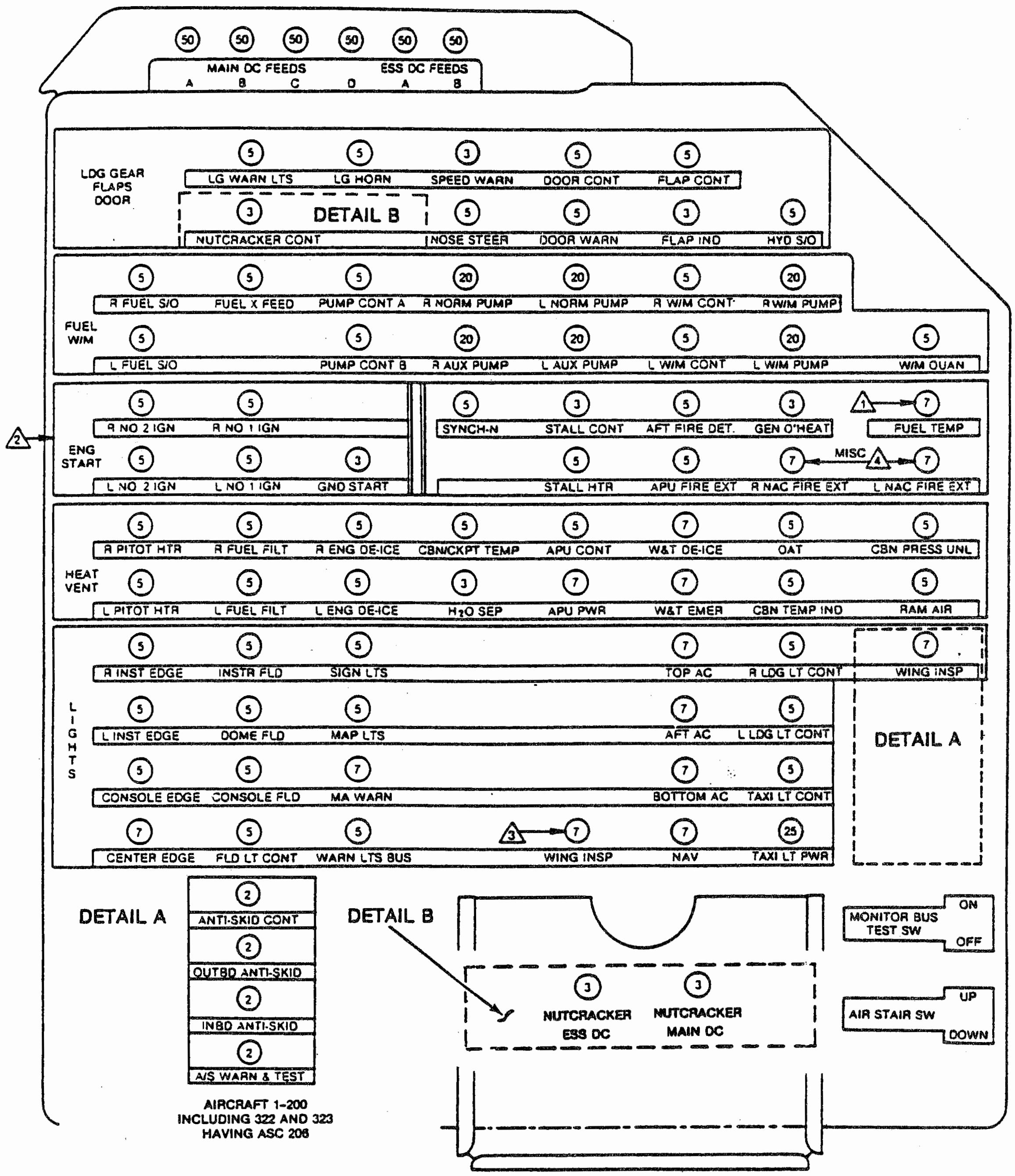


1. AIRCRAFT 1-200 INCLUDING 322 AND 323 HAVING ASC 114.
2. AIRCRAFT 1-93 INCLUDING 114 HAVING ASC 108A AND AIRCRAFT 94-200 INCLUDING 322 and 323.
3. AIRCRAFT 1-78 INCLUDING 114 NOT HAVING ASC 121, ASC 121 DELETED THIS SYSTEM.
4. AIRCRAFT 1-200 INCLUDING 322 AND 323 HAVING ASC 210 THE NAMEPLATE HAS BEEN CHANGED TO READ FIRE EXT SHOT #1 AND FIRE EXT SHOT #2

Pilots Circuit Breaker Panel
(Aircraft 4 - 8 and 10 - 45 and 114)
Figure 6.

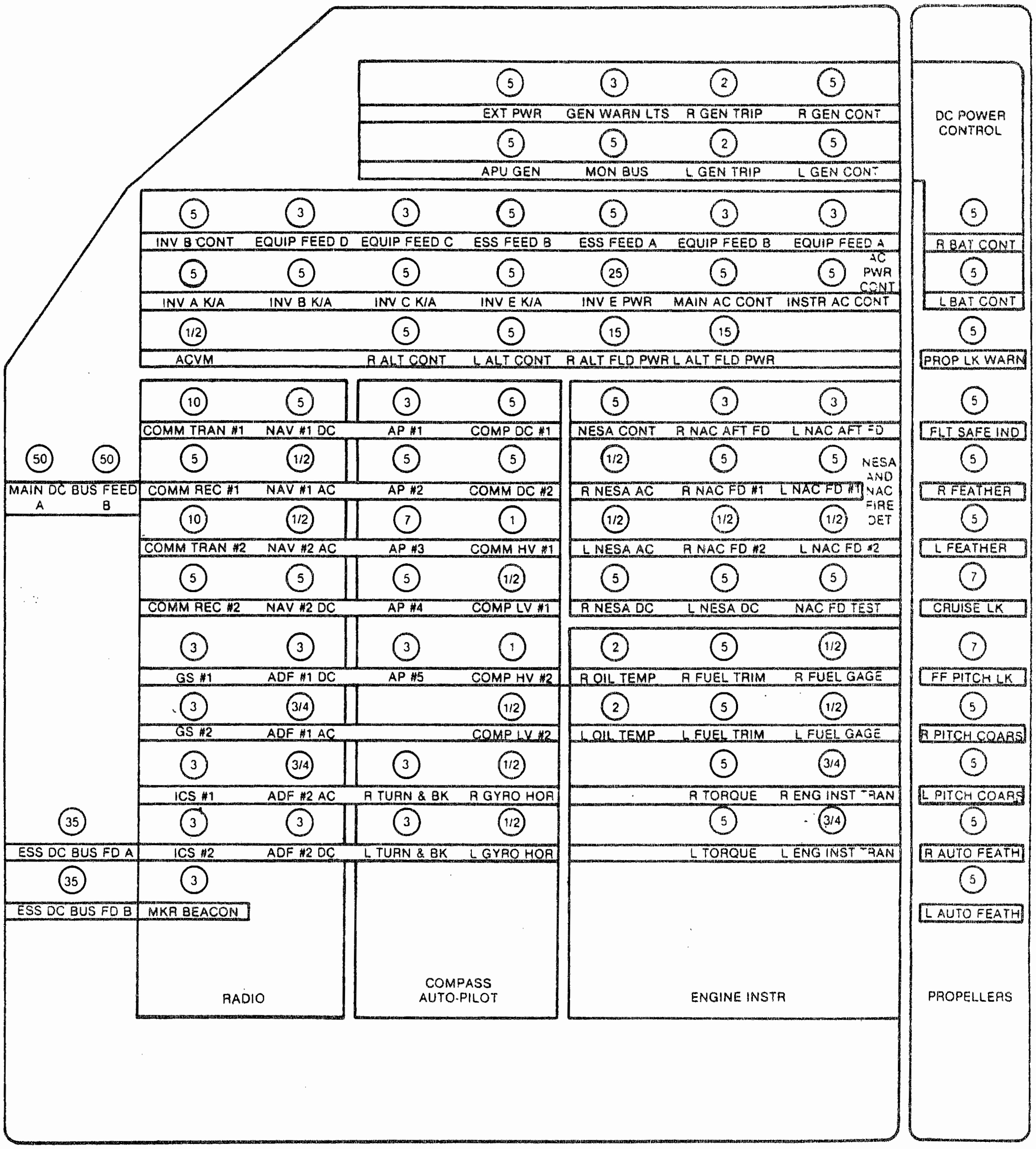


Pilots Circuit Breaker Panel
(Aircraft 46 - 148, 322 and 323)
Figure 7.

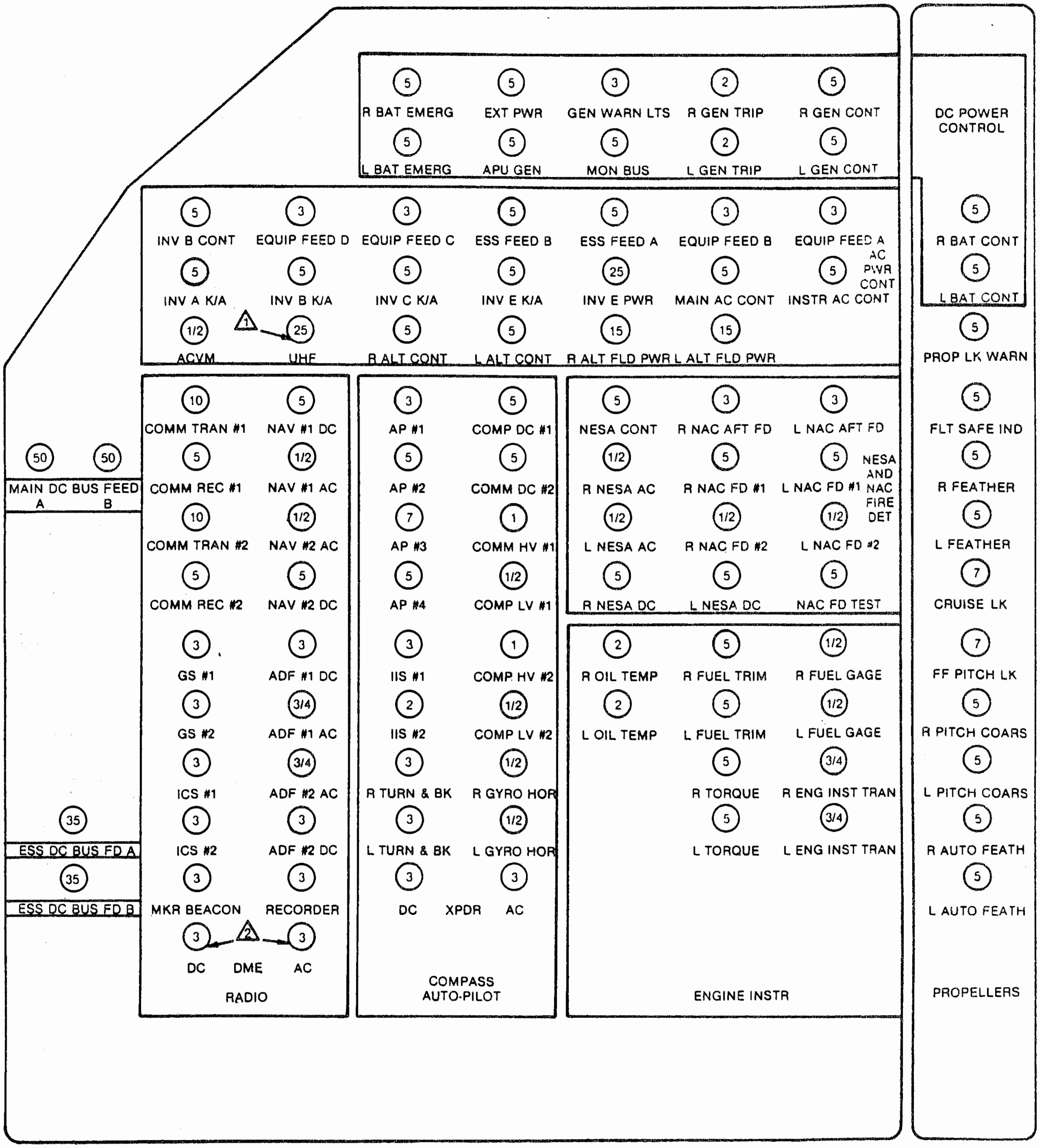


Pilots Circuit Breaker Panel
(Aircraft 149 - 200)
Figure 8.

1. AIRCRAFT 1-200 INCLUDING 322 AND 323 HAVING ASC 114.
2. AIRCRAFT 1-148 INCLUDING 322 AND 323 HAVING ASC 170A AND AIRCRAFT 149-200
3. AIRCRAFT 1-200 INCLUDING 322 AND 323 HAVING ASC 206
4. AIRCRAFT 1-200 INCLUDING 322 AND 323 HAVING ASC 210 NAMEPLATE READS FIRE EXT SHOT #1 AND FIRE EXT SHOT #2



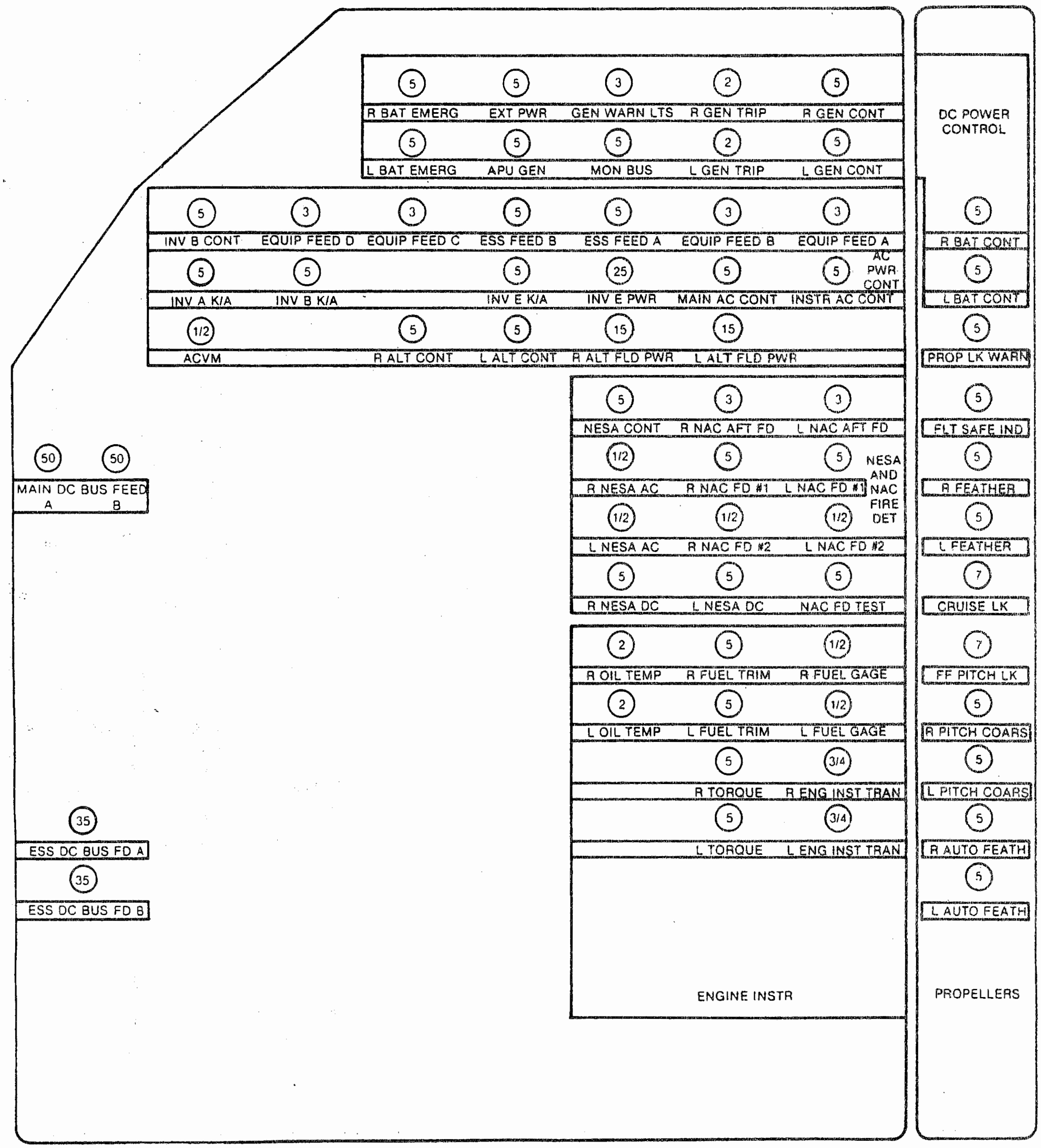
Copilots Circuit Breaker Panel
(Aircraft 2 only)
Figure 9.



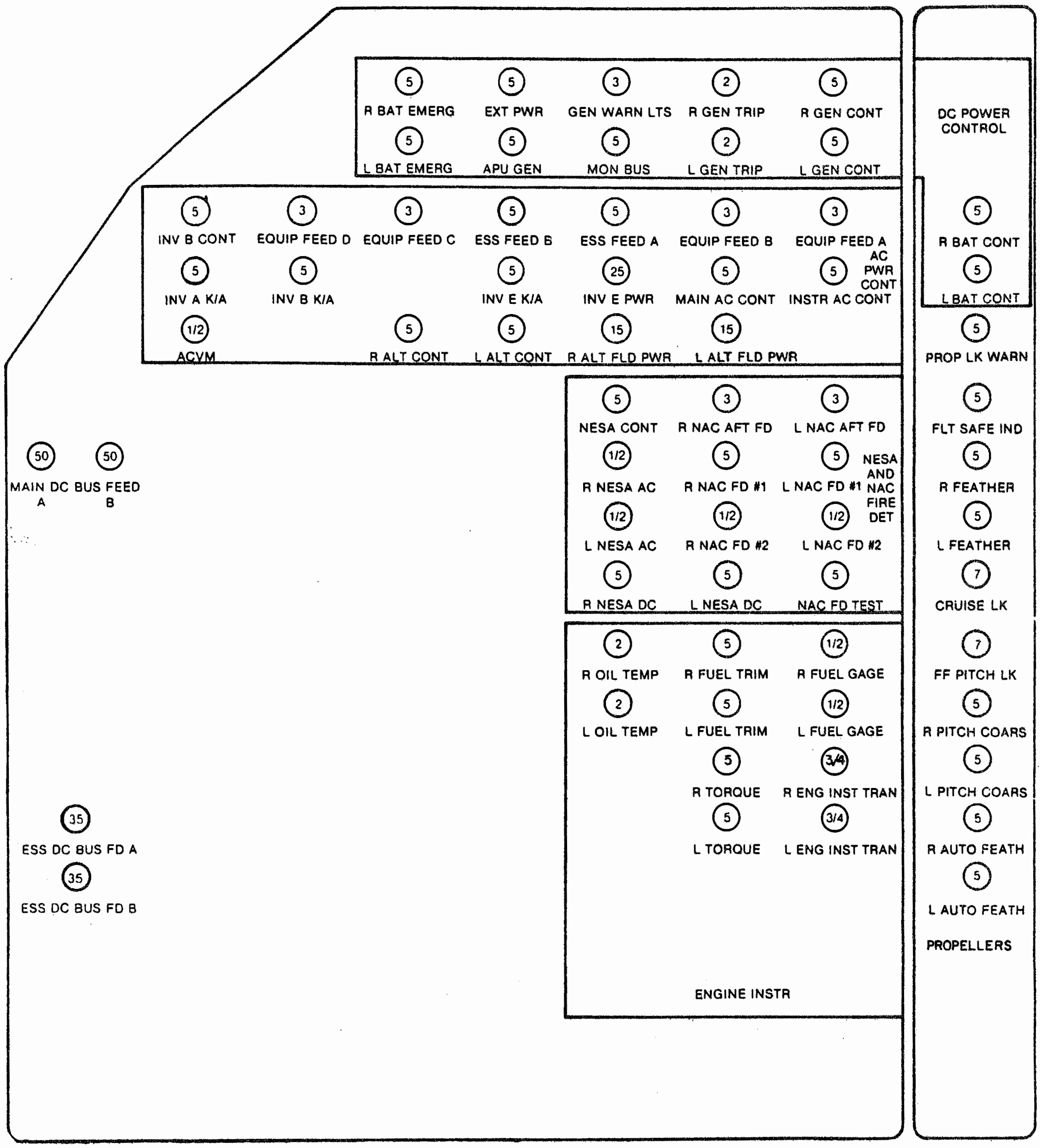
1 AIRCRAFT NO. 1 ONLY.

2 AIRCRAFT NO. 9 ONLY.

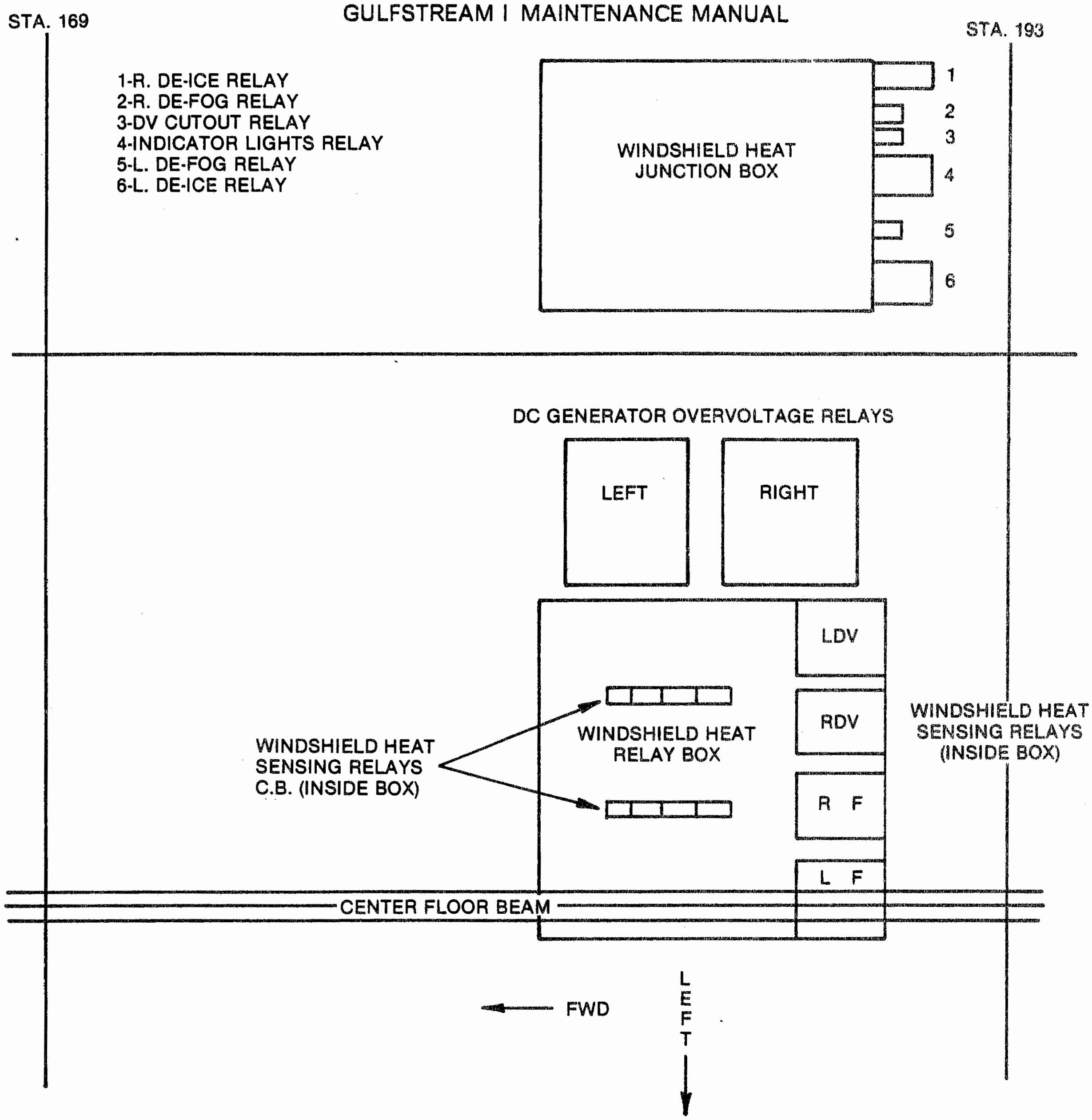
Copilots Circuit Breaker Panel
(Aircraft 1, 3 and 9)
Figure 10.



Copilots Circuit Breaker Panel
(Aircraft 4 - 8 and 10 - 45 and 114)
Figure 11.

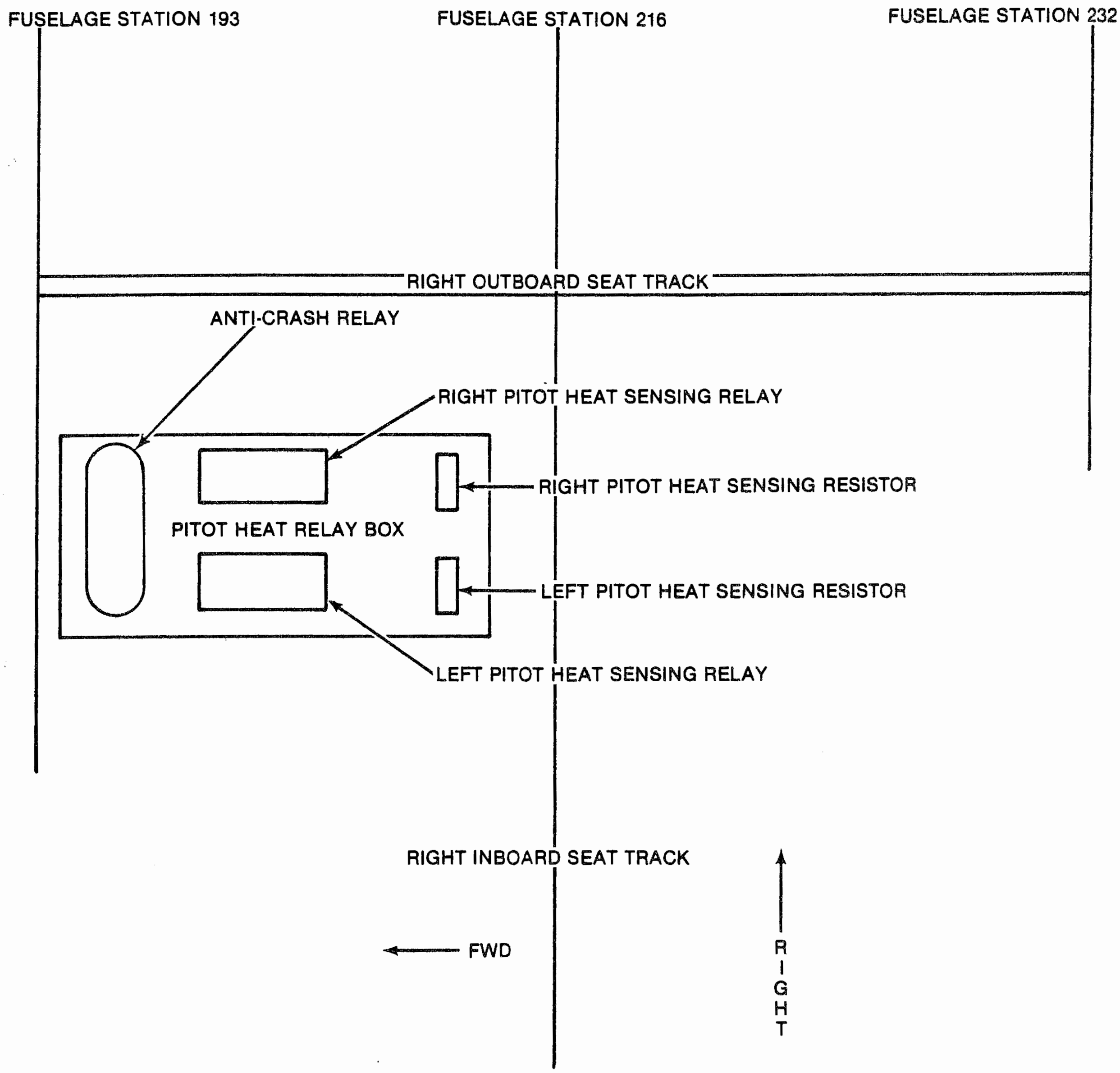


Copilots Circuit Breaker Panel
(Aircraft 46 - 200, 322 and 323 excluding 114)
Figure 12.



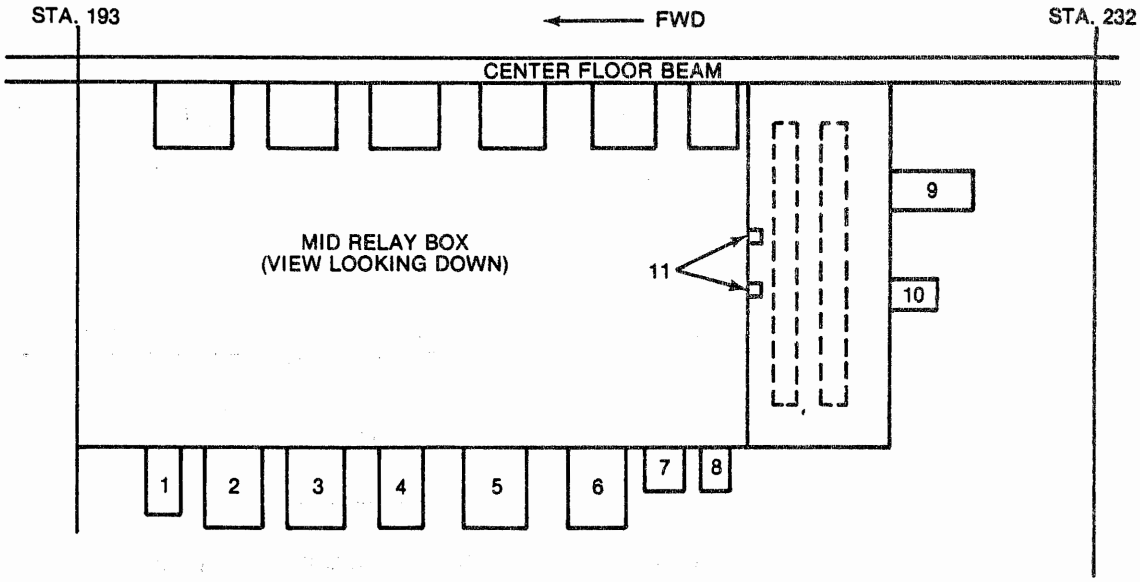
COMPARTMENT: ELECTRONICS
STATIONS: 169-193 UNDER FLOOR
VIEW: LOOKING DOWN
ACCESS: WINDSHIELD HEAT JUNCTION BOX-REMOVE FLOORBOARD #4
WINDSHIELD HEAT RELAY BOX AND
DC GENERATOR OVERVOLTAGE RELAYS-REMOVE FLOORBOARD #2

Equipment Location — Fuselage Station 169 - 193 Under Floor
Figure 13.

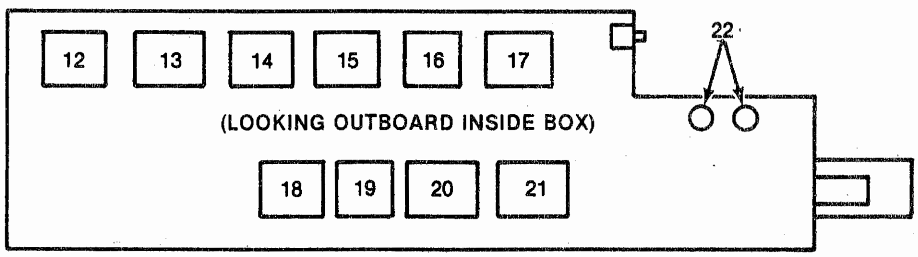


COMPARTMENT: FORWARD CABIN
STATIONS: 193 to 216 UNDER FLOOR
VIEW: LOOKING DOWN-RIGHT SIDE OF CENTER
ACCESS: ANTI-CRASH RELAY AND PITOT HEAT RELAY BOX-REMOVE FLOORBOARD #6

Equipment Location — Fuselage Station 193 - 216 Under Floor
Figure 14.



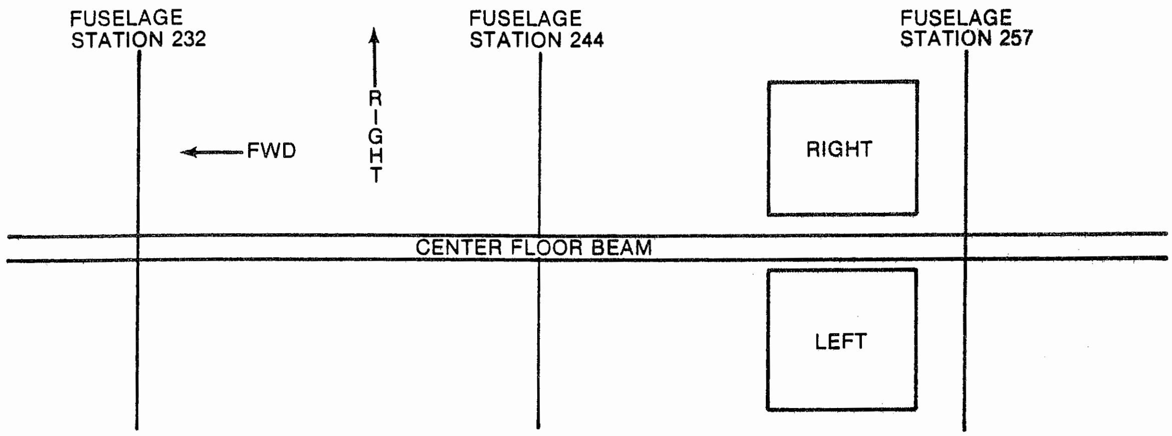
1-L. DEICE POWER SENSE RELAY
2-L. WATER METHANOL VALVE RELAY
3-R. WATER METHANOL VALVE RELAY
4-FLAP CONTROL RELAY
5-#1 NUTCRACKER RELAY
6-#2 NUTCRACKER RELAY
7-SUPERCHARGER PROTECTION RELAY
(AIRCRAFT 1 THROUGH 60 INCLUDING
114 HAVING ASC 81 INCORPORATED
AND AIRCRAFT 61 THROUGH 200
INCLUDING 322 AND 323, EXCLUDING 114)
8-R. DEICE POWER SENSE RELAY
9-FWD FIRE DETECTOR TEST RELAY
10-DOOR INSIDE CONTROL RELAY
11-DC VOLT -AMMETER C.B.s



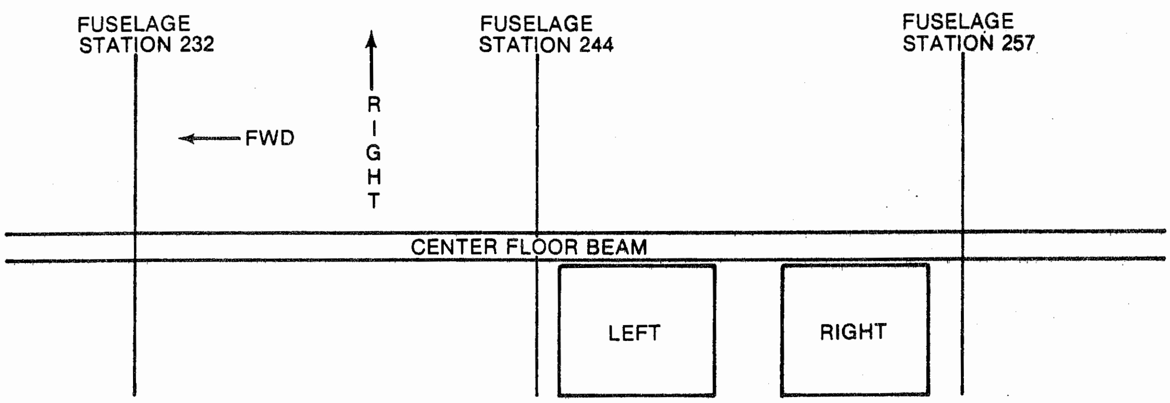
12-L. ALTERNATOR FIELD POWER RELAY
13-R. ALTERNATOR FIELD POWER RELAY
14-R. NORMAL FUEL PUMP RELAY
15-R. AUXILIARY FUEL PUMP RELAY
16-L. NORMAL FUEL PUMP RELAY
17-L. AUXILIARY FUEL PUMP RELAY
18-HYDRAULIC PUMP RELAY
19-TAXI LIGHT RELAY
20-L. WATER METHANOL PUMP RELAY
21-R. WATER METHANOL PUMP RELAY
22-ALTERNATOR FIELD BIAS RESISTORS (UNDER COVER)

COMPARTMENT: FORWARD CABIN
VIEW: AS SHOWN ABOVE
STATION: 193 TO 232
ACCESS: MID RELAY BOX-REMOVE FLOOR-BOARD #8

Equipment Location — Fuselage Station 193 - 232 Under Floor
Figure 15.



ALTERNATOR FIELD POWER TRANSFORMER—RECTIFIERS — LOCATION — AIRCRAFT 4 & 5 ONLY

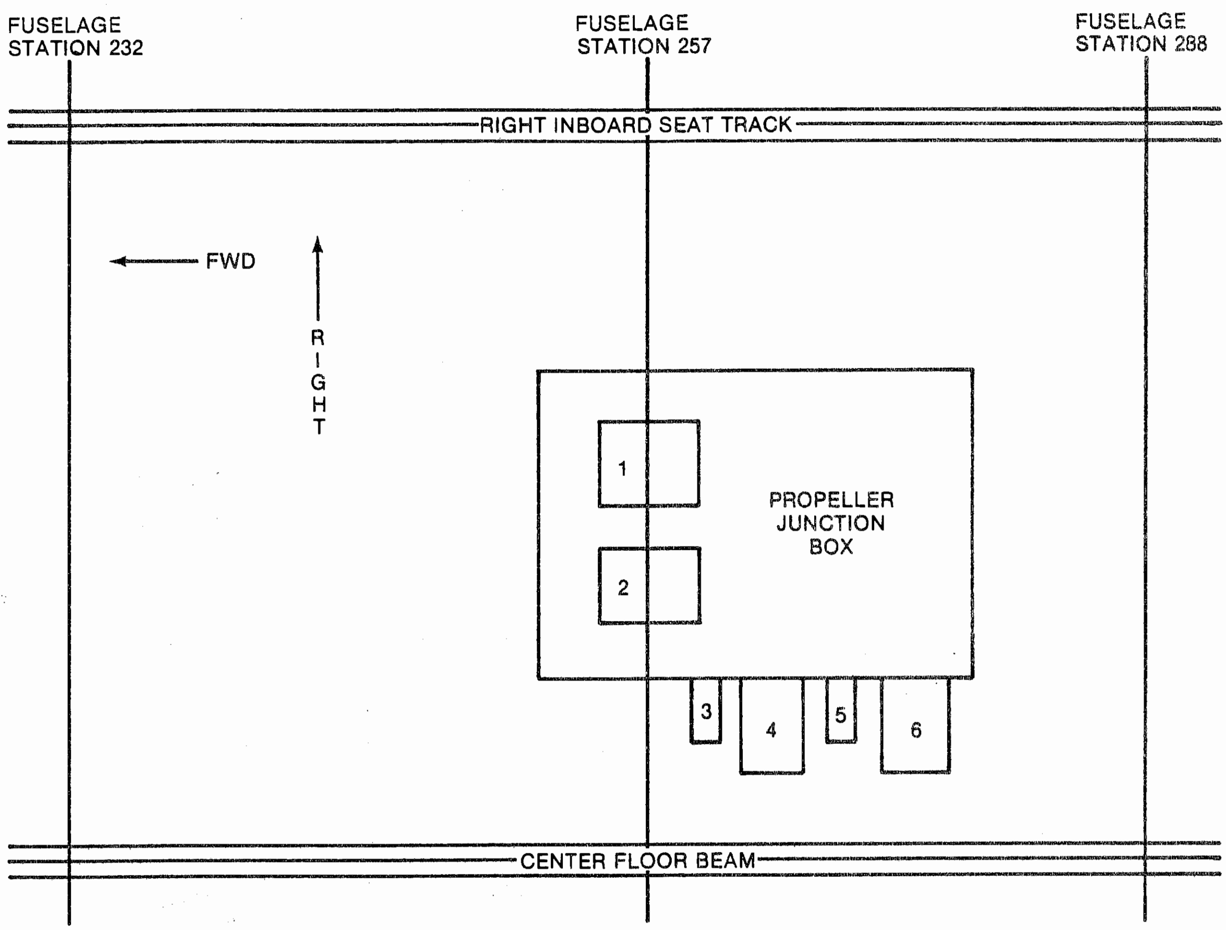


ALTERNATOR FIELD POWER TRANSFORMER - RECTIFIER -
LOCATION - AIRCRAFT 6 THRU 200 INCLUDING 322 & 323

COMPARTMENT: FORWARD CABIN
VIEW: LOOKING DOWN
ACCESS: AIRCRAFT 4 & 5 — RIGHT TRANSFORMER—RECTIFIER — REMOVE FLOORBOARD #11
LEFT TRANSFORMER—RECTIFIER — REMOVE FLOORBOARD #12

AIRCRAFT 6 THRU 200 INCLUDING 322 & 323 - LEFT AND
RIGHT TRANSFORMER - RECTIFIERS - REMOVE FLOORBOARD #12

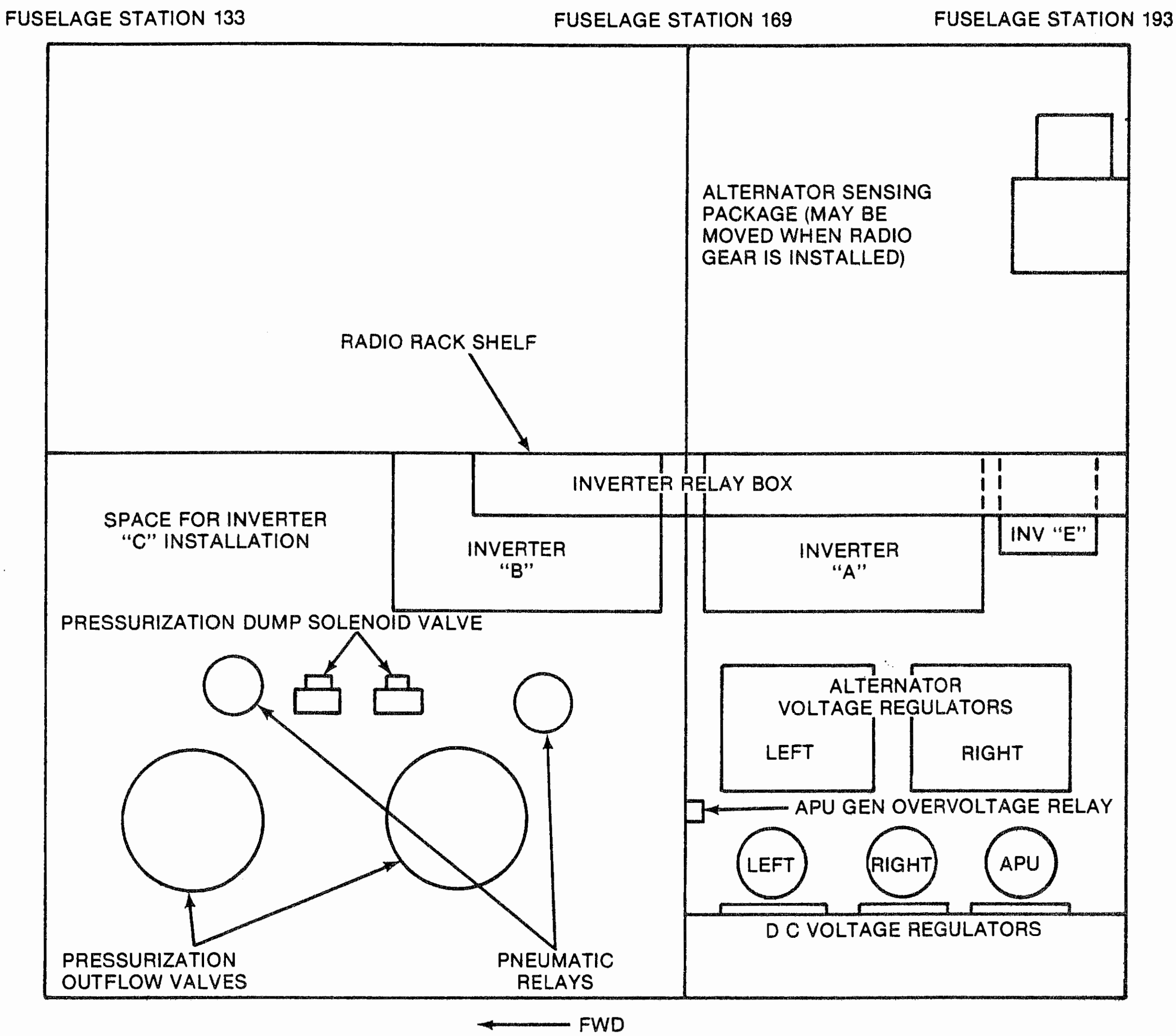
Equipment Location — Fuselage Station 232 - 257 Under Floor
Figure 16.



1. PROPELLER RELAY "X"
2. PROPELLER RELAY "Z"
3. RIGHT PROP. ISOLATING RELAY
4. LEFT PROP. CONTROL RELAY
5. LEFT PROP. ISOLATING RELAY
6. RIGHT PROP. CONTROL RELAY

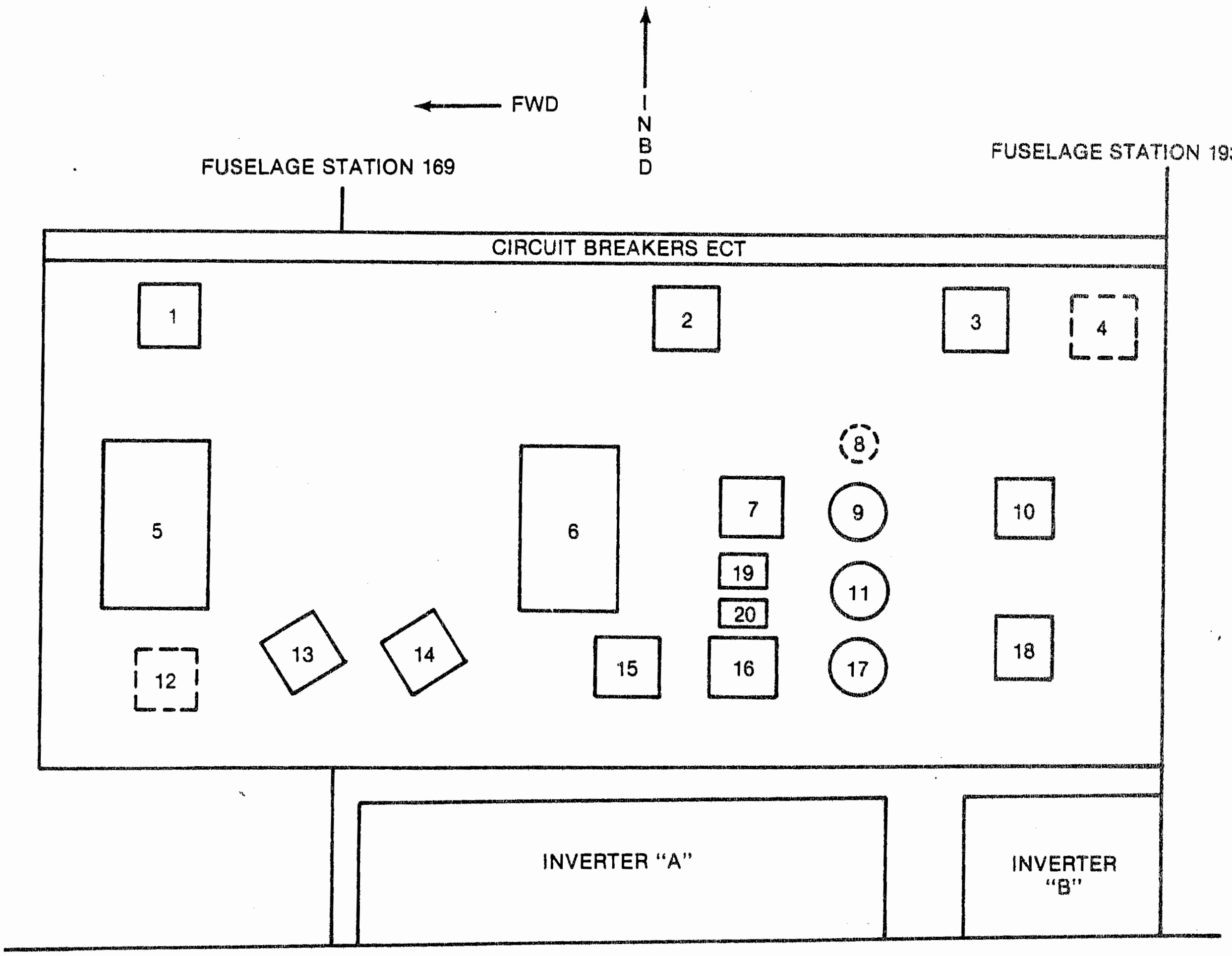
COMPARTMENT: FORWARD CABIN
VIEW: LOOKING DOWN
ACCESS: PROPELLER JUNCTION BOX — REMOVE FLOORBOARD #11

Equipment Location — Fuselage Station 232 - 288 Under Floor
Figure 17.



COMPARTMENT: ELECTRONICS
VIEW: LOOKING OUTBOARD
ACCESS: REMOVE TWO VERTICAL PANELS UNDER THE RADIO RACK SHELF

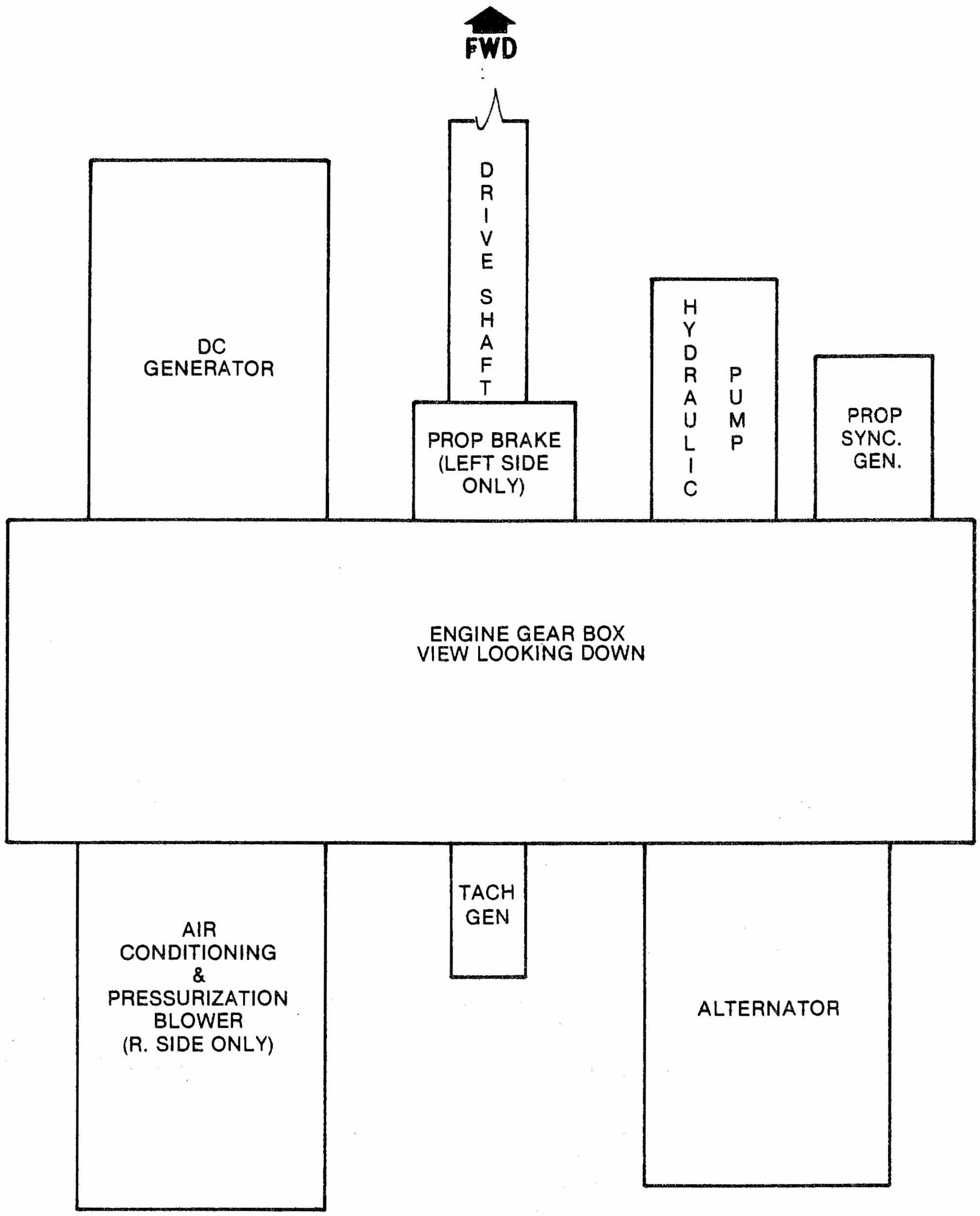
Equipment Location — Fuselage Station 133 - 193 (Right Side)
Figure 18.



VIEW: LOOKING UPWARD AT UNDERSIDE OF BOX
ACCESS: REMOVE TWO PANELS UNDERSIDE OF BOX

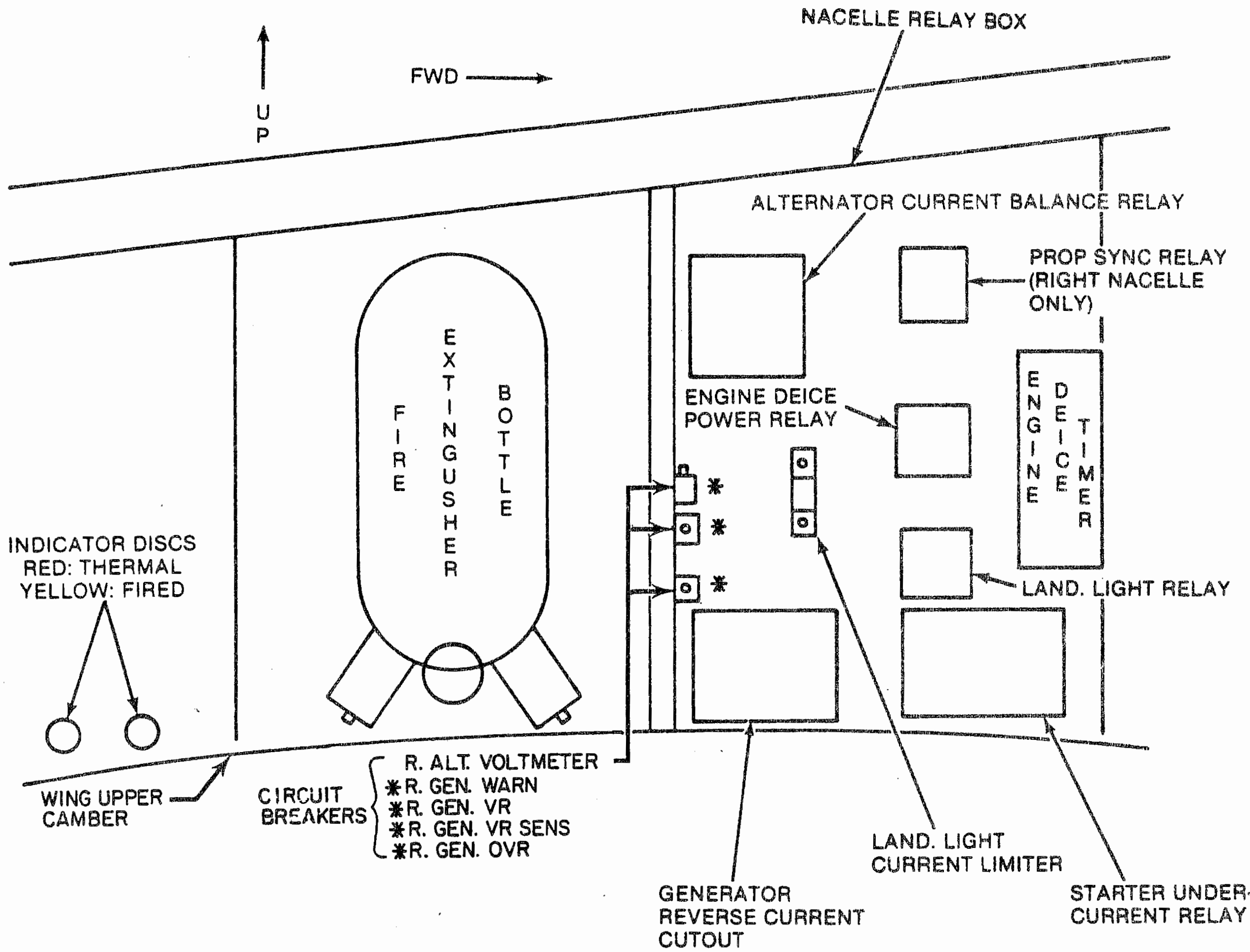
1 - INVERTER RELAY "B-2"
2 - MONITOR BUS RELAY
3 - INVERTER RELAY "A-1"
4 - INVERTER RELAY "B-3" (SPACE PROV. ONLY)
5 - WINDSHIELD POWER TRANSFER RELAY
6 - INVERTER RELAY "E-2"
7 - INVERTER RELAY "E-1"
8 - INVERTER RELAY "B-4" (SPACE PROV. ONLY)
9 - EQUIPMENT AC BUS SENSE RELAY
10 - NORMAL DC BUS TIE #1 RELAY
11 - INSTRUMENT AC BUS SENSE RELAY
12 - INVERTER RELAY "C-1" (SPACE PROV. ONLY)
13 - NORMAL DC BUS TIE RELAY # 2 (AIRCRAFT 1 THROUGH 17 INCLUDING 114.) INVERTER RELAY "B-1" (AIRCRAFT 18 THROUGH 200 INCLUDING 322 AND 323, EXCLUDING 114.)
14 - INVERT RELAY "B-1" (AIRCRAFT 1 THROUGH 17, INCLUDING 114.) NORMAL DC BUS TIE RELAY (AIRCRAFT 18 THROUGH 200, INCLUDING 322 AND 323.)
15 - RIGHT FEATHER PUMP RELAY
16 - LEFT FEATHER PUMP RELAY
17 - MAIN AC BUS SENSE RELAY
18 - EMERGENCY DC BUS TIE RELAY
19 - LEFT STARTER INTERLOCK RELAY
20 - RIGHT STARTER INTERLOCK RELAY

Equipment Location — Inverter Relay Box
Figure 19.



ACCESS: REMOVE PANEL ON TOP OF NACELLE

Equipment Location — Engine Gearbox
Figure 20.

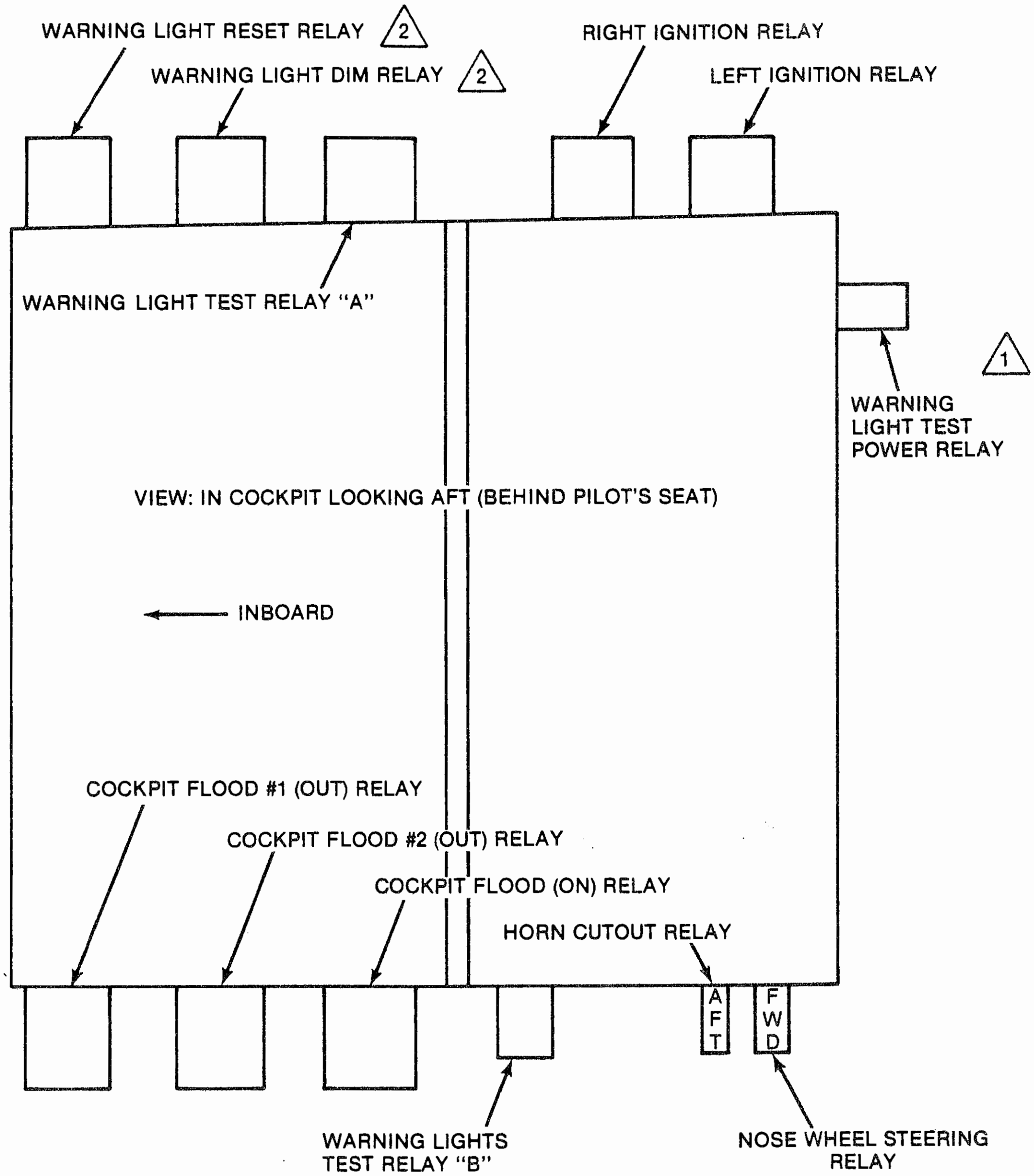


NOTE: FIRE EXTINGUISHER BOTTLE AND NACELLE RELAY BOX ARE LOCATED:
RIGHT NACELLE: OUTBOARD SIDE (SEE ABOVE)
LEFT NACELLE: INBOARD SIDE

ACCESS: REMOVE PANEL FROM SIDE OF NACELLE

* AIRCRAFT 1 THROUGH 200 INCLUDING 322 AND 323 HAVING ASC 231 INCORPORATED.

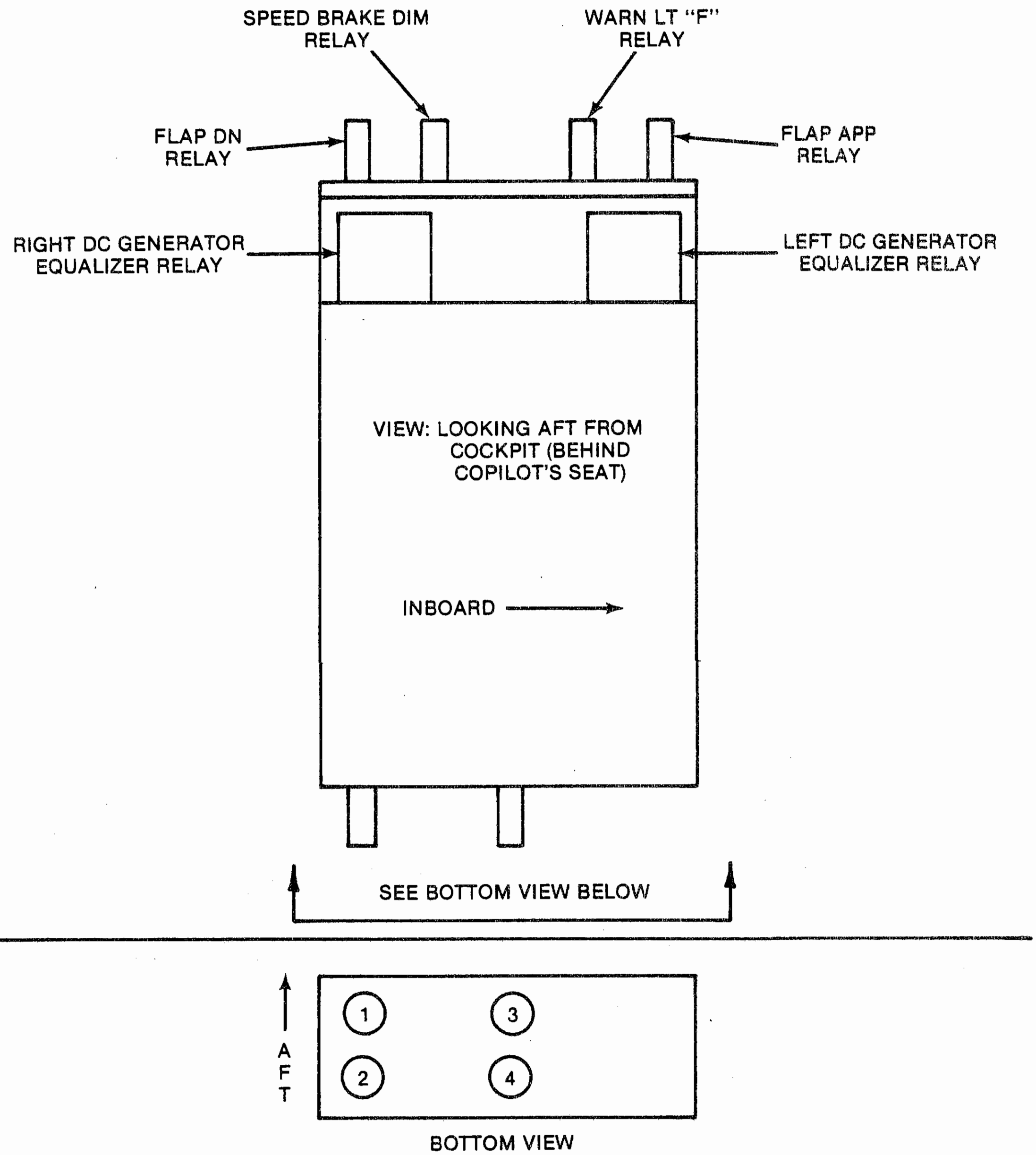
Equipment Location — Right Nacelle Relay Box
Figure 21.



1. INSTALLED ON AIRCRAFT 1 THROUGH 120 ONLY, INCLUDING 322 AND 323.

2. AIRCRAFT 1 THROUGH 50 INCLUDING 114 HAVING ASC 51 INCORPORATED AND AIRCRAFT 51 THROUGH 200 INCLUDING 322 AND 323.

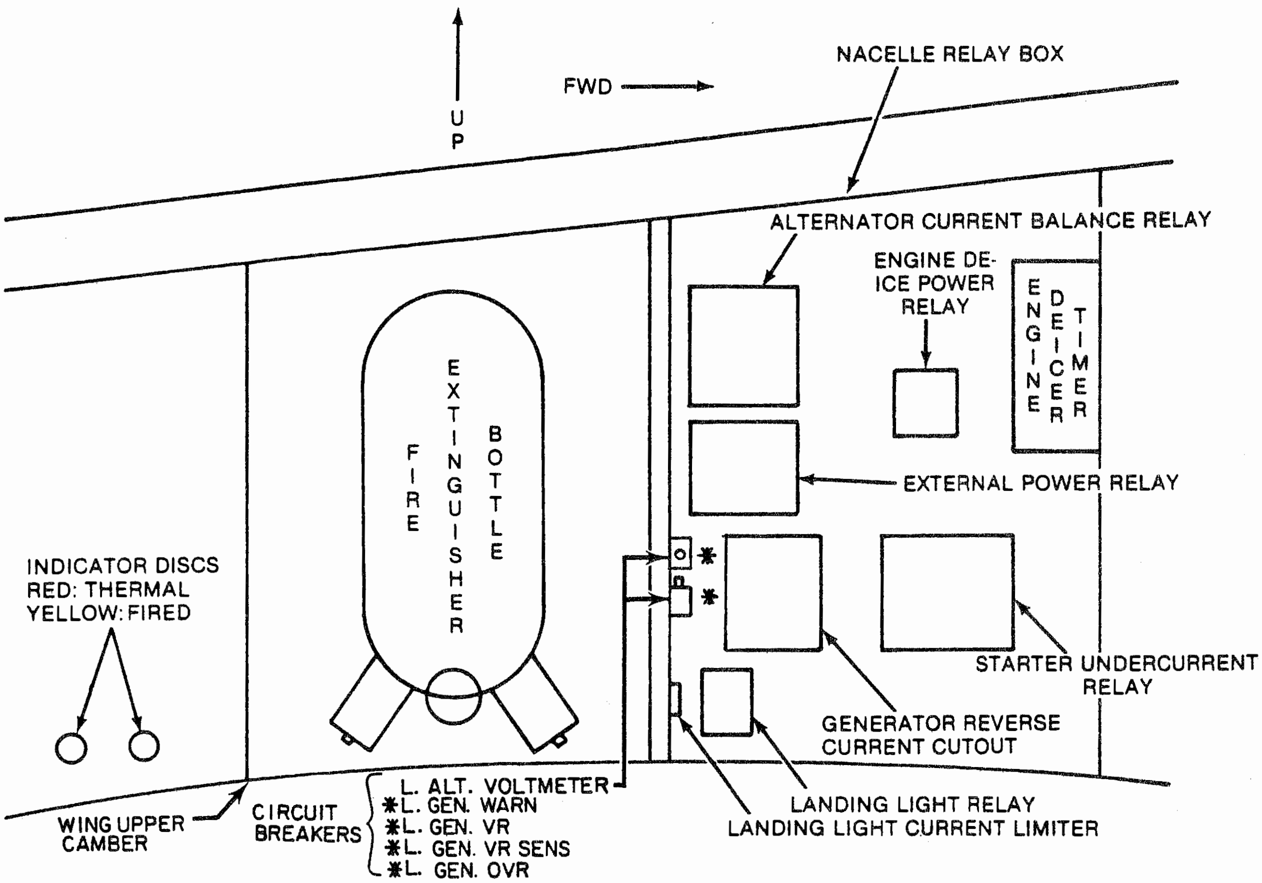
Equipment Location — Fuselage Station 133 Forward Side Pilots Relay Box
Figure 22.



1. RIGHT DC BUS CONTROL RELAY
2. LEFT DC BUS CONTROL RELAY
3. FUEL GAGE TEST RELAY
4. APU GENERATOR WARNING RELAY

NOTE: THIS DIAGRAM INCORPORATES ASC #121 (DELETION OF SYNC PILOT RELAY)
THIS DIAGRAM INCORPORATES ASC #108A (ADDITIONAL WARNING PROVISIONS)
PART I ONLY

Equipment Location — Fuselage Station 133 Forward Side Copilots Relay Box
Figure 23.

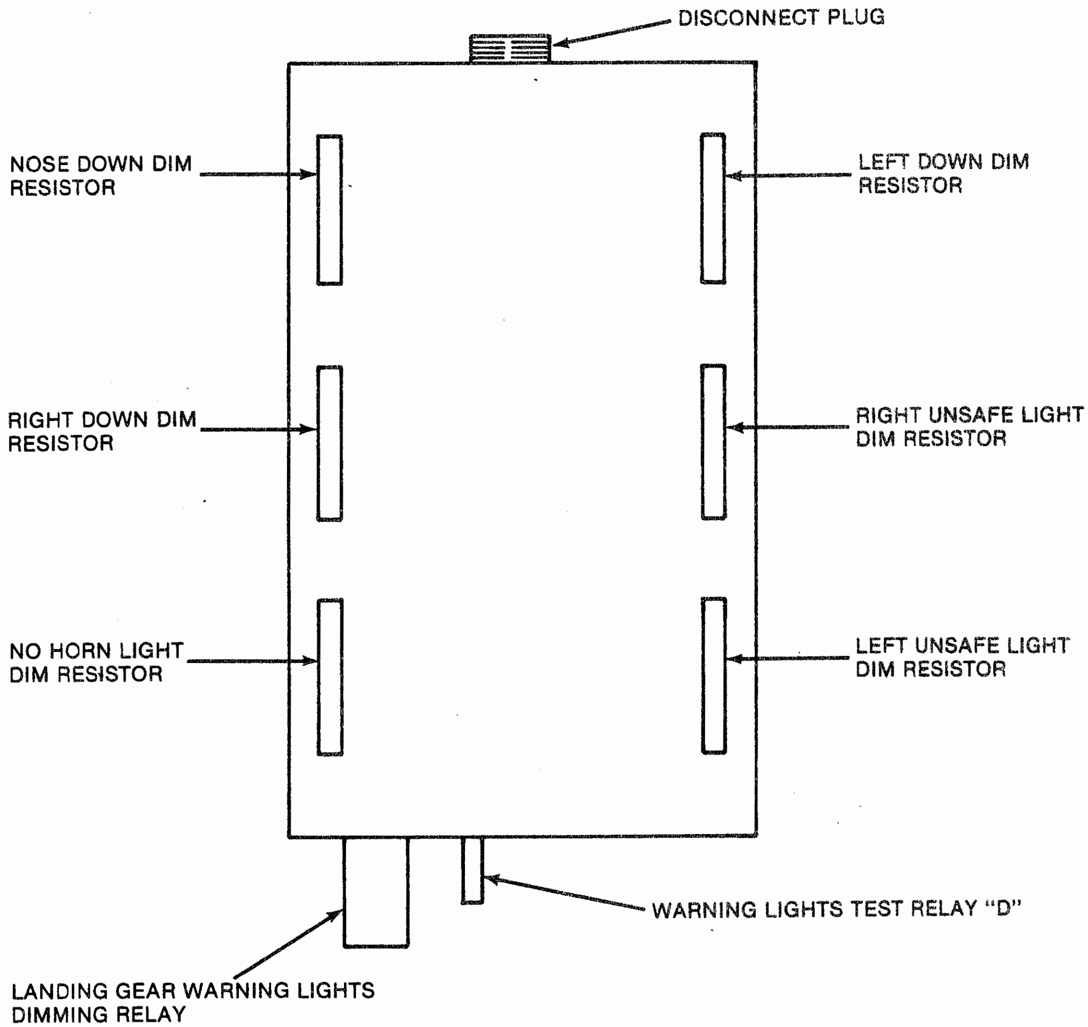


NOTE: FIRE EXTINGUISHER BOTTLES AND NACELLE RELAY BOXES ARE LOCATED:
RIGHT NACELLE: OUTBOARD SIDE
LEFT NACELLE: INBOARD SIDE (SEE ABOVE)

ACCESS: REMOVE PANEL FROM SIDE OF NACELLE

* AIRCRAFT 1 THROUGH 200 INCLUDING 322 AND 323 HAVING ASC 231 INCORPORATED.

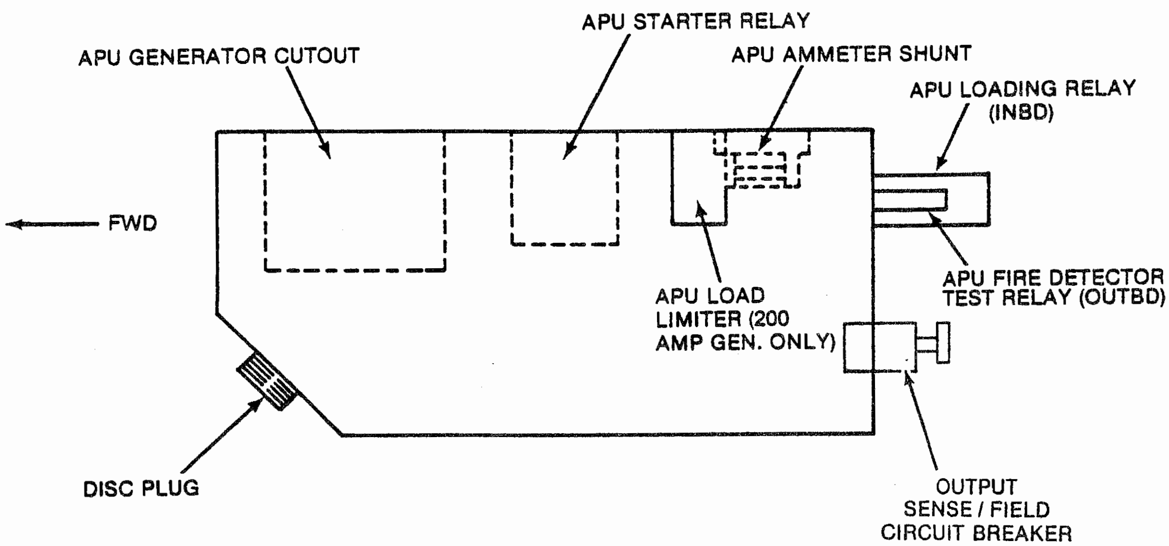
Equipment Location — Left Nacelle Relay Box
Figure 24.



NOTE: ALL RESISTORS ARE LOCATED INSIDE BOX.
LOCATION: NOSE WHEEL WELL LEFT SIDE, BETWEEN FUSELAGE STATION 85 AND 90.

ACCESS: OPEN NOSE GEAR CLAMSHELL DOORS AND REMOVE COVER FROM RESISTOR BOX.

Equipment Location — Landing Gear Resistor Box
Figure 25.

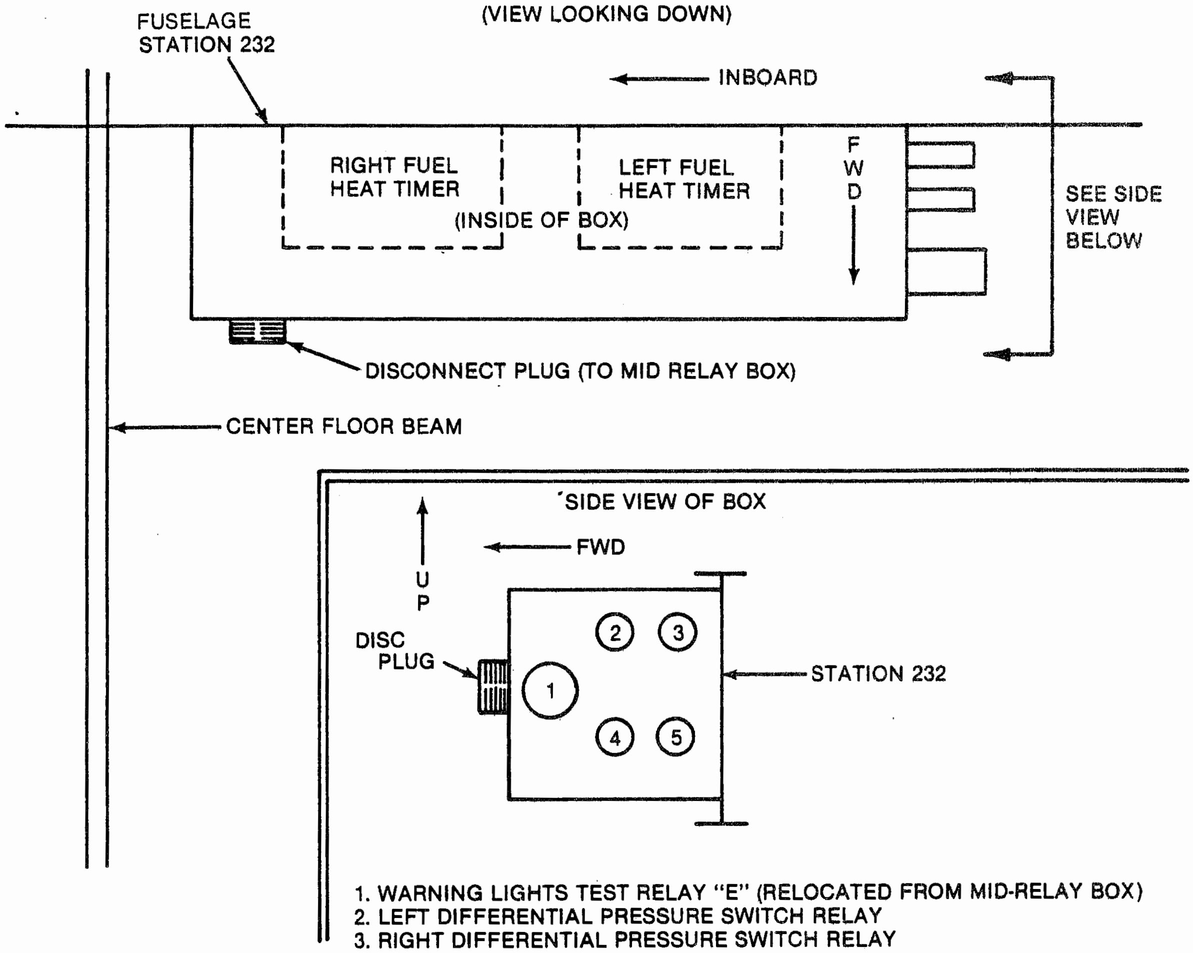


VIEW: LOOKING INBOARD, BOX MOUNTED ON AFT SIDE OF APU CONTAINER.

ACCESS: FROM OUTSIDE OF AIRCRAFT — LEFT SIDE OF AFT FUSELAGE, DUAL HINGED ACCESS DOOR. BETWEEN FUSELAGE STATIONS 580 - 610.

NOTE: ALL DOTTED ITEMS ARE LOCATED INSIDE OF BOX.

Equipment Location — APU Relay Box
Figure 26.



1. WARNING LIGHTS TEST RELAY "E" (RELOCATED FROM MID-RELAY BOX)
2. LEFT DIFFERENTIAL PRESSURE SWITCH RELAY
3. RIGHT DIFFERENTIAL PRESSURE SWITCH RELAY
4. LEFT FUEL FILTER HEAT RELAY
5. RIGHT FUEL FILTER HEAT RELAY

COMPARTMENT: FORWARD CABIN
STATIONS: FUSELAGE STATION 232 — LEFT SIDE UNDER FLOOR
VIEW: AS INDICATED
ACCESS: REMOVE FLOORBOARD #8

NOTE: THIS DIAGRAM IS ONLY EFFECTIVE AFTER THE INSTALLATION OF ASC. 117
(FUEL FILTER HEAT TIMERS)

Equipment Location — Fuel Filter Heat Timer Box
(Aircraft 1 - 99 and 114 having ASC 114 and Aircraft 100 - 200, 322 and 323)
Figure 27.

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DC POWER SUPPLY AND DISTRIBUTION SYSTEM — DESCRIPTION / OPERATION

1. General

The engine-driven dc generators provide the normal source of electrical power for the three-bus, dc power distribution system. Each generator is rated at 300 amps, 30 volts at 4000-8000 generator rpm. The generators are paralleled, and a bus equalizing system is incorporated. Each generator system includes a reverse current cutout, a carbon pile voltage regulator, an overvoltage relay, a bus equalizing relay, a generator control switch, a generator trip switch , a generator failure warning light, a generator overheat warning light, a volt-ammeter and a shunt. The left and right generator controls and volt-ammeters are placarded L GEN and R GEN.

Two nickel-cadmium batteries provide a standby source of dc power for the aircraft. During normal operation, the batteries are connected to the main dc bus through the left and right normal battery contactors, and to the essential dc bus through the left and right battery emergency relays. During emergency operation, the main dc bus may be isolated from the essential dc bus by means of the battery contactor, thereby removing nonessential loads from the battery. The batteries can be utilized to start the engines and the APU when external dc power is not available. Aircraft 1 - 200, 322 and 323 having ASC 194, have a battery failure monitor system which provides a means of detecting a failure (thermal runaway) in either battery. Two digital readout battery ammeters provide a continuous indication of the charge/discharge rate of each battery.

A system for connecting an external dc power source to the aircraft is provided. It includes an external power receptacle, an external power relay and an external power control switch. (See Figure 1 and Figure 2) Aircraft 1 - 200, 322 and 323 having ASC 193A contain a ground power overvoltage and reverse polarity protection relay package and an external power distribution relay.

The auxiliary power unit (APU) includes a dc generator which provides a limited source at power for battery charging, or an emergency source of power in the event of failure of both generators during flight. (Aircraft 1 - 60 and 114 were fitted with 50 amp, 27 volt APU generators. Aircraft 61 - 200, 322 and 323, are equipped) with 200 amp, 27 volt APU generators.) It can also be used for ground operations (within its limitations) when the engines are not running and external power is not available. A voltage regulator, a reverse current cutout, a generator control switch, and a generator failure indicating light are provided for this generator. This generator output is fed to the essential dc bus. This installation includes overvoltage and overload protective devices.

The three-bus dc power distribution system (main, essential, and monitor) is so arranged that all buses are energized by the engine-driven generators during normal operation. In the event of failure of both generators, the system can be brought into emergency operation. in emergency operation, the essential dc bus is disconnected from the main dc bus, then the batteries or the APU generator supply the essential dc bus, and the main dc and monitor dc bus are de-energized.

The generators and the external power receptacle are connected to the main dc bus by cables. Any equipment having large dc load requirements is fed directly from the tie cables. The left and right circuit breaker panels each contain part of the essential dc and part of the main dc bus. These buses are connected to the tie cables by feeders which have circuit breaker protection at both ends. The essential dc bus is connected to the main dc bus by two normal bus tie relays. The APU generator output is fed to the essential dc bus, while the batteries may be connected to the main and/or the essential dc bus, depending upon the mode of operation. The monitor dc bus is connected to the main dc bus by the monitor bus relay. This relay automatically de-energizès the monitor dc bus when the aircraft is on the ground and the power control levers are in the idle position, or in flight, when both generators fail.

2. Engine-Driven DC Generators Operation

(See Figure 2 thru Figure 5)

With both engines operating and both generators switches ON, each generator is connected to the main dc bus tie cables by its reverse current cutout when the generator voltage exceeds the bus voltage by 0.35 to 0.65 volt. The output of each generator is maintained at 27.7 $\pm$ 0.2 volts by its voltage regulator which acts to vary the generator field voltage to compensate for changes in engine speed and loading. Load currents of each generator are compared by the equalizing bus system which, operating through the voltage regulators, func-

tions to assist each generator to assume its proper share of the total load by varying the field voltages proportionally. Each volt-ammeter provides a continuous indication of the output voltage and current of each generator. (See Figure 6)

If any malfunction permits a generator to produce 31-33 volts, the overvoltage relay of that generators overvoltage sensing unit closes and energizes the trip relay which is an integral part of the overvoltage sensing unit. The trip relay interrupts the generator field circuit, thereby reducing the generator output to zero. At the same time, the reverse current cutout operates to disconnect the malfunctioning generator from the main dc bus tie cables, and to deenergize the bus control relay which causes the generator warning light part of the master warning system to light up. The trip relay also causes the generator equalizer bus relay to be deenergized, since only one generator is operating to provide all the load requirements, and deenergizes the generator's reverse current cutout. The generator remains disconnected from the main dc bus until the overvoltage control is returned to its normal operating condition. This is accomplished by setting the generator switch to RESET. When the switch is set to RESET, the reset coil of the trip relay is energized, closing the trip relay contacts, completing the field circuit, and establishing generator voltage. With the generator switch in the ON position, the reverse current cutout relay connects the generator to the main dc bus. At the same time, the generator bus control relay is energized, causing the generator warning light to to go out. Placing the switch in the ON position also energizes the bus equalizer relay, bringing the bus equalizing system into operation. If the generator warning light again comes on, the generator switch should be in the OFF position. If both generators malfunction, each is disconnected from the main dc bus tie cables by its own system. Since both the right and left bus control relays are de-energized, both generator warning lights come on and the monitor dc bus relay control circuit is interrupted. De-energizing the monitor dc bus relay disconnects the monitor dc bus load from the main dc bus. The batteries then supply power to the main and essential dc buses only. Setting the BATT switch to the EMER position disconnects the batteries and the essential dc bus from the main dc bus and connects the batteries to the essential dc bus only.

Each generator is provided with an overheat warning light which is controlled by a thermal switch mounted in the exhaust end of its cooling duct. When the air passing through the exhaust end of the cooling duct reaches 300° ± 10°F the switches close and the overheat warning light comes on.

A trip switch is provided, on the upper overhead panel, under the gang bar, for each generator to actuate the trip relay coil of the generator's overvoltage relay. Setting this switch to TRIP disconnects the generator from the main dc bus tie cables in the same way that an overvoltage condition disconnects the generator from the main dc bus. The generator switch must be set to RESET and then to ON before the generator can again be brought into operation after tripping.

3. Battery Operation

(See Figure 2 thru Figure 5)

Each of the 24-volt, 34-amp-hour batteries is connected to the main dc bus when its own normal battery contactor is energized, or to the essential dc bus when its own emergency battery relay is energized. Both normal battery contactors are energized when the BATT switch is set to NORM. This connects the batteries to the main dc bus, providing power to energize the No. 1 and No. 2 normal bus tie relays which connect the essential dc bus to the main dc bus. Setting the BATT switch to EMER energizes the left and right emergency battery relays these relays connect the batteries to a common point, providing power to energize the emergency bus tie relay. Energizing the emergency bus tie relay connects the batteries to the essential dc bus only. Each relay control circuit in the battery and bus tie system is protected by a 5-amp circuit breaker. In addition, the normal and emergency bus tie relays each have blocking diodes in their circuitry to prevent the battery potential from reaching the main dc bus and the emergency cross tie cable through the normal battery contactors and the emergency battery relays' coils, and the bus tie relay's coils.

On Aircraft 1 - 200, 322 and 323 having ASC 192 two push button indicator light switches are installed on the upper overhead panel as a means of individually disconnecting the batteries in flight. By depressing the left and right BATT DISC switch that respective battery normal and emergency relays are de-energized. An amber light in the switch will indicate that the battery has been disconnected.

Provisions are made to energize the main dc and the essential dc buses from outside the aircraft with all cockpit switches off. This is accomplished by means of the OUTSIDE BATTERY SWITCH which is located on the aft outboard side of the left nacelle, adjacent to the external power receptacle. Placing the switch to the ON

position energizes the left normal battery contactor and the No. 1 normal bus tie relay providing the left battery is connected and is sufficiently charged. By energizing these relays, power is placed on the main dc and essential dc buses, allowing the operator to raise the main airstair door to open the landing gear clam shell doors without entering the aircraft. The outside battery switch warning light (part of the master warning system) will light up whenever the switch is on and the dc buses are powered. An outside BATTERY SW. CKT. BKR. located in the left battery compartment provides protection for this circuit.

WARNING: DO NOT TAKEOFF WITH THE OUTSIDE BATTERY SWITCH IN THE ON POSITION. IN ANY SITUATION REQUIRING AN EMERGENCY CONFIGURATION, POWER CANNOT BE REMOVED FROM THE MAIN DC BUS WITH THE OUTSIDE BATTERY SWITCH IN THE ON POSITION.

CAUTION: ACCESS TO THE OUTSIDE BATTERY SWITCH IS BY MEANS OF A FLUSH MOUNTED, THUMB RELEASE, HINGED DOOR. THE OUTSIDE BATTERY SWITCH MUST BE IN THE OFF POSITION BEFORE DOOR CAN BE CLOSED.

When either engine is started or cranked, the related start interlock relay is energized to bring the emergency bus tie system into operation. Each of these relays is energized as long as its related starter relay is energized. each relay provides a ground to energize the emergency battery relays and the emergency bus tie relay, even through the BATT switch is set to NORM. This minimizes starting current line drop.

4. **External Power System Operation**

The external power system includes the provisions for connecting the aircraft to an external source of 28-volt dc power, sampling this power for proper voltage and polarity, and connection to the dc power distribution system. An external power receptacle is located on the aft outboard side of the left nacelle, which permits connecting of an external dc power source to the aircraft. The receptacle is a standard three-pin, heavy duty receptacle which mates with standard three-socket oval plugs. When external power is connected to the aircraft, the external power relay must be operated to connect the power source to the main dc bus tie cables. This relay is similar to the generator reverse current cutout. Power to control the relay is obtained from the external power source through the small pin and socket of the external power plug and receptacle, the 5-amp external power circuit breaker, and the EXT PWR switch is located on the upper overhead panel. When the circuit breaker is in and the switch is set to ON. The external power relay is energized, connecting the power source to the main dc bus only. In order to apply power to the essential dc bus, the BATT switch must be placed in the NORM position.

Aircraft 1 - 200, 322 and 323 having ASC 193A have a ground power overvoltage and reverse polarity protection system, eliminating the need to make a BATT switch selection to power the essential dc bus and charge the battery. An overvoltage relay, a latching relay, and an external power distribution relay have been added. The two position EXT PWR switch has been replaced with a three-position switch incorporating a DC RESET position. If external power connected to the aircraft system is less than 29 volts dc and of the proper polarity, the overvoltage relay package will allow power to the external power relay. Closing the external power relay energizes the external power distribution relay, and provides a ground for the left and right battery normal relays and the Number 1 and Number 2 bus tie relays. Aircraft 1 - 200, 322 and 323 having ASC 193A, an engine start interlock relay is added. This relay is energized when the START MASTER switch is selected to START or CRANK position, opening a set of normally closed contacts, isolating the ground power overvoltage latching relay, and preventing an overvoltage condition from interrupting a crank or start cycle.

5. **APU Generator Operation**

(See Figure 7 and Figure 8)

The APU, through its accessory section gear train and a drive belt, drives a 50 or 200 amp dc generator. Generator output of both units is regulated at 27.0 $\pm$ 0.2 volts. Switching circuit power is taken directly from the self-excited generator, therefore the generator must be putting out and the APU up to operating rpm before the generator can be placed on the line. Switching circuit operation for the 50 amp generator is as follows: power is taken from generator output, routed to the APU generator circuit breaker, then the APU generator control switch located on the upper overhead panel. When the switch is placed in the ON position, power continues on through the closed contacts of the APU loading relay (energized when APU is up to operating rpm - 95%) to the circuit limiter (Aircraft 1 - 60 and 114 having ASC 262 and Aircraft 61 - 200, 322 and 323) and then to the voltage relay coil of the reverse current cutout. Providing that the generator output is of correct polarity, and is 0.35-0.65 volts above the voltage of the essential dc bus, the main contactor of the reverse

current cutout will close, connecting generator output to the essential dc bus. The switching circuit of the 200 amp generator installation is virtually the same with the exception that it also passes through a set of closed contacts of an overvoltage relay (set to actuate at 28.0 $\pm$ 1.0 volt field voltage) and through an overload control (a thermal switch, which when overheated, due to an overload condition breaks the control circuit). APU generator amperage is indicated on the APU generator ammeter (0-300 amps or 0-150) located on the upper overhead panel. The other half of this volt-ammeter indicates essential dc bus voltage, and will indicate APU generator voltage since the generator output is connected to the essential dc bus.

An APU GEN OFF light mounted on the upper overhead panel will light up only when the APU generator switch is on and the generator is putting out up to its reverse current cutout but is not connected to the buss. Since power for the light is derived directly from the generator output it will not come on if the drive belt is snapped, if the generator is not putting out due to a malfunction, or if the overvoltage relay is tripped (200 amp installation).

On Aircraft 1 - 100 and 114 having ASC 158A and Aircraft 101 - 200, 322 and 323, (See Figure 8), the APU GEN OFF warning light will remain on as long as the APU is running at its operating RPM and the APU generator is not supplying power to the essential dc bus regardless of the position of the generator switch. This system will give a true indication of a malfunction since the warning light is not dependent upon the APU generator for power to come on.

6. DC Power Distribution

(See Figure 9)

Distribution of dc power is accomplished using the three bus system these are the main, essential, and monitor dc buses. The engine-driven dc generators and the external power receptacle are connected to the main dc bus by means of the main dc bus tie cables. The cables run from each nacelle to the main power junction box located under floorboard No. 11 and from the main power junction box forward to the inverter relay box via the inverter fuse panel. Any equipment having large dc load requirements is fed directly from the inverter fuse panel. The essential dc bus is connected to the main dc bus by the contacts of the No. 1 and No. 2 normal bus tie relays when they are energized. The APU generator tie cable runs forward form the APU relay box to the inverter relay box, and is connected to the essential dc bus by means of a 200 amp current limiter. Extensions of all three dc buses are located in both the right and left Fuselage Station 133 circuit breaker panels. The main and essential dc bus extensions are connected to the main and essential buses in the inverter relay box by feeders which have circuit breakers at the inverter relay box end as well as the circuit breaker panel end. The monitor bus is connected to the main dc bus by the monitor dc bus relay in the inverter relay box. The monitor dc bus extensions in the circuit breaker panels are each fed by a single feeder: both feeders being protected by a single circuit breaker at the inverter relay box end. The essential dc bus feeds these circuits considered to be of prime importance to the safety of flight while the main dc bus feeds those circuits considered to be of secondary importance. The monitor dc bus controls or feeds those circuits which would impose heavy loads on the batteries in the event of failure of both generators.

With both engines operating and the dc generators' control switches set to ON, the output of the generators is connected to the main dc bus. If the generators are not used, and external power is connected, the external power unit output feeds the main dc bus. Application of power to the essential dc bus is controlled by the battery switch. Setting the BATT switch to NORM energizes the left and right normal battery contactors and the No. 1 and No. 2 normal bus tie relays. Aircraft 1 - 200, 322 and 323 having ASC 193A application of external power to the batteries and the essential bus is automatic with the battery switch left in the OFF position. The contactors connect the batteries to the main dc bus, while the normal bus tie relays connect the essential dc bus to the main dc bus. If the generators or external power are not used to supply power, the main and essential dc bus are energized solely by the batteries. With the BATT switch set to EMER, the left and right emergency battery relays are energized, connecting the batteries to a common point and providing power to energize the emergency bus tie relay. The emergency bus tie relay connects the batteries to the essential dc bus. In this case, if the generators are used or if external power is used, the main dc bus is fed by the generators or external power, and the essential dc bus is independently fed by the batteries (with the BATT switch in EMER).

If the APU generator is used to supply power (generators and external power not used), the output of this generator feeds the essential dc bus. With the BATT switch set to NORM, the APU generator feeds the main and essential dc buses and the batteries. With the BATT switch set to EMER, the APU generator feeds the

essential dc bus and the batteries. After the APU generator is put on the line, the BATT switch can be turned off allowing the APU generator to feed the essential dc bus only.

CAUTION: DO NOT OVERLOAD THE APU GENERATOR. TO MAINTAIN GENERATOR OUTPUT WITHIN LIMITS. DISCONNECT (TURN OFF) ALL LOADS NOT REQUIRED. DEPLETED BATTERIES MUST BE FULLY CHARGED, ONE AT A TIME, BEFORE APPLYING ANY OTHER LOADS TO THE APU GENERATOR, OR THE BATTERIES SHOULD BE DISCONNECTED FROM THE BUS DISTRIBUTION SYSTEM BY PULLING THEIR RESPECTIVE CONTROL CIRCUIT BREAKERS. THE LOAD CAPACITY OF THE 50 AMP IS 150 AMPS FOR ONE SECOND, 75 AMPS FOR FIVE MINUTES AND 50 AMPS THEREAFTER. EXCEEDING THESE LIMITS SERIOUSLY DAMAGES THE GENERATOR. THE LOAD CAPACITY FOR THE 200-AMP GENERATOR IS 200 AMPS. ANY OVERLOAD, EVEN MOMENTARY, MAY DAMAGE THE GENERATOR OR ITS COMPONENTS.

The batteries installed in the aircraft are of sufficient capacity to take care of normal requirements. If large loads are applied to them (such as large inverter, radar, etc.) without generator or external power,the limited amount of electrical energy will be used up rapidly. A separate dc power bus was designed into this aircraft specifically to prevent using those units which require high currents, unless a generator or external power were also on the bus. This bus is called the monitor dc bus. Most high current consumption units receive control power form this bus, as well as a few circuits which are directly connected to it. The system, through relays, sense engine-driven generator operation. If either or both generators are on the line, a signal will be applied to connect the monitor dc bus to the main dc while the aircraft is in flight. The loads therefore will be carried by the operating generators, not the batteries.

If the aircraft is on the ground, another difficulty arises. The generators require outside air blast for cooling. Below 11,000 rpm, the propeller is in flat pitch and therefore does not produce a significant movement of air through the generators. The possibility of generator overheating arises. To prevent this, switches which sense power lever position are added in series with the generator sensing circuit. On the ground then, in order to operate any unit controlled or powered by the monitor dc bus, it is necessary that at least one engine generator be on the line and a that a power lever be advanced above the 11,000 rpm position. In flight, slip stream provides the generator cooling air, therefore a circuit bypassing the power lever switches is controlled by the nutcracker system. In flight there must be at least one engine-driven generator on the line, to maintain power on the monitor dc bus. With the system as described, it would be necessary on the ground to run at least one engine, when necessary to check the large inverters or any other equipment controlled or on the monitor dc bus. A monitor dc bus test switch is provide to bypass all the automatic sensing circuits. The monitor dc bus is connected to the main dc bus through the monitor dc bus relay when the test switch is in the ON position. The test switch is located on the lower outboard side of the pilots circuit breaker panel. Power to energize the monitor dc bus relay comes from the main dc bus through the MON. BUS circuit breaker. From the circuit breaker, the circuit is wired to one pole of the number 1 nutcracker relay. This relay senses whether the aircraft is on ground or in flight. As noted above, the generator requires prop wash or slipstream to provide air blast for cooling purposes. The characteristics of this propeller require at least 11,000 RPM before it assumes a pitch and starts to move air. Because of this characteristic, the monitor dc bus control circuit is routed, on the ground, through the power lever switches by the nutcracker relay. If either power lever is in the advanced position a path is provided in the circuit. Power is then fed to the A-l terminal on both generator bus control relays. The A-2 terminals of both relays are connected together and are then connected to the monitor dc bus relay coil. Either or both generator bus control relays in the energized position will therefore complete the circuit closing the monitor dc bus relay. The generator bus control relay will be energized whenever its generator is on the line.

The monitor dc bus will drop out immediately when either the last generator disconnects from the line or the last power lever is retarded. In this way, the batteries are protected against high discharge by heavy loads, (except for starting), and the generator is protected against overheating due to extensive periods of heavy loading without cooling air. In flight, circuit operation is identical with the exception that the nutcracker relay, in its relaxed position, bypasses the power lever switches. The circuit is wired directly from the flight position terminal A-3 on the nutcracker relay to the generator bus control relays. The generators are provided with blast air by impact pressure for cooling. In flight the greater concern is heavy drains on the batteries in the event of the loss of both generators. With the monitor dc bus control in normal operation, in flight, the bus will drop out immediately on loss of the last generator, providing the monitor dc bus test switch is OFF. This will minimize the loss of emergency battery capacity.

CAUTION: DO NOT USE THE APU GENERATOR TO FEED THE MONITOR DC BUS. IT WAS NOT DESIGNED FOR THIS PURPOSE.

NEVER PLACE THE MONITOR DC BUS TEST SWITCH ON UNTIL A MAIN ENGINE GENERATOR OR EXTERNAL POWER IS ON THE LINE. OTHERWISE THE BATTERIES MAY BE DEPLETED.

WITH THE MAIN ENGINE GENERATORS ON THE LINE, USE OF THE MONITOR DC BUS TEST SWITCH DEPENDS UPON THE FLIGHT INSTRUMENTATION INSTALLED. ASSUMING THE INSTRUMENT LOAD REQUIRES RUNNING THE LARGE INVERTER, PROPER PROCEDURE WOULD BE TO SWITCH THE MONITOR DC BUS TEST SWITCH ON FOR GROUND OPERATIONS. IF THIS IS DONE MONITOR THE FOLLOWING CONDITIONS: LOW ENGINE RPMS, HIGH OATS, EXTENDED PERIOD OF TIME AT THE END OF THE RUNWAY.

THE GENERATORS DEPEND UPON PROP WASH FOR COOLING AIR DURING GROUND OPERATIONS, WITHOUT COOLING, THE GENERATORS CAN OPERATE FOR 30 MINUTES UNDER A 200 AMP LOAD BEFORE APPROACHING THEIR OVERHEAT LIMIT. FOR A LONGER OPERATING PERIOD, RUN THE ENGINES AT LEAST 11,000 RPM.

TO ACHIEVE AUTOMATIC LOAD MONITORING OF THE LARGE INVERTERS AND RADAR DURING FLIGHT, THE MONITOR DC BUS TEST SWITCH MUST BE OFF.

7. Essential DC Bus Implications.

The table below lists all circuits which are powered by the essential dc bus when the aircraft is manufactured. The heads of all circuits breakers fed by this bus are painted with Dayglo orange for quick identification. All abbreviations listed below indicate the panel nomenclature.

Circuit	Location	
	Pilots CB Panel	Copilots CB Panel
INSTRUMENTS:		
Stall Warning System	STALL CONT.	
Outside Air Temperature	OAT	
Instrument AC Power		INV E K/A INV E PWR INST AC CONT
Air Speed Warning System	SPEED WARN (Note 1 below)	
Left Pitot Heat	L PITOT HTR (Note 2 below)	
ENGINE:		
Fuel Pumps (two)	L & R NORM PUMP PUMP CONT A	
Fuel Cross-Feed	FUEL X FEED	
Fuel Shutoff	L & R FUEL S/0	
Water Methanol Shutoff	L & R W/M CONT.	
Hydraulic Shutoff	HYD S/0	
Fuel Trim and Indication		L & R FUEL TRIM
Torque Pressure Indication		L & R TORQUE
Ignition and Indication	L & R IGN PWR (Note 3 below) IGN IND (Note 3 below) L #1 & #2 IGN (Note 4 below) R #1 & #2 IGN (Note 4 below)	
Warning Lights	WARN LTS BUS	
Fire Detection	AFT FIRE DET GEN O'HEAT	L & R NAC AFT FD L & R NAC FD #1 NAC FD TEST
Fire Extinguisher	L & R NAC FIRE EXT APU FIRE EXT	
(1) A/C 1 - 124, 322 and 323 having ASC 152 and A/C 125 - 200. **(2)** A/C 2 - 148, 322 and 323 having ASC 171 and A/C 149 - 200. **(3)** A/C 1 - 148, 322 and 323 not having ASC 170A. **(4)** A/C 1 - 148, 322 and 323 having ASC 170A and A/C 149 - 200.		
DEICING:		
Fuel Filter Deicing	L & R FUEL FILT	
Wing and Tail Deicing (EMER)	W & T EMER	
LIGHTING:		
Interior- Cockpit	DOME FLD L & R INST EDGE	
	Exterior - Navigation	NAV LTS

Circuit	Location	
	Pilots CB Panel	Copilots CB Panel
CONTROLS:		
Flap Control and Indication	FLAP CONT FLAP IND	
AIR CONDITIONING-PRESSURIZATION:		
	CABIN PRESS UNL RAM AIR	
PROPELLER:		
Cruise Lock Flight Fine Pitch Lock Manual Feather Prop. Lock Warning Lights		CRUISE LK FF PITCH LK L & R FEATHER PROP LK WARN
GENERATORS:		
Generator Trip and Control Generator Warning Lights APU Control APU Generator	APU CONT APU PWR	L & R GEN TRIP L & R GEN CONT GEN WARN LTS APU GEN (Not on bus but essential to APU generator operation).
WARNING:		
Landing Gear Indication	LG DOWN LTS (Note 5 below) NUTCRACKER CONT (Note 6 below) LG WARN HORN (Note 7 below) LG WARN LTS	
(5) A/C not having ASC 108A and A/C having ASC 108A, Part II only.		
(6) A/C having ASC 108A, Part I and A/C 94 - 200, 322 and 323, excluding 114.		
(7) Changed to essential dc bus on A/C having ASC 108A and A/C 94 - 200, 322 and 323, excluding 114.		

8. **Engine Start Interlock Operation**

(See Figure 10)

During engine starting, it is important to have the lowest possible resistance path between nacelles, so the batteries may share the load. To accomplish this, starting interlock relays are used, which actuate in unison with each starting relay. All engine starting is accomplished with BATT switch in NORM position. The purpose of the starter interlock circuit is to close the left and right emergency battery and emergency bus-tie relays, bringing in the third cable for the duration of starter motor operation. (Locate the L eng start interlock relay on Figure 100. Current to energize the relay is provided when a start cycle is instituted. Closing this relay completes the ground leg of both emergency battery relays and the emergency bus-tie relay. when the start cycle is completed or interrupted, power is broken to the start interlock relay, opening the ground circuits to the three emergency relays again isolating the emergency bus feeder system. For a right engine start, identical action occurs except that the right engine start interlock is utilized.

When the start cycle is completed or interrupted, power is broken to the start interlock relay, opening the ground circuits to the three emergency relays again isolating the emergency bus feeder system. For a right engine start, identical action occurs except that the right engine start interlock is utilized.

9. Arrangement of Fuselage Station 193 Fuse Panel

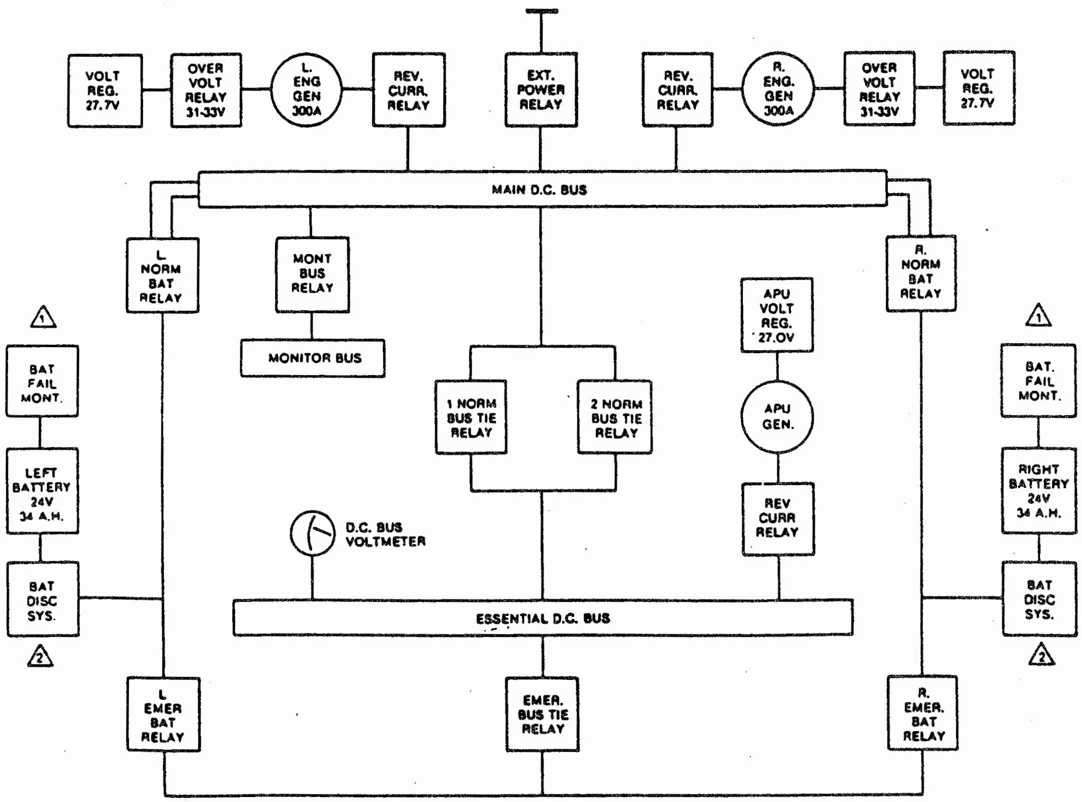
It has been noted that some customizing agencies have rearranged the fuses on the circuit breaker panel at Fuselage Station 193 to obtain more space for radio wiring. The manufacturer has no objection to such arrangement so long as all of the following conditions are met.

- Operating and maintenance personnel associated with the aircraft must be made aware of the change.
- Fuses must continue to be connected to their proper bus bars, and to their proper wires.
- Fuse rating size must be matched to use, not physical position.
- Fuse identification decal must be switched to agree with new locations.

In addition to the above parameters, some basic electronic installation data are worth emphasizing. Radio, auto pilot and compass circuits operate in general at a very low power level. They are, therefore, very susceptible to outside influences. Thus, routing electronic cables away from main power cables is a good practice and, although space in the vicinity of the fuse panel is at a premium, every bit of separation helps. If some intermingling of cables becomes necessary, judgement should be exercised as to which to mix. For example, routing a propeller feathering power wire in the midst of a compass cable would undoubtedly cause wild indicator gyrations while feathering. Even worse, should such a compass be used to supply auto pilot heading data, violent maneuvers could result.

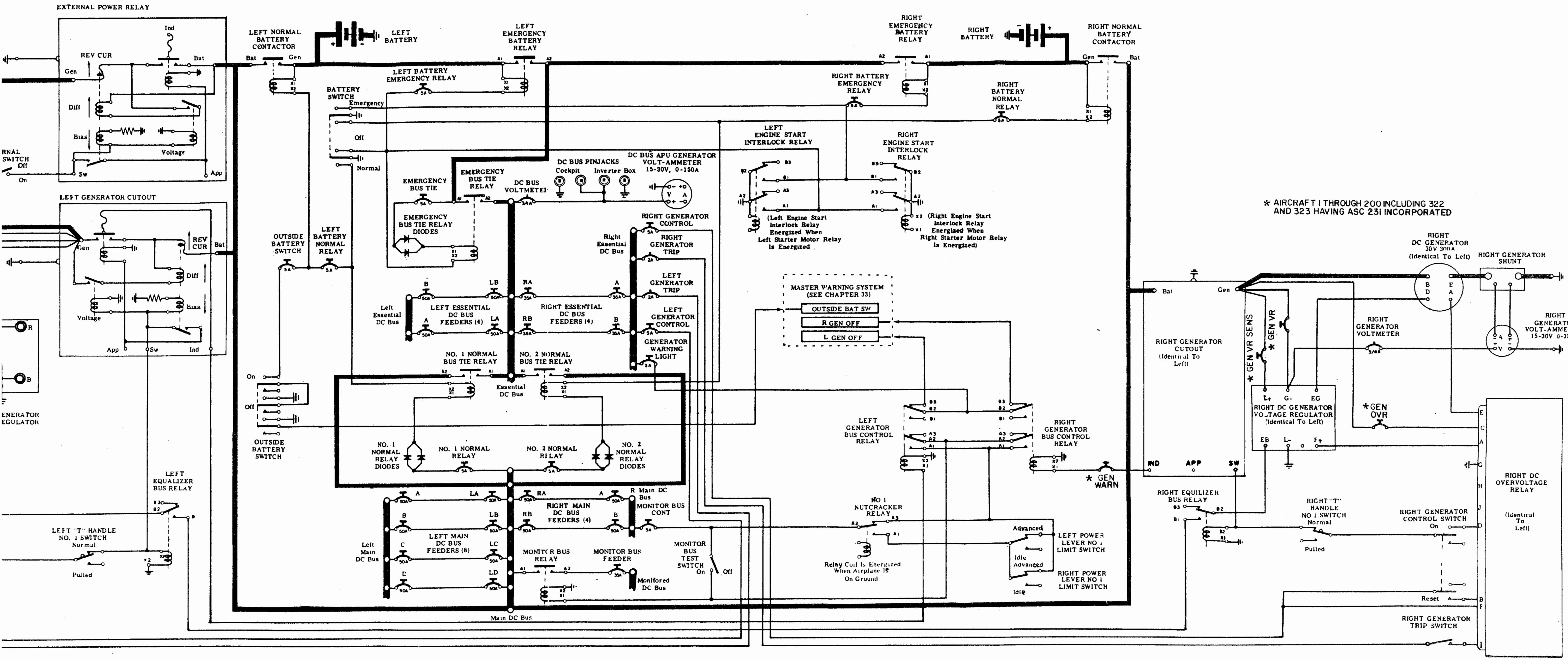
10. Major Components and Locations

Unit	No. Per A/C	Location
Generator, Engine-Driven	2	One on each engine-driven Accessory Gearbox.
Voltage Regulator, DC Generator.	2	Below Inverter Relay Box and Radio Rack.
Overvoltage Relay, DC Generator.	2	Fuselage Station 181 - under FLR-2
Battery, 24V -34 AH Nickel Cadmium type.	2	One in outboard skin of each Nacelle - under the wing.
External Power Receptacle.	1	Outboard skin of Left Nacelle aft of the Battery.
External Power Distribution Relay *	1	Right Fuselage Station 133 Relay Panel
External Power Relay.	1	Left Nacelle Relay Box Inboard Left Nacelle.
External Power Overvoltage Relay Panel	1	Fuselage Station 169 Left Side Upper Floor.
Reverse Current Cutout Relay DC Generator.	2	One in Left and Right Nacelle Relay Box.
Battery Relays (contactors).	4	One Normal and Emergency Relay aft end of each Battery Box.
Bus-Tie Relays	3	Two Normal and one Emergency all in Inverter Relay Box.
No. 1 Nutcracker Relay.	1	Mid-Relay Box under FLR-8
Monitor Bus Relay.	1	Inverter Relay Box.
Starter Interlock Relays	2	Inverter Relay Box.
Engine Start Interlock Relay *	1	Right Fuselage Station 133 Relay Panel
Equalizer Relay, DC Generator	2	Right Fuselage Station 133 Relay Box.
Generator Bus Control Relay	2	Right Fuselage Station 133 Relay Box.
*: Aircraft 1 - 200, 322 and 323 having ASC 193A.		

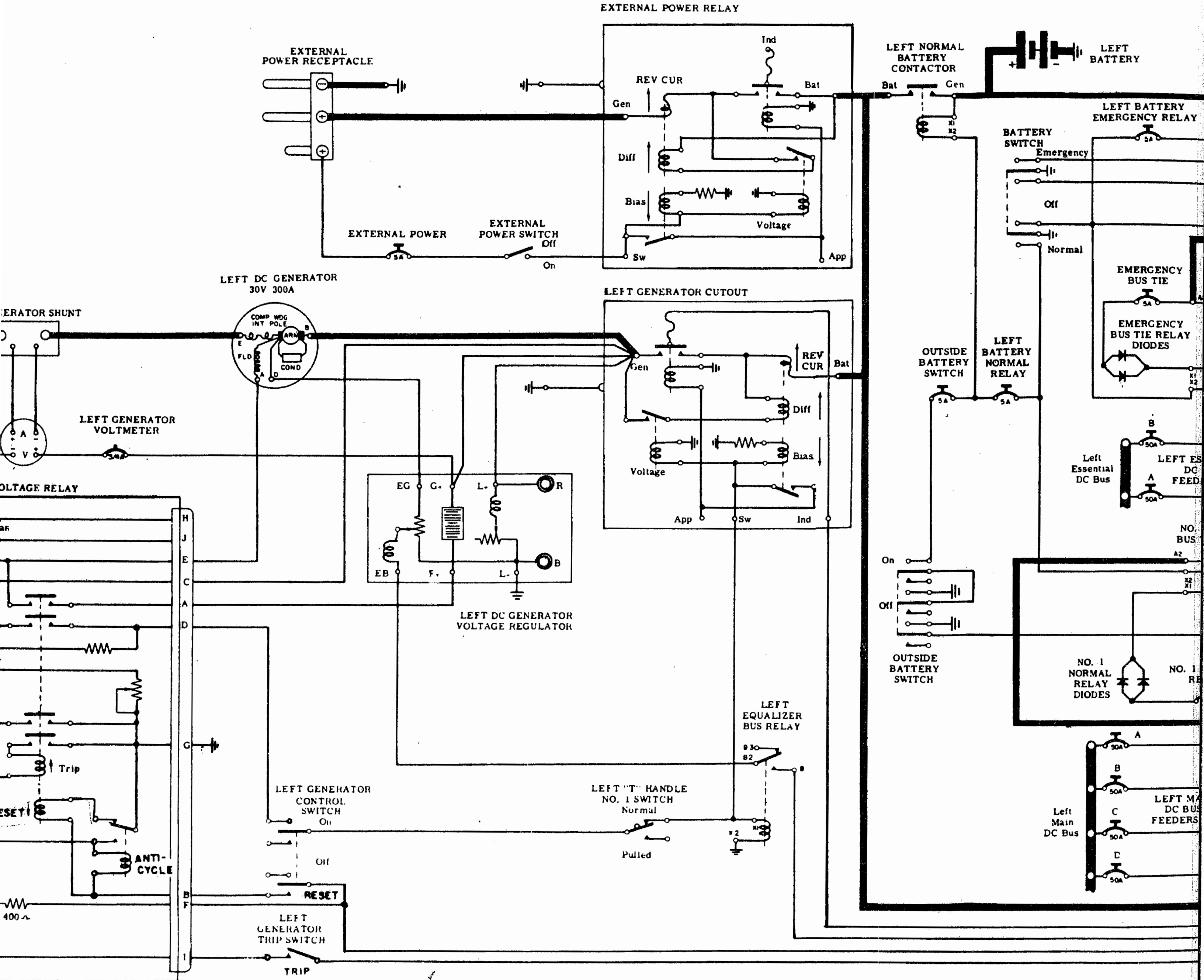


1. AIRCRAFT 1 THROUGH 200 INCLUDING 322 AND 323 HAVING ASC 194 INCORPORATED.
2. AIRCRAFT 1 THROUGH 200 INCLUDING 322 AND 323 HAVING ASC 192 INCORPORATED.

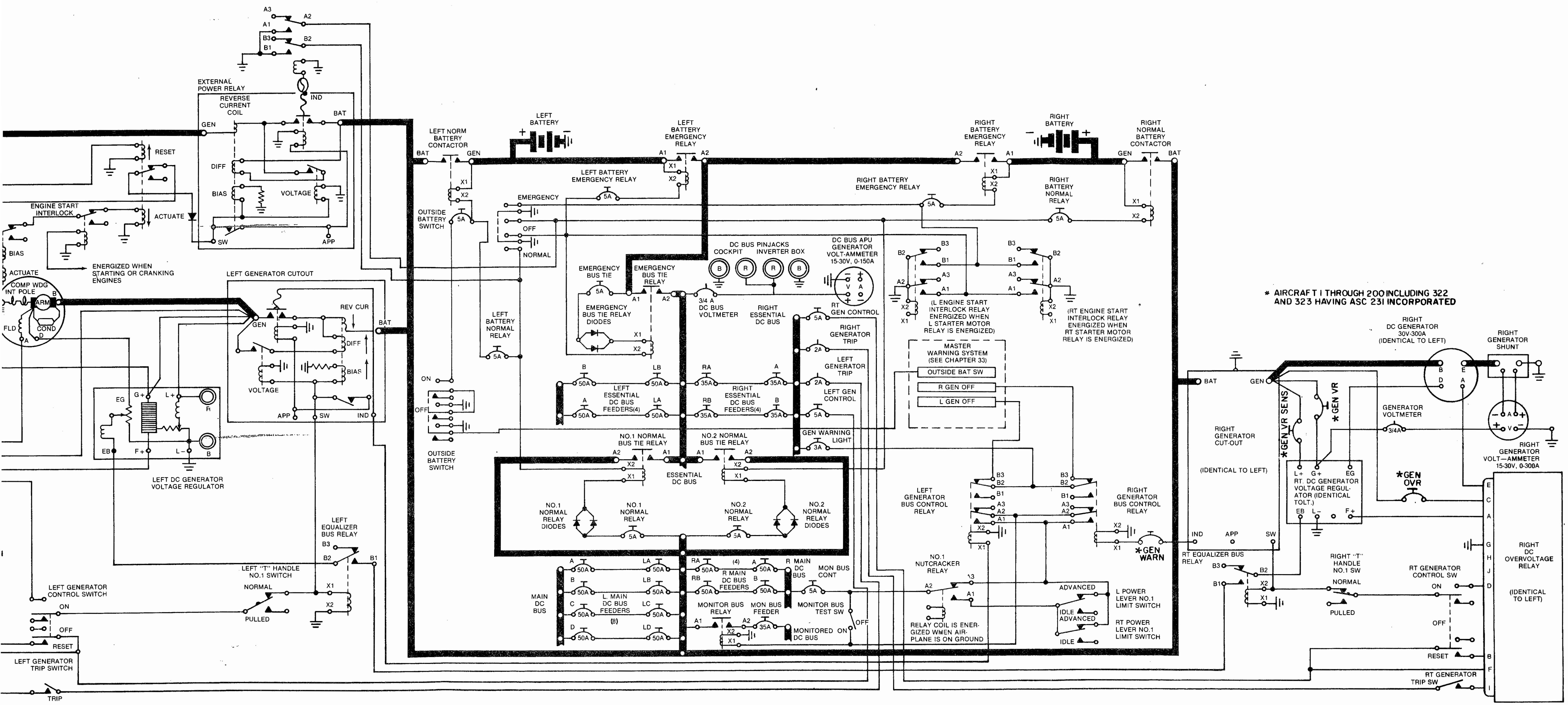
DC Power System Block Diagram
Figure 1.



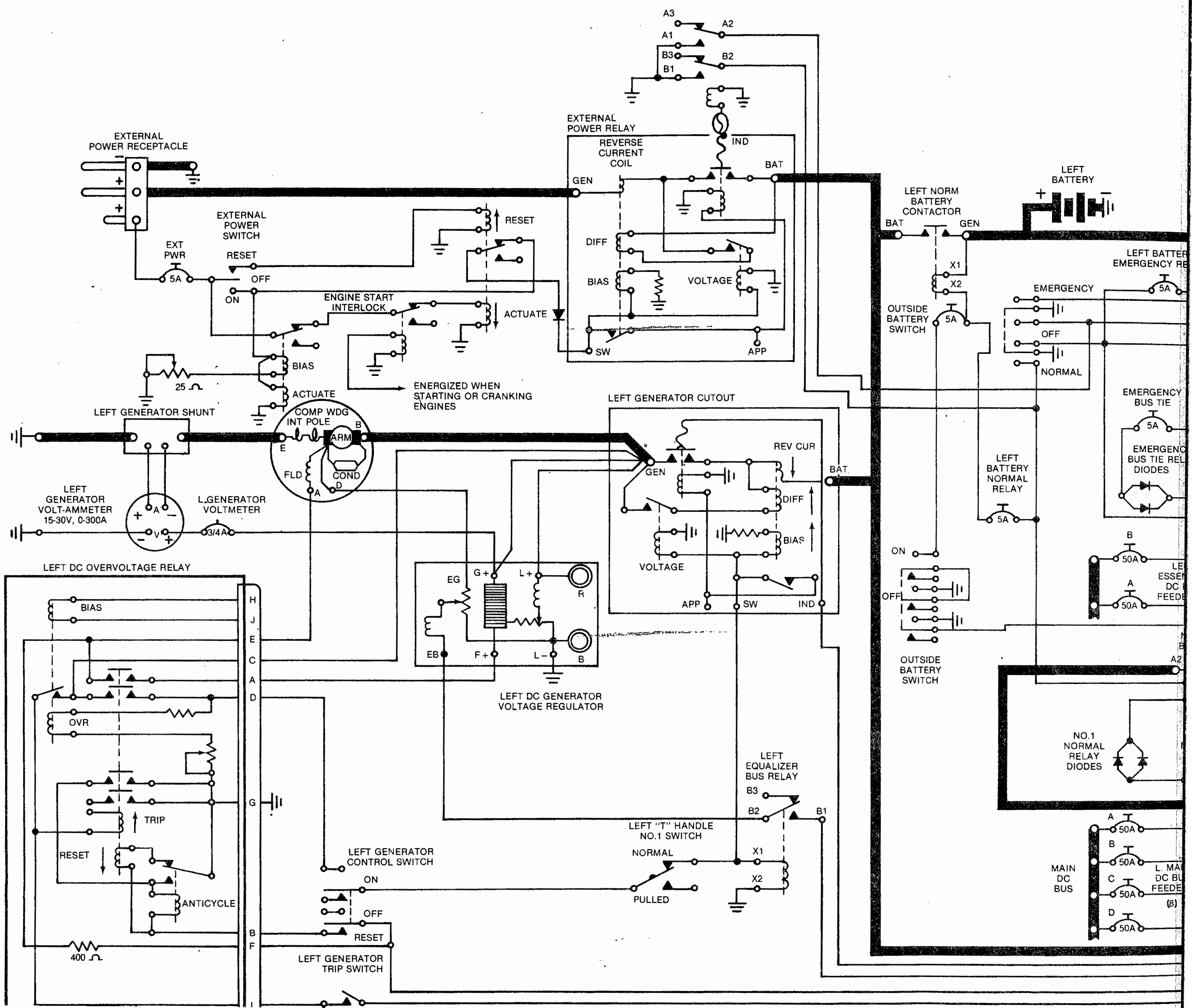
DC Power Supply Circuit — Schematic
Figure 2.



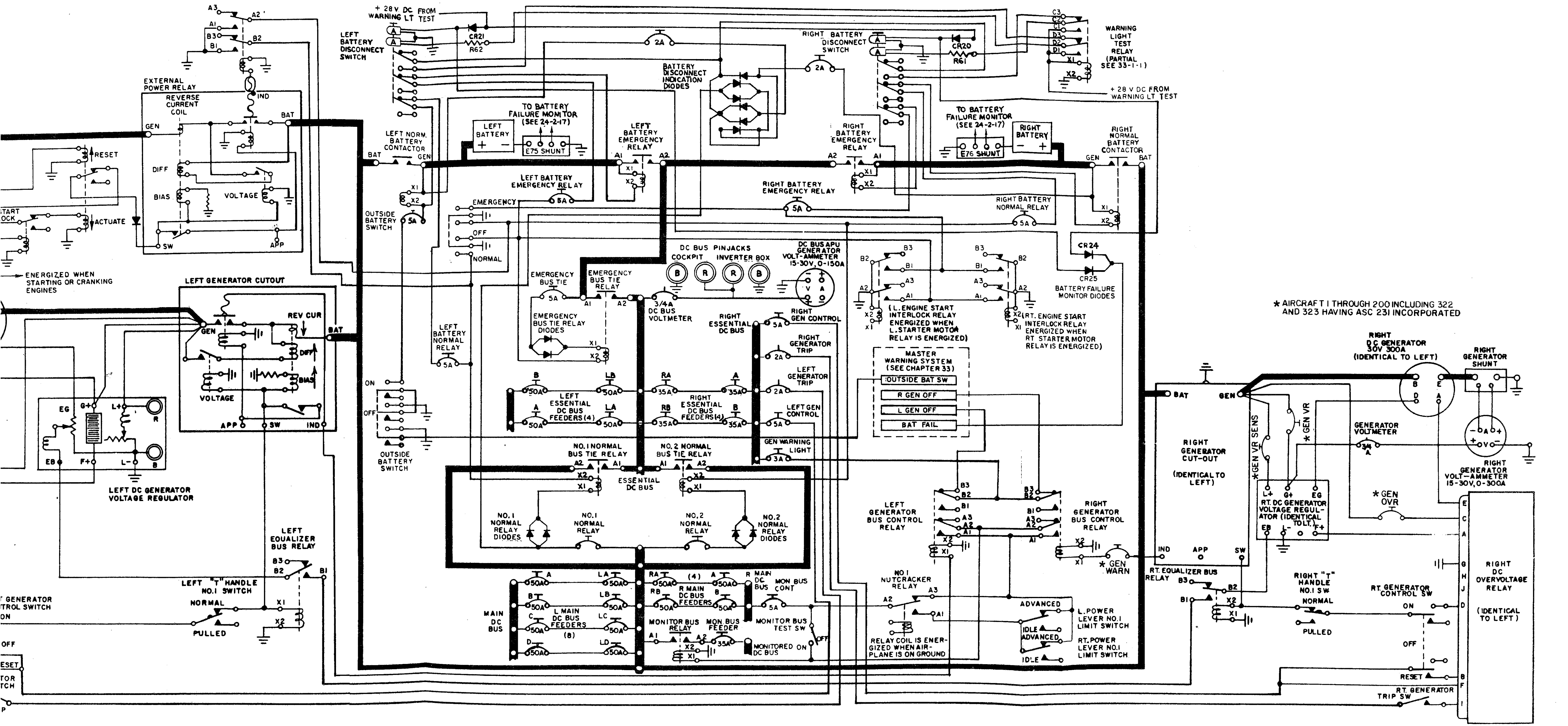
EXTERNAL POWER RELAY
EXTERNAL POWER RECEPTACLE
REV CUR
Ind
Bat
Gen
Diff
Bias
Voltage
Sw
App
LEFT NORMAL BATTERY CONTACTOR
LEFT BATTERY
LEFT BATTERY EMERGENCY RELAY
BATTERY SWITCH
Emergency
Off
Normal
EXTERNAL POWER
EXTERNAL POWER SWITCH
On
LEFT DC GENERATOR 30V 300A
LEFT GENERATOR CUTOUT
EMERGENCY BUS TIE
EMERGENCY BUS TIE RELAY DIODES
OUTSIDE BATTERY SWITCH
LEFT BATTERY NORMAL RELAY
LEFT GENERATOR VOLTMETER
Left Essential DC Bus
LEFT DC GENERATOR VOLTAGE REGULATOR
NO. 1 NORMAL RELAY DIODES
Trip
ANTI-CYCLE
LEFT EQUALIZER BUS RELAY
LEFT GENERATOR CONTROL SWITCH
LEFT "T" HANDLE NO. 1 SWITCH
Pulled
RESET
LEFT GENERATOR TRIP SWITCH
TRIP
Left Main DC Bus
LEFT MAIN DC BUS FEEDERS



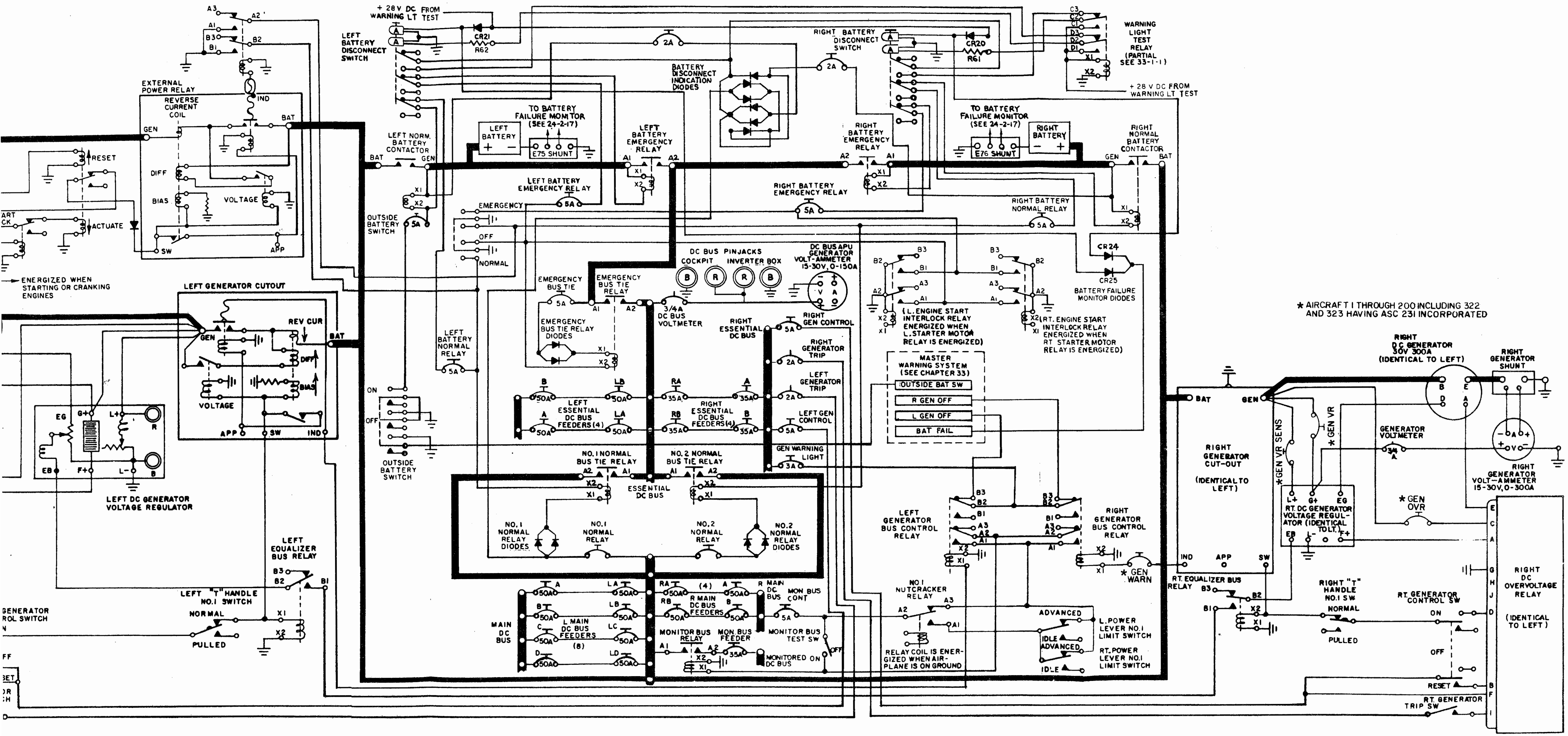
DC Power Supply Circuit — Schematic
(Aircraft having ASC 193A)
Figure 3.



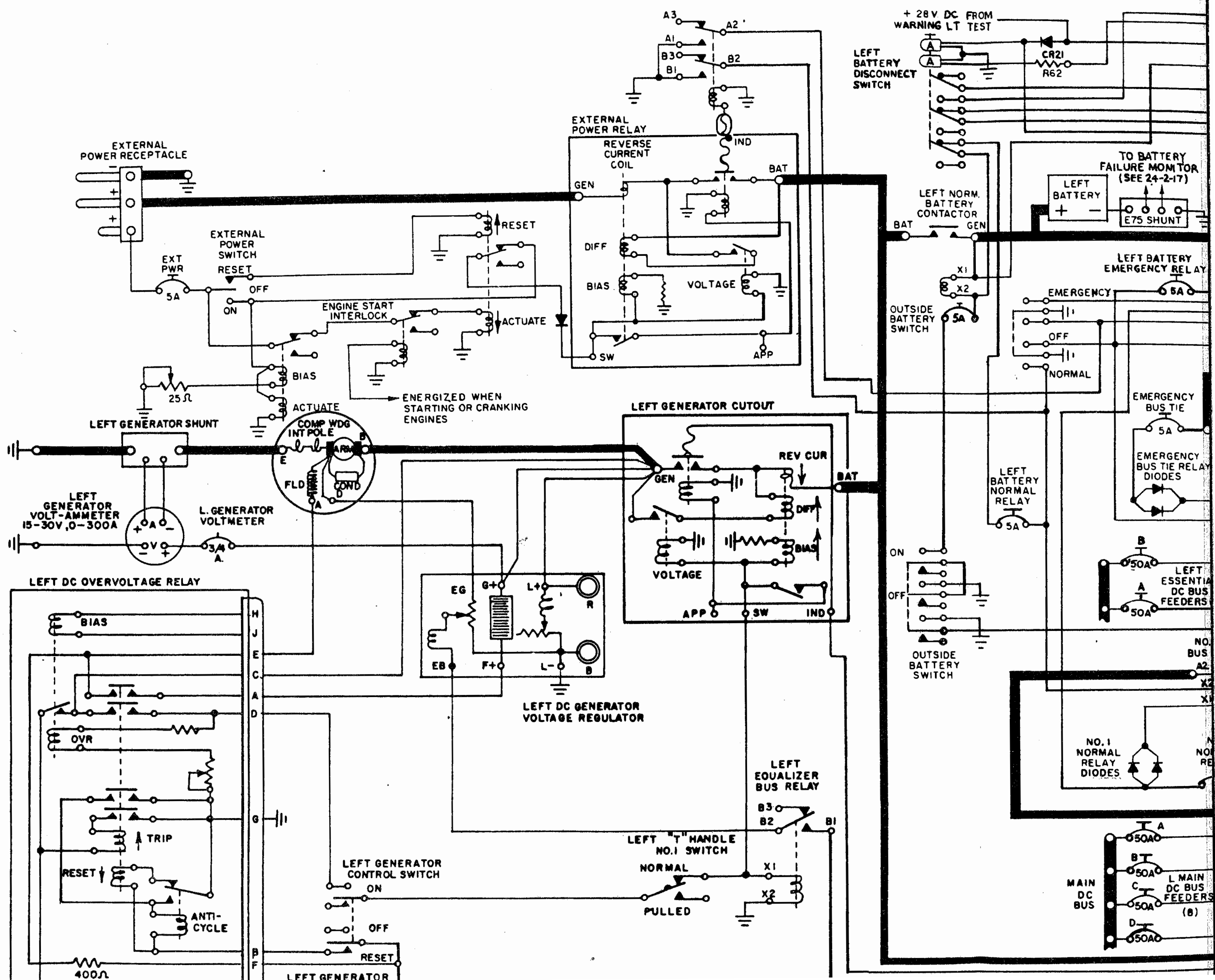
EXTERNAL POWER RECEPTACLE
EXTERNAL POWER SWITCH
EXT PWR
5A
RESET
OFF
ON
ENGINE START INTERLOCK
ACTUATE
BIAS
25
ENERGIZED WHEN STARTING OR CRANKING ENGINES
EXTERNAL POWER RELAY
REVERSE CURRENT COIL
GEN
BAT
IND
DIFF
VOLTAGE
SW
APP
LEFT BATTERY
LEFT NORM BATTERY CONTACTOR
X1
X2
OUTSIDE BATTERY SWITCH
EMERGENCY
NORMAL
LEFT BATTERY EMERGENCY RE
EMERGENCY BUS TIE
EMERGENC BUS TIE REL DIODES
LEFT BATTERY NORMAL RELAY
LEFT GENERATOR SHUNT
COMP WDG INT POLE
ARM
COND
FLD
LEFT GENERATOR VOLT-AMMETER 15-30V, 0-300A
L GENERATOR VOLTMETER
3/4A
LEFT GENERATOR CUTOUT
REV CUR
LEFT DC OVERVOLTAGE RELAY
OVR
TRIP
ANTICYCLE
400
EG
EB
LEFT DC GENERATOR VOLTAGE REGULATOR
LEFT GENERATOR CONTROL SWITCH
LEFT GENERATOR TRIP SWITCH
LEFT EQUALIZER BUS RELAY
LEFT "T" HANDLE NO.1 SWITCH
PULLED
NO.1 NORMAL RELAY DIODES
MAIN DC BUS
50A
L MA DC BU FEEDE
(8)
LE ESSE DC FEED



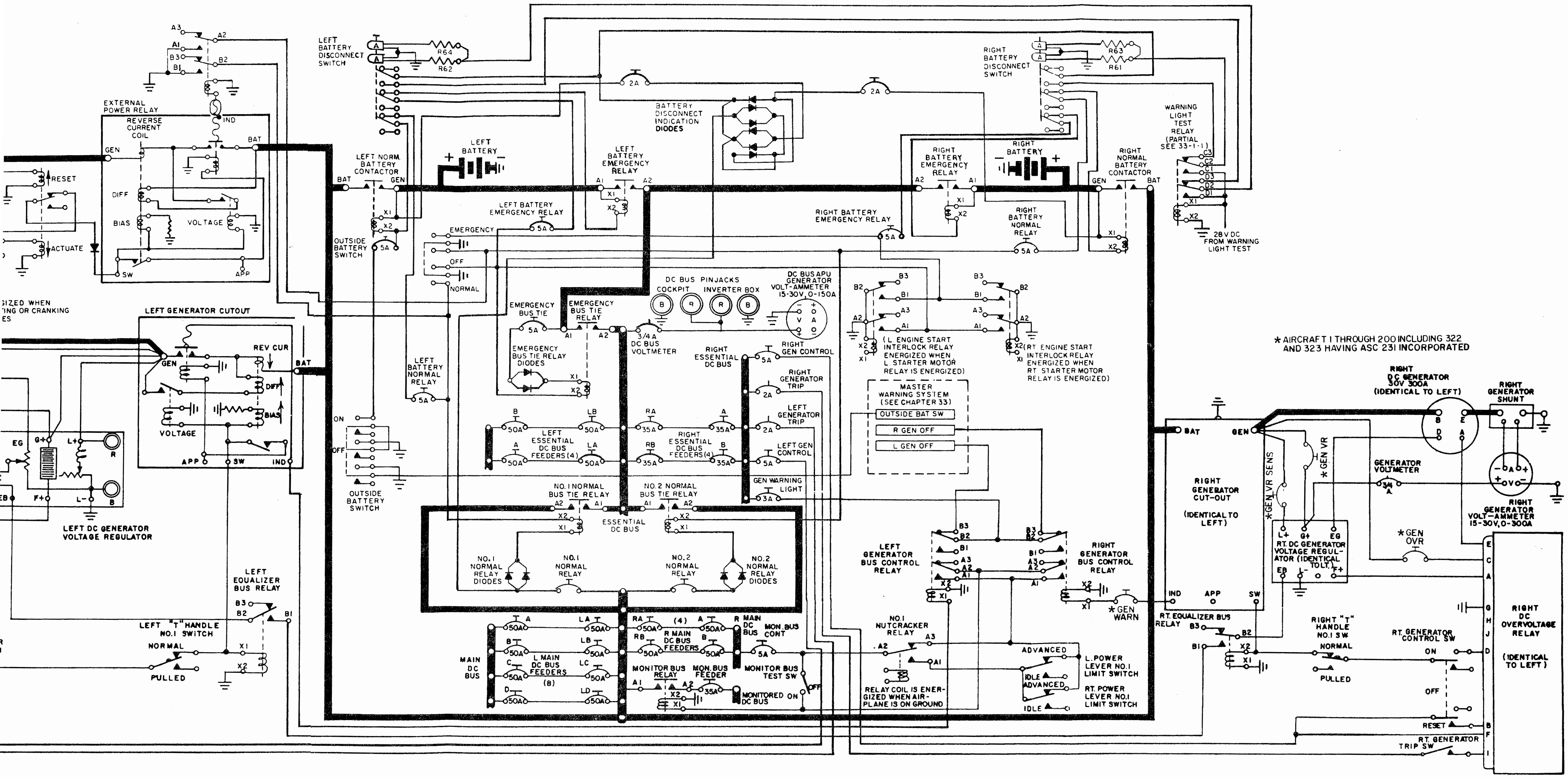
DC Power Supply Circuit – Schematic
(Aircraft having ASC 192, 193A and 194)
Figure 4.



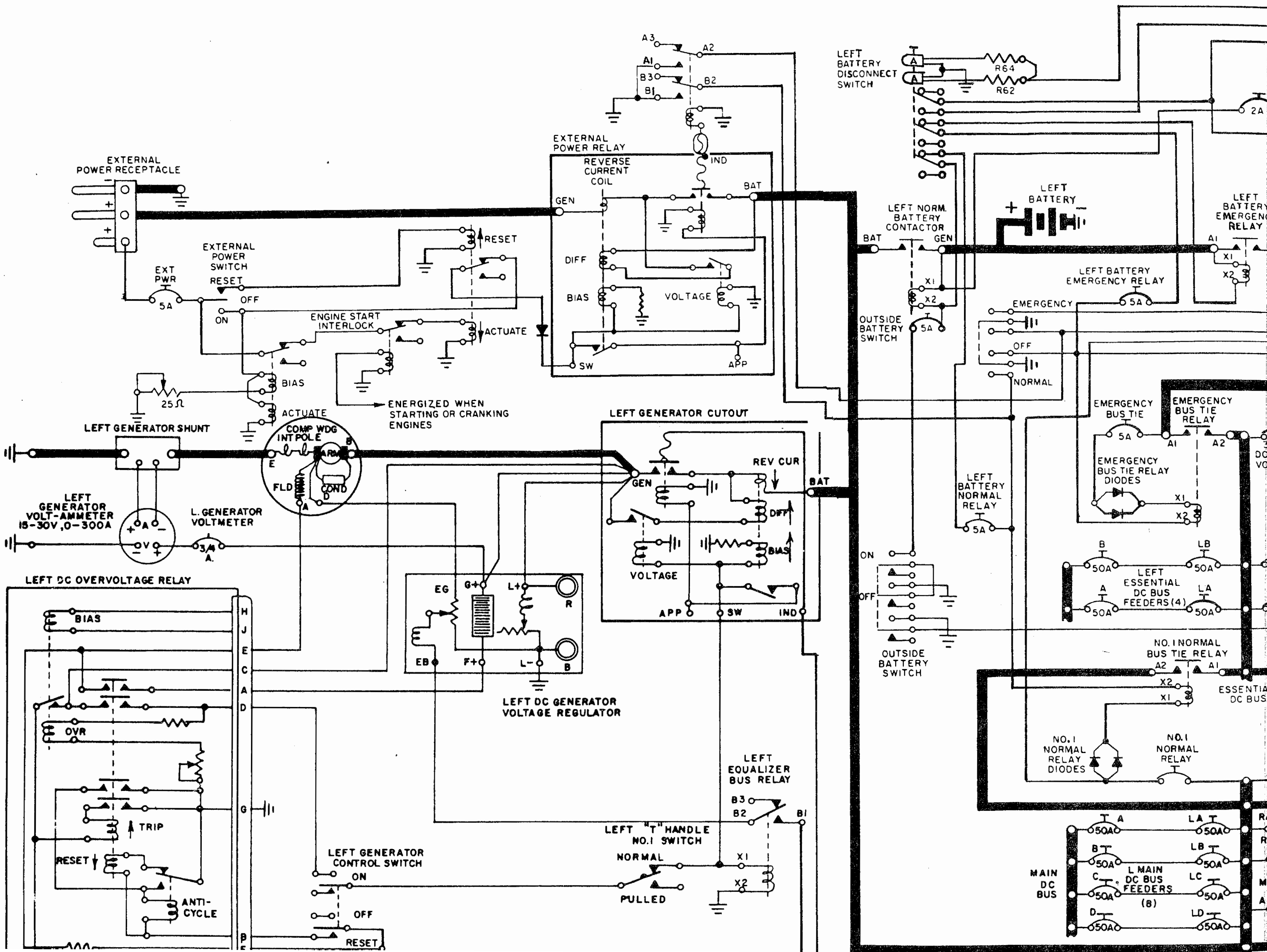
DC Power Supply Circuit — Schematic
(Aircraft having ASC 192, 193A and 194)
Figure 4.

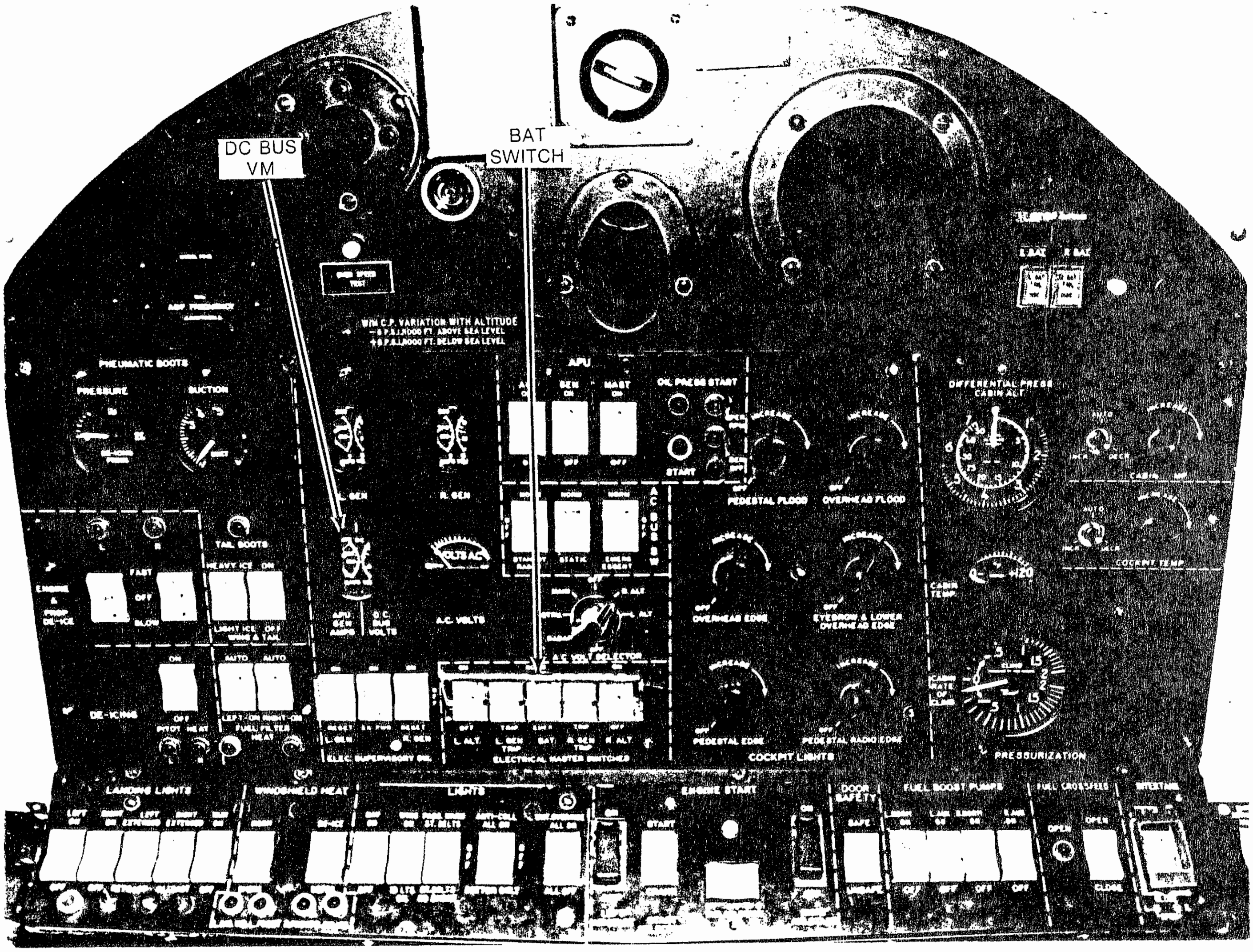


+ 28 V DC FROM WARNING LT TEST
LEFT BATTERY DISCONNECT SWITCH
CR21
R62
EXTERNAL POWER RELAY
REVERSE CURRENT COIL
IND
GEN
BAT
DIFF
BIAS
VOLTAGE
SW
APP
EXTERNAL POWER RECEPTACLE
EXTERNAL POWER SWITCH
RESET
OFF
ON
EXT PWR
5A
ENGINE START INTERLOCK
ACTUATE
ENERGIZED WHEN STARTING OR CRANKING ENGINES
25Ω
LEFT GENERATOR SHUNT
COMP WDG
INT POLE
ARM
FLD
COND
LEFT GENERATOR VOLT-AMMETER 15-30V, 0-300A
L. GENERATOR VOLTMETER
3/4 A.
LEFT DC OVERVOLTAGE RELAY
OVR
TRIP
ANTI-CYCLE
400Ω
LEFT GENERATOR CONTROL SWITCH
LEFT GENERATOR
LEFT DC GENERATOR VOLTAGE REGULATOR
EG
G+
L+
EB
F+
L-
R
B
LEFT GENERATOR CUTOUT
REV CUR
LEFT EQUALIZER BUS RELAY
B3
B2
B1
X1
X2
LEFT "T" HANDLE NO. 1 SWITCH
NORMAL
PULLED
TO BATTERY FAILURE MONITOR (SEE 24-2-17)
LEFT BATTERY
E75 SHUNT
LEFT NORM. BATTERY CONTACTOR
OUTSIDE BATTERY SWITCH
LEFT BATTERY EMERGENCY RELAY
EMERGENCY
NORMAL
EMERGENCY BUS TIE
EMERGENCY BUS TIE RELAY DIODES
LEFT BATTERY NORMAL RELAY
LEFT ESSENTIAL DC BUS FEEDERS
50A
NO. 1 NORMAL RELAY DIODES
MAIN DC BUS
L MAIN DC BUS FEEDERS (8)

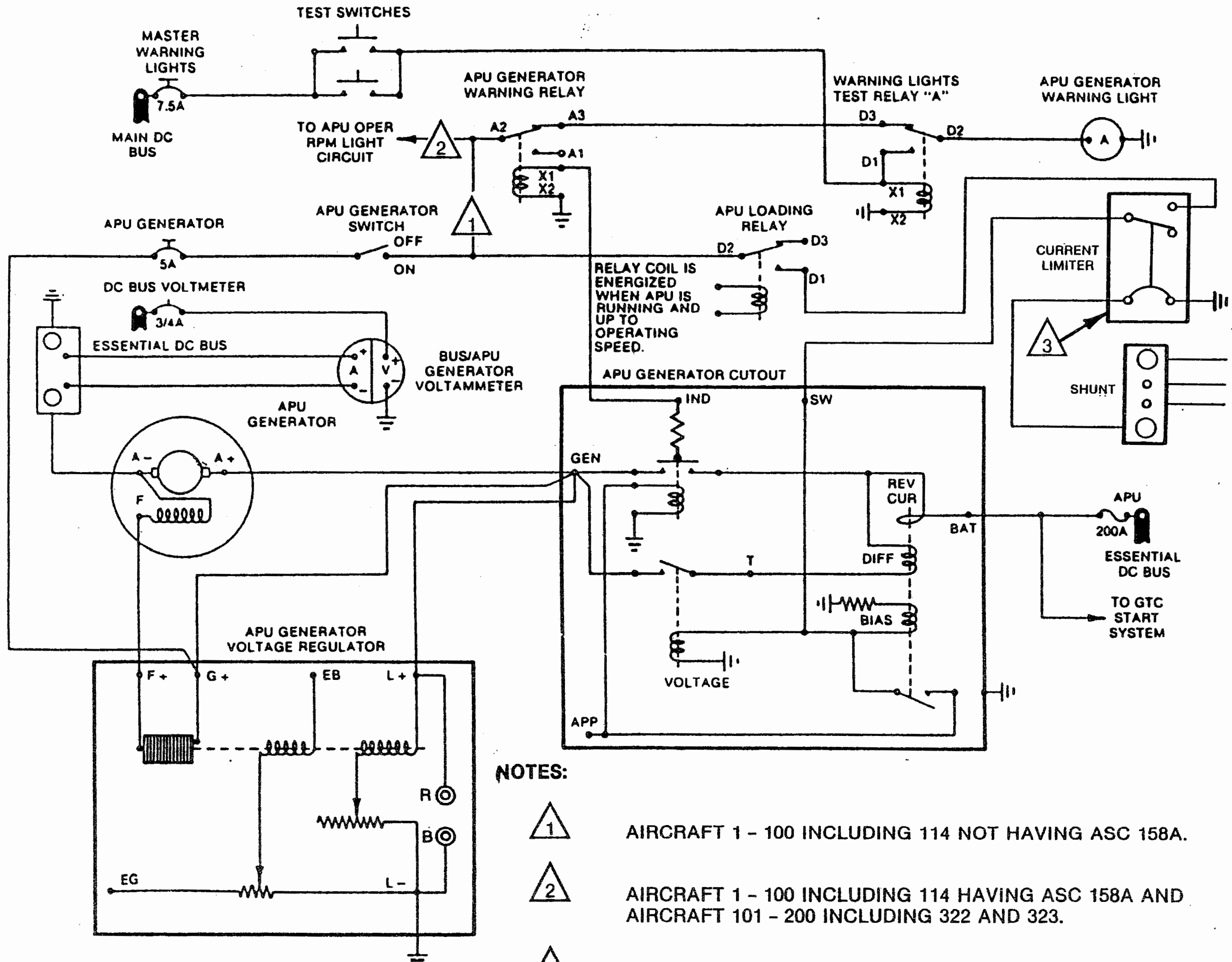


DC Power Supply Circuit — Schematic
(Aircraft having ASC 192, 193A)
Figure 5.


EXTERNAL POWER RECEPTACLE
EXT PWR
5A
EXTERNAL POWER SWITCH
RESET
OFF
ON
ENGINE START INTERLOCK
RESET
ACTUATE
BIAS
ACTUATE
25Ω
ENERGIZED WHEN STARTING OR CRANKING ENGINES
EXTERNAL POWER RELAY
REVERSE CURRENT COIL
GEN
BAT
IND
DIFF
BIAS
VOLTAGE
SW
APP
A3
A2
A1
B3
B2
B1
LEFT BATTERY DISCONNECT SWITCH
R64
R62
LEFT BATTERY
LEFT NORM. BATTERY CONTACTOR
BAT
GEN
X1
X2
OUTSIDE BATTERY SWITCH
5A
LEFT BATTERY EMERGENCY RELAY
LEFT BATTERY EMERGENCY RELAY
5A
EMERGENCY
OFF
NORMAL
2A
LEFT GENERATOR SHUNT
COMP WDG
INT POLE
ARM
COND
FLD
E
B
D
A
LEFT GENERATOR VOLT-AMMETER 15-30V, 0-300A
L. GENERATOR VOLTMETER
3/4 A.
LEFT GENERATOR CUTOUT
GEN
REV CUR
BAT
DIFF
BIAS
VOLTAGE
APP
SW
IND
EMERGENCY BUS TIE
5A
EMERGENCY BUS TIE RELAY
A1
A2
EMERGENCY BUS TIE RELAY DIODES
X1
X2
LEFT BATTERY NORMAL RELAY
5A
ON
OFF
OUTSIDE BATTERY SWITCH
B
LB
LEFT ESSENTIAL DC BUS FEEDERS (4)
A
LA
50A
NO. 1 NORMAL BUS TIE RELAY
A2
A1
X2
X1
ESSENTIAL DC BUS
NO. 1 NORMAL RELAY DIODES
NO. 1 NORMAL RELAY
LEFT DC OVERVOLTAGE RELAY
BIAS
OVR
TRIP
RESET
ANTI-CYCLE
H
J
E
C
A
D
G
B
F
EG
G+
L+
R
EB
F+
L-
B
LEFT DC GENERATOR VOLTAGE REGULATOR
LEFT EQUALIZER BUS RELAY
B3
B2
B1
LEFT "T" HANDLE NO. 1 SWITCH
NORMAL
PULLED
X1
X2
LEFT GENERATOR CONTROL SWITCH
ON
OFF
RESET
MAIN DC BUS
L MAIN DC BUS FEEDERS (8)
A
B
C
D
LA
LB
LC
LD
50A



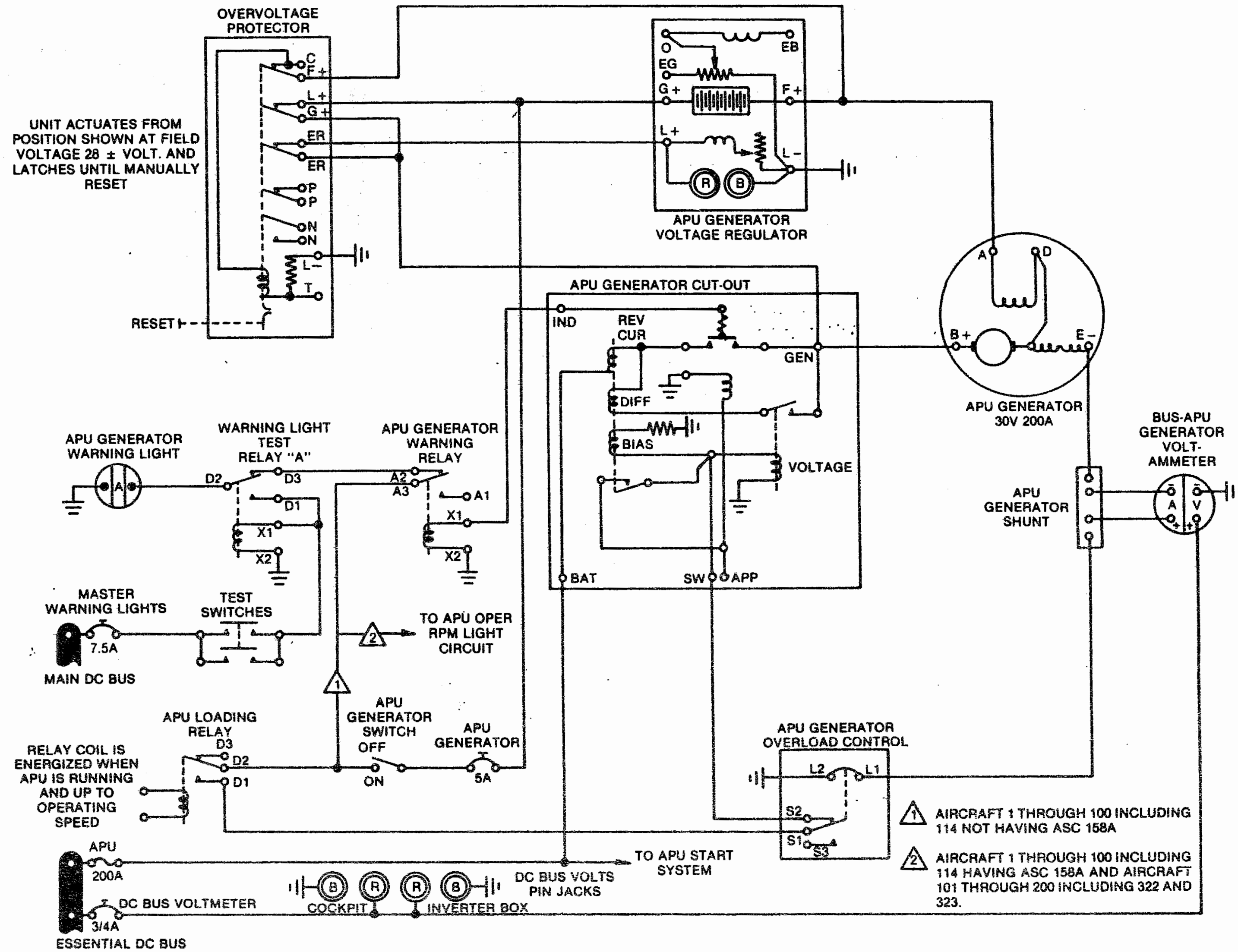
DC Power System Controls and Indicators in Cockpit
Figure 6.



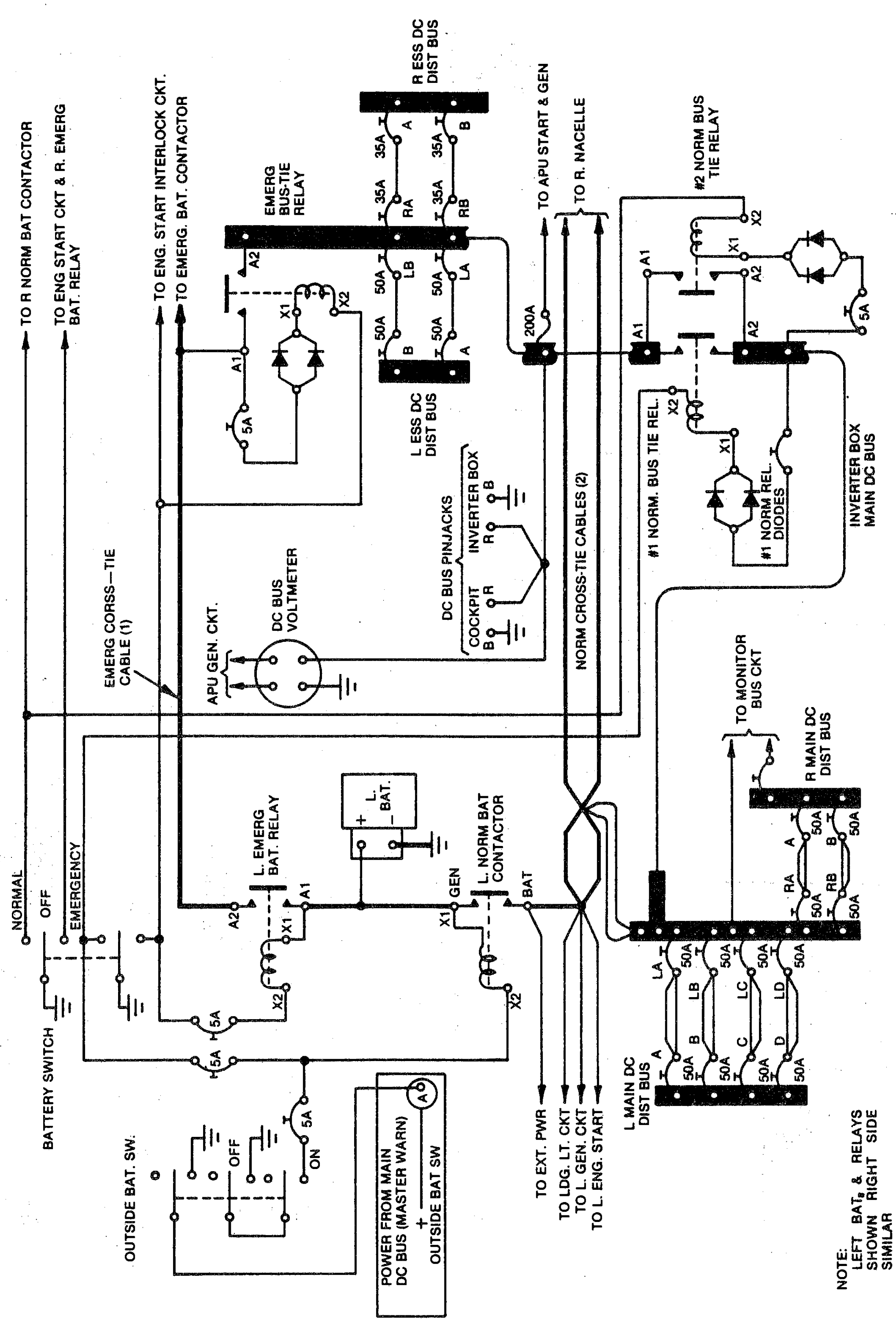
50 Amp APU Generator Control Circuit — Schematic
Figure 7.

NOTES:

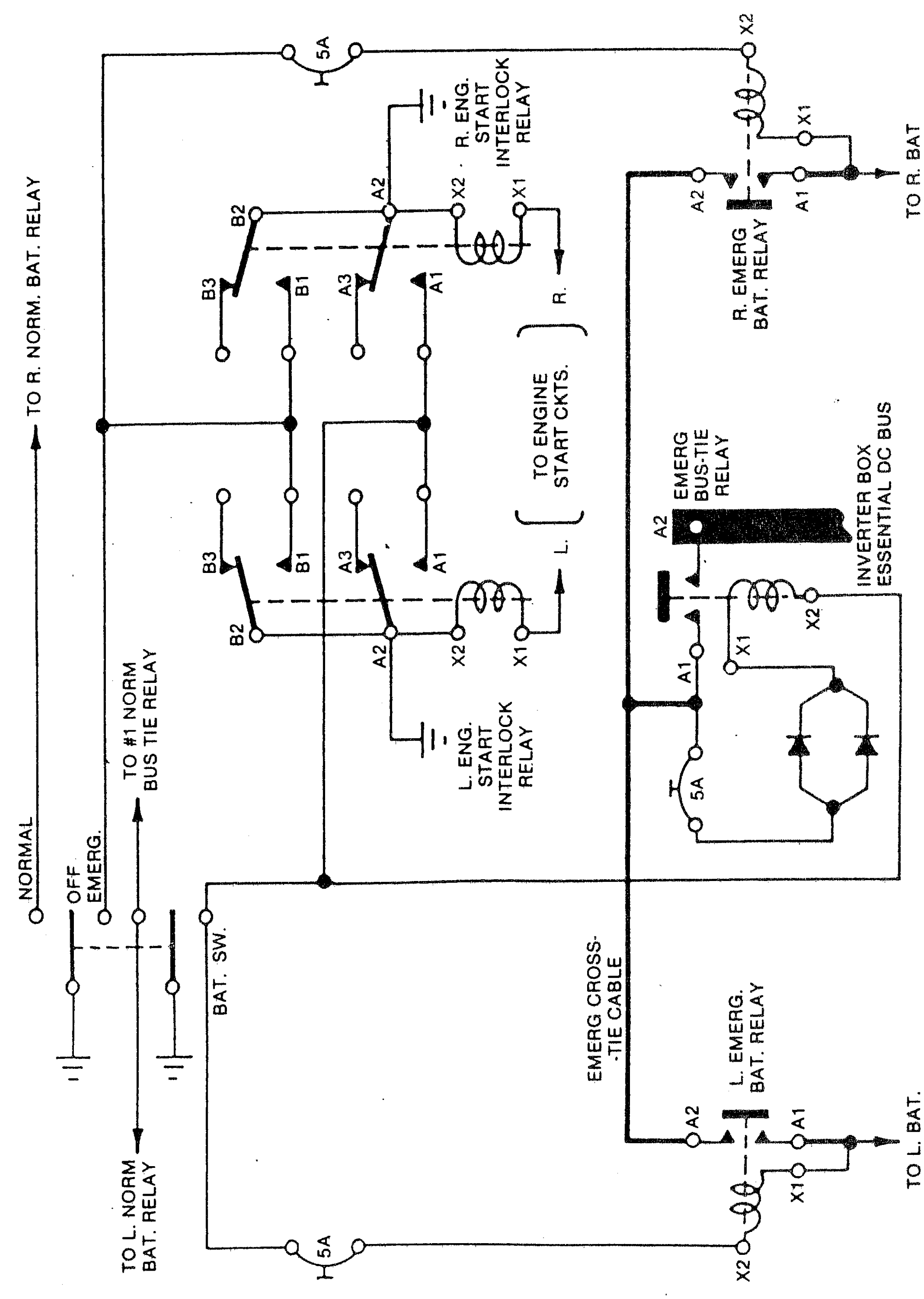
1. AIRCRAFT 1 - 100 INCLUDING 114 NOT HAVING ASC 158A.
2. AIRCRAFT 1 - 100 INCLUDING 114 HAVING ASC 158A AND AIRCRAFT 101 - 200 INCLUDING 322 AND 323.
3. AIRCRAFT 1 - 60 INCLUDING 114, HAVING ASC 262 AND AIRCRAFT 61 - 200, 322 AND 323.



200 Amp APU Generator Control Circuit — Schematic
Figure 8.



DC Power Distribution — Simplified Schematic
Figure 9.



DC Power Engine Start Interlock — Schematic
Figure 10.

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DC POWER SUPPLY AND DISTRIBUTION SYSTEM — FAULT ISOLATION

1. Fault Isolation

Lack of dc bus voltage indication in any portion of the essential dc bus check, indicates that one of the supply paths to the bus is faulty. When a path is found to be inoperative, isolate the defective unit as follows:

A. Measure applicable bus tie relay coil resistance. With BATT switch OFF, each coil should measure approximately 67 ohms at room temperature.

B. Measure diode resistance in both directions. In one direction, continuity should be obtained; in the other direction, an infinite resistance should be obtained.

NOTE: Each circuit uses two diodes wired in parallel. It is necessary to check each diode separately. Unless this is done, a faulty (open) diode will never be discovered until its mate fails.

C. With an ohmmeter, measure resistance across closed circuit (depressed) breaker. Resistance should be approximately 1/6 ohm.

D. With BATT switch in NORM, measure voltage at terminal X1 of battery contactor relay for battery voltage. Terminal X2 should be grounded through BATT switch.

E. Replace any defective units and repeat applicable portion of Essential DC Bus — Operational Test.

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DC POWER SUPPLY AND DISTRIBUTION SYSTEM — MAINTENANCE PRACTICES

1. Essential DC Bus — Operational Test

NOTE: Proper operation of the essential dc bus control circuitry is mandatory for safe operation of the aircraft. To protect against equipment failure, two normal bus tie relays and associated circuits are provided, therefore, to prevent a condition of false security from existing, it is necessary to prove that both of the normal bus tie relays and their associated circuitry are functioning properly.

A. With aircraft on ground, and landing gear handle in the down position, check that the following circuit breakers are engaged:

All dc bus feeders circuit breakers in the inverter relay box and the left and right circuit breaker panels.

- R BATT EMER
- L BATT EMER
- R BATT CONT
- L BATT CONT

NOTE: Step B. thru E. apply to Aircraft 1 - 200, 322 and 323 having ASC 192.

B. Place BATT switch to NORM and pull No. 1 and 2 normal bus tie relay circuit breakers alternately, note DC bus voltmeter indication. Voltage should be approximately 24 volts with either circuit breaker in, zero volt with both breakers out. Reset both breakers.

C. Depress R BATT DISC switch (indicator light should come on), note left battery voltage on DC bus voltmeter. This test left normal battery relay. Depress R BATT DISC switch (light out). Depress L BATT DISC to test right normal battery relay. Disconnect both batteries (both lights on), DC voltmeter should indicate zero volts. If all the indications were normal, left and right battery normal and both bus tie relays would be operational.

D. Place BATT switch to EMER and repeat Step C. to check left and right battery emergency and single emergency bus tie relays.

E. Ensure L and R BATT DISC lights are out. Place the BATT switch in OFF.

NOTE: Step F. through M. apply to Aircraft 1 - 200, 322 and 323 not having ASC 192.

F. Pull No. 1 and No. 2 normal bus tie relay circuit breakers.

G. Set BATT switch to NORM and engage No. 1 normal bus tie relay circuit breaker. DC bus voltmeter on upper overhead panel should read battery voltage (approximately 24V). This checks No. 1 normal bus tie relay and associated circuitry.

H. Pull No. 1 normal bus tie relay circuit breaker. DC bus voltmeter indication should drop to 0 volt.

I. Engage No. 2 normal bus tie relay circuit breaker. DC bus voltmeter should indicate battery voltage (approximately 24V). This checks No. 1 normal bus tie relay and associated circuitry.

J. Reset No. 1 normal bus tie relay circuit breaker.

K. Place BATT switch to EMER DC bus voltmeter should indicate battery voltage (approximately 24V). Pull Emergency DC bus tie circuit breaker. DC bus voltmeter should drop to zero volt. This checks the emergency bus tie relay and associated circuitry.

L. Reset Emergency DC bus tie circuit breaker.

M. Set BATT switch to OFF.

N. Return all switches and circuit breakers to normal position.

2. Battery Normal and Emergency Relay — Removal / Installation

A. Removal

(1) Remove battery on side involved. (See Battery — Removal / Installation). (See Section 24-2-6)

(2) Locate relay to be removed in aft section of battery compartment.

- Normal battery relay is a black, square unit.
- Emergency battery relay is a conventional contactor with a cylindrical solenoid enclosure.

(3) Disconnect, insulate, and identify electrical leads.

(4) Remove and retain relay mounting hardware.

(5) Remove relay.

B. Installation

(1) Install relay using hardware previously removed.

(2) Connect electrical leads as previously identified to appropriate terminals and ensure good electrical connection.

(3) Install battery previously removed. (See Battery — Removal / Installation). (See Section 24-2-6)

NOTE: All circuit breakers in step D. are located on Copilots circuit breaker panel.

(4) Check operation of relay involved using either NORM or EMER BATT switch position (if normal relay is involved place BATT switch in NORM position and pull (open) L BATT CONT or R BATT CONT circuit breaker. If emergency relay is involved place BATT switch in EMER position and pull (open) L BATT EMER or R BATT EMER circuit breaker), check DC BUS VOLTS voltmeter in cockpit for voltage.

(5) Place BATT switch to OFF position after completion of check, then disconnect batteries if no longer required.

NOTE: The following steps are applicable to Aircraft 1 - 200, 322 and 323 having ASC 192.

(6) Check operation of relay involved using either NORM or EMER BATT switch position (if normal relay is involved place BATT switch in NORM position, if emergency relay is involved place BATT switch in EMER position), and depress L BATT DISC or R BATT DISC switch (as applicable). Check DC BUS VOLTS voltmeter in cockpit for voltage.

(7) Place BATT switch to OFF after completion of check, battery case has been isolated from aircraft ground. Removal of battery connector is not necessary but may be accomplished if desired. (Do not disconnect batteries with BATT DISC switches.)

3. Normal Bus-Tie Relays (1 and 2), Emergency Bus-Tie Relay — Removal / Installation

A. Removal

(1) Remove right side of inverter relay box.

(2) Remove, identify, and insulate wires from relays.

(3) Remove mounting hardware from relays.

(4) Remove relays from chassis.

B. Installation

(1) Install relays on chassis.

(2) Install mounting hardware on relays.

(3) Connect identified wires to relays and check connections.

(4) Perform Essential DC Bus — Operational Test, this Section.

(5) Inspect area for foreign objects, security of all attachments.

(6) Install right side of inverter relay box.

ENGINE-DRIVEN DC GENERATOR — DESCRIPTION / OPERATION

1. Description

DC power generation is provided by two 300 amp engine-driven generators, one on each accessory gearbox. These generators are geared at a ratio to engine speed such that they will produce 100 percent rated electrical output at between 7500-15,000 engine rpm. Since ground idle of the Dart engine is from 6500-7500 rpm, the generators will produce nearly full power at idle. Each engine generator system is made up of the generator, a differential type reverse current cutout relay, a voltage regulator, an overvoltage relay, a generator ON-OFF-RESET switch, a generator TRIP switch and a combination of volt ammeter. Voltage build-up at the generator is always of correct polarity because of a tickler circuit built into each overvoltage relay. In addition, both systems are tied together through an equalizer circuit. The equalizer circuit serves two functions. When one generator is on the line and the second generator switch is placed in the ON position, the equalizer circuit acts to create the differential in voltage, in the proper direction, necessary for the second generator to come on the line. When both generators are functioning and on the line, the equalizer circuit acts to equalize the load between both units.

Each generator is cooled by ram air obtained from an air inlet scoop on the left side of the nacelle. The scoop is attached to the generator shroud assembly at the forward end of the generator by a blast tube. Cooling air passes through the generator and out the aft end to an exhaust outlet on the left side of the nacelle via an exhaust tube. A thermal switch is installed in the exhaust tube so that when air passing through the generator reaches an abnormal temperature, the switch closes and the generator overheat warning light (left or right) lights up. This switch is set to close at 300 ± 10°F.

NOTE: The main dc generators for Gulfstream I aircraft are Jack and Heintz No. 30007-003 or Bendix No. 30E20-41A. These generators can be considered interchangeable and can be freely mixed.

The generator control switches and meters are located in the cockpit on the upper overhead panel (See Figure 1) There are two dual purpose meters, one for each generator, which indicate both amps and volts on separate scales. The ammeter scale is 0-300 amps. The voltmeter scale is 15-34 volts. The voltmeter section is wired to the G + terminal of the voltage regulator. This terminal in turn is connected to the GEN terminal of the reverse current cutout relay, thereby sensing generator potential. The ammeter section is connected across the shunt in the ground leg of the generator, thereby sensing all current flow.

Each generator system has two manually operated control or selector switches. The manual TRIP switch is located on either side of the battery switch on the upper overhead panel in the group labeled ELECTRICAL MASTER SWITCHES. These switches are the single throw single pole type with momentary on, spring loaded to off. The on position is labeled TRIP. This switch may be actuated separately in case only one system needs to be tripped. In case of emergency, when the GANG BAR is pulled, the left and right generator TRIP switches are two of the five switches actuated.

The other switch in each generator system is the generator control switch. This is a double pole double throw switch with a center OFF position. The lower on position is momentary and is labeled RESET. The switches themselves are labeled L. GEN and R. GEN and are located either side of the EXT PWR switch in the group placarded as ELEC SUPERVISORY SW on the upper overhead panel. The on-off portion of the switch is for normal control of the generator system. The RESET position is used after the system has been tripped. The generators must not be placed on the line until a minimum of 8000 rpm is attained.

In addition to the above control switches, a switch actuated by the FIRE PULL T-handle is in the generator control circuit. The FIRE PULL T-handle is located on the eyebrow panel, and there is a separate handle for each engine. Pulling the handle breaks the generator control circuit (this is only one function of the T-handle). In effect, this accomplishes the same thing as switching the generator control switch to the off position.

Each of the generators have a GEN HOT and GEN OFF light on the master warning display panel.

The GEN HOT light (left or right) is controlled by an overheat switch mounted in the exhaust end of the generator cooling duct. When the air passing through the exhaust duct reaches 300 ± 10°F the switch will close, completing a circuit causing the left or right GEN HOT light to light up.

The GEN OFF (left or right) indication circuit informs the crew when a generator is not on the line. These lights are controlled by the operation of the particular generator's reverse current cutout relay. These are positive indication, when ON, that the particular generator is not connected to the bus.

Note on the internal detail schematic of the reverse current cutout relay, Figure 3, that a flexible wire is connected to the main contactor bar. From there it is connected to the IND terminal. This contactor bar has no power on it in the open position. As soon as the main contactor closes the contactor bar bridges the GEN and BATT contacts becoming part of the circuit. The ND terminal in turn has power on it. This power is utilized to operate the generator bus control relay for its generator system. Note on the monitor bus and generator indication schematic, Figure 2, that the IND terminal is connected to the coil of the generator bus control relay. When the reverse current relay is open the generator bus control relay is relaxed. With this relay relaxed the power from the essential bus tied to B2 terminal is fed to B3 terminal which is wired to the GEN OFF light in the master warning system. The light is on in this condition. When the reverse current relay closes, power is applied to the generator bus control relay coil and the contacts shift. Power from B3 is transferred to B1 which is an open terminal and the GEN OFF light should go out.

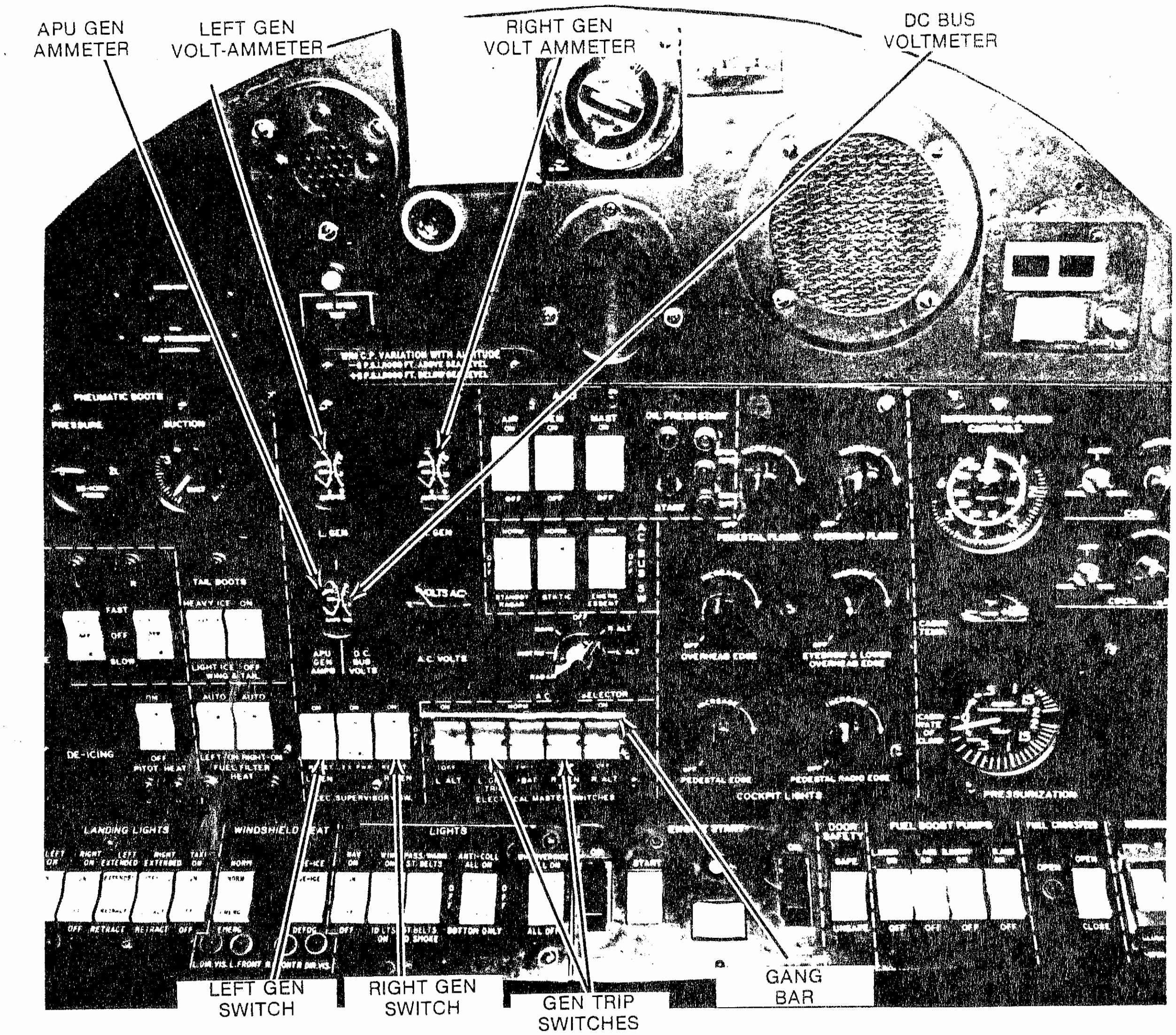
2. Operation

(See Figure 3)

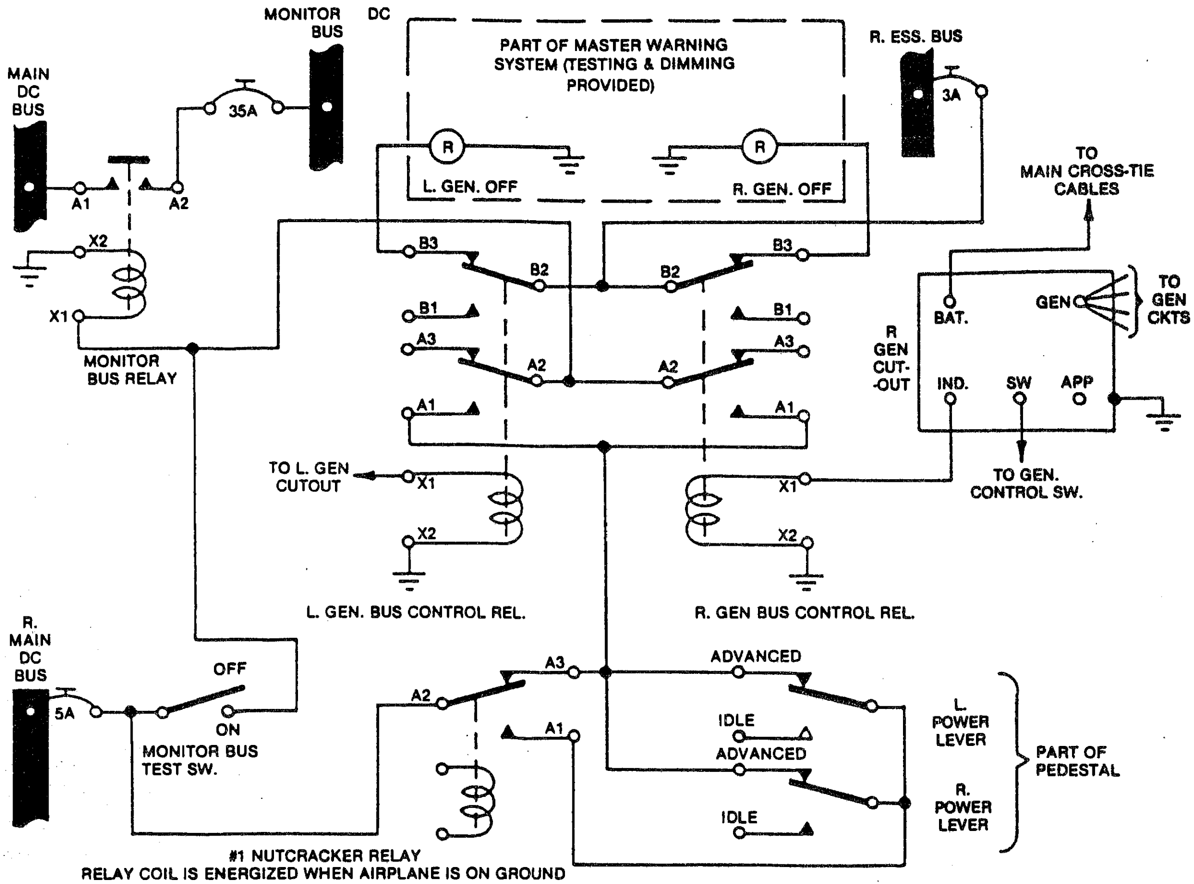
NOTE: Aircraft 1 - 200, 322 and 323 having ASC 231 have four circuit breakers installed, in each nacelle, to protect the wiring and equipment. (See Figure 4)

The generator armature is mechanically connected to the engine accessory gearbox and turns at an rpm directly proportional to engine speed. Voltage is provided by the generator when its armature is rotated. The positive output is from the B terminal of each generator. The principle of operation of these generators is similar to that used in most dc installations, where the dc generator is the normal source of power. In a generator, voltage is produced by electro-magnetic action. The armature is rotated in a magnetic field of alternate north and south poles. This generates a voltage in each of the armature coils, first in one direction, then in the other. To counteract this effect and to produce a voltage of constant polarity (B positive and E negative), the armature-coil connections are periodically reversed by the commutator and brushes, which must be set to do this when the armature-coil voltages are near the zero value through which they pass in changing from one direction to the other. This process called commutation, is accompanied by sparking if the brush setting is wrong.

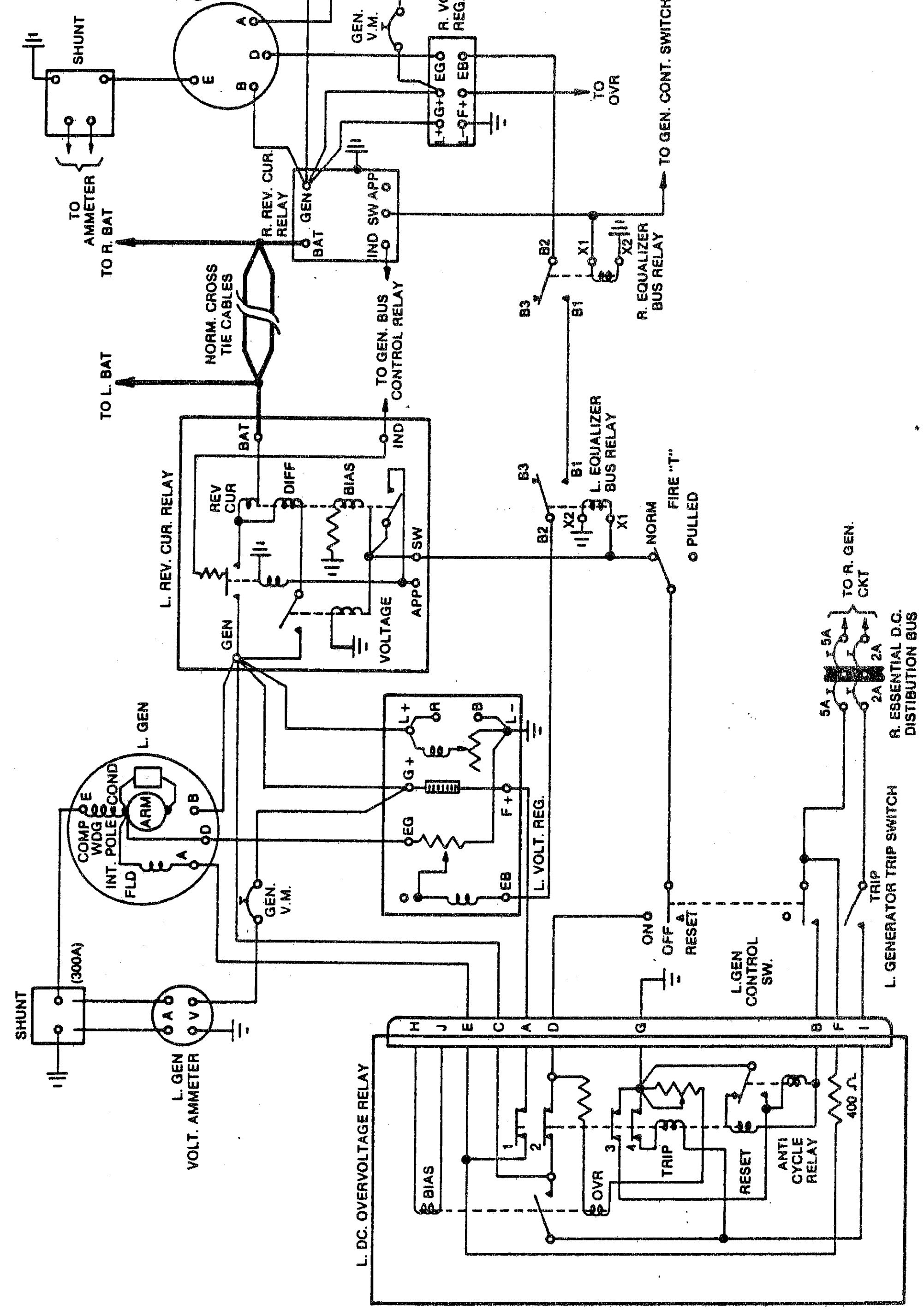
The magnetic field is established first by the magnetism of the field poles, usually referred to as residual magnetism. As a generator is started up, it generates a small voltage due to the initial magnetism, and this voltage produces a current in the field through the field circuit which increases the magnetism and therefore the voltage. This produces more current in the field, higher voltage, more field current and so on. This process called build up continues until the voltage and field current finally reach a point where one is just sufficient to produce the other. Speed of armature rotation is also important. At any moment, the voltage generated depends on the field currents and on the so-called saturation curve. which is a plot of volts versus field amps at any particular speed. At any moment, the voltage required to produce a given current in the field depends on the resistance of the field circuit. When the voltage and current reach a normal point for a given speed, they are maintained at this point by inserting a voltage regulator in the shunt field circuit to increase or decrease the resistance of this circuit as necessary. When a generator is loaded, the currents flowing in the armature coils tend to reduce and to distort the magnetic field set up by the current in field coils. To prevent sparking at the brushes due to the condition of different commutation voltages in the armature coils, compensating and inter-pole fields are provided. Load current is conducted through these field coils to counteract the effects of load currents in the armature coils. In this way, virtually sparkless operating at all loads and speeds within the rating is possible with proper brush settings.



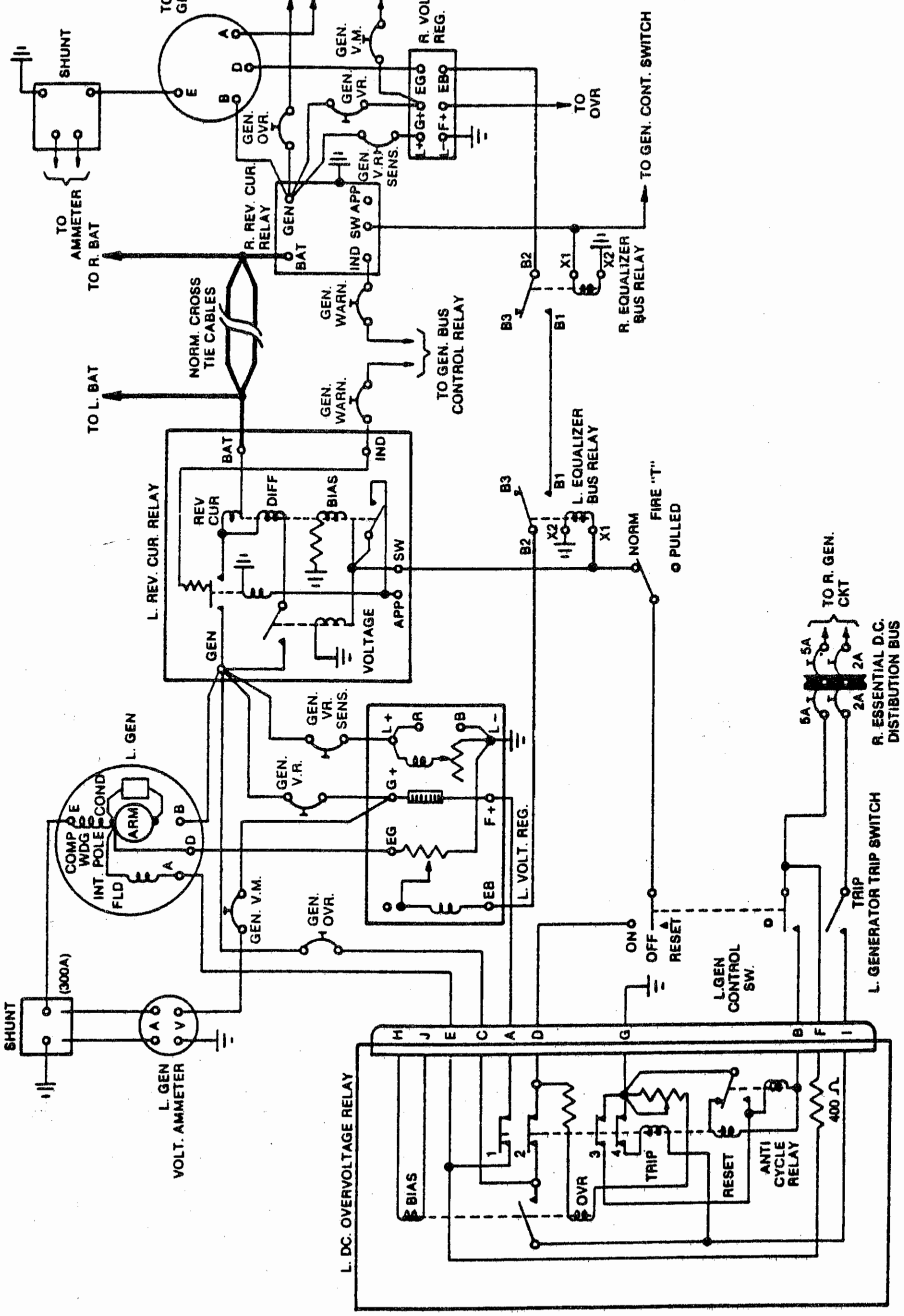
DC Generator Metering and Control Switch Locations
Figure 1.



DC Power Monitor Bus and Generator Indication — Simplified Schematic
Figure 2.



DC Power Generator Circuits — Simplified Schematic
Figure 3.



DC Power Generator Circuits — Simplified Schematic
(Aircraft 1 - 200, 322 and 323 Having ASC 213)
Figure 4.

ENGINE-DRIVEN DC GENERATOR — MAINTENANCE PRACTICES

1. **General**

   A. Commutator film is deposited by running in the generator at a gradually increasing loads over a period of time. Once deposited, the film is maintained by normal operation. The film is normally put in place during final bench check by the manufacturer.

   B. The commutator film can be destroyed by solvents (MEK, etc.) and by oil or hydraulic fluid. Destruction can also be caused by high speed operation at light loads. High altitude operation aggravates any commutator problem.

   C. Compliance with the following will prevent many generator problems:

      (1) Properly parallel the generators on the earliest engine run possible. (See DC Generator Paralleling — Adjustment, see Section 24-2-2.

      (2) Inspect generator air out vent for brush dust periodically. Heavy deposits indicate abnormal wear.

      (3) Protect the generator and cooling ducts from fluids and vapors. If the accessory section, accessory

      (4) section Cowling or engine cowling is to be washed down, protect the generator or remove it. If contamination is suspected, blow generator out with clean air and observe closely for signs of brush wear.

      (5) During engine runs, ensure at high engine RPM generator is loaded. If the aircraft bus load is insufficient to get both generators on line, either increase bus load, or alternately select generators so that the high engine RPM has the loaded generator.

      (6) Never run the generator unloaded. During paralleling adjustments, strictly limit period of operation and observe speed limitations.

2. **DC Generators — Removal / Installation**

   A. Removal

   CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

   IF AIRCRAFT ELECTRICAL SYSTEM MUST BE ENERGIZED DURING TIME GENERATOR IS REMOVED, PULL APPROPRIATE GENERATOR CONTROL CIRCUIT BREAKER. TAPE AND STOW GENERATOR LEADS BEFORE ENERGIZING ELECTRICAL SYSTEM.

      (1) Disconnect external power and place BATT switch OFF.

      (2) Gain access to generator by removing top accessory gearbox access panel.

      (3) Disconnect and tag electrical leads. Label leads to help during installation.

      (4) Disconnect generator cooling air inlet and exhaust ducts.

      (5) Index mark generator and gearbox to facilitate installation.

      NOTE: If a replacement generator is to be installed, note position of generator terminals and cooling air outlet.

      (6) Depress spring-loaded lock on the attaching ring and loosen nut on quick disconnect clamp.

      (7) Support generator and remove generator from gearbox.

   B. Installation

      (1) If replacement generator is to be installed, note positioning of cooling air outlet and mounting adapter. Transfer mounting adapter and cooling air inlet and exhaust shrouds from old generator. Torque nuts securing mounting adapter to 180 - 200 inch pounds.

      NOTE: Vespal spline modification may be installed on all aircraft utilizing Dowty Rotol and Bendix Service Bulletins, this Section.

(2) If Vespel spline is installed, inspect spline adapter for cleanliness. See DC Generator Spline Adapter (Vespel Spline) — Removal / Installation procedure if replacement is required.

(3) Perform Step (a) below for Vespel spline or Step (b) below for standard spline.

(a) Position generator so that splined shaft starts to engage spline adapter: push generator against mounting pad.

(b) Lubricate generator drive shaft (See DC Generator Spline — Service / Lubrication, this Section). Position generator so that splined shaft starts to engage with gearbox spline, push generator against mounting pad and rotate approximately 15° clockwise.

(4) Apply light coat of engine gearbox oil to quick release mounting adapter, receiving flange and bolt threads on quick disconnect clamp.

(5) Install generator on gearbox and secure with a quick disconnect clamp. Tighten nut on clamp from slack position to a torque value of 100 inch pounds. Loosen nut and repeat Step two additional times to ensure correct seating. Lock nut with spring-loaded lock device.

(6) Connect electrical lead connectors (3/8 inch studs) and remove tags. Torque to 95-110 inch pounds.

(7) Connect small field lead connectors and torque to 30-40 inch pounds.

(8) Connect cooling air inlet and exhaust ducts.

(9) Inspect area, for foreign objects, damage and leaks.

(10) Check electrical operation and perform DC Generator Paralleling — Adjustment, see Section 24-2-2.

(11) Inspect area for presents of foreign objects, security of all attachments.

(12) Install top accessory gearbox access panel.

3. DC Generator Spline Adapter (Vespel Spline) — Removal / Installation

A. Removal

(1) Remove generator. See DC Generator — Removal / Installation, this Section.

(2) Using spline removal tool, (P/N 11595EP30101), push removal key through adapter until cleared of adapter.

(3) Rotate key until indexer lock can be inserted to secure removal key, index lock and spline adapter together.

(4) Slide spacer over threaded portion of removal tool and secure with special washer and nut.

(5) Tighten nut until spline adapter is clear of quillshaft. Discard spline adapter.

B. Installation

WARNING: USE CLEANING SOLVENTS IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAME.

(1) Prior to installation, ensure splines on drive and adapter are clean. (Clean with cloth dampened with cleaning solvent and 1/2 inch stiff nylon bristle brush).

NOTE: If for any reason an adapter is removed, a new adapter should be used. This applies even after partial insertion and removal. Adapter should be replaced at each generator replacement.

(2) Mount insertion tool (P/N 110684/1-1) to gearbox adapter.

(3) Coat entire outside surface of spline adapter with Dow Corning DC-200-1000 silicone oil, (specification VV-D-1078 grade 1000 CS) or equivalent.

(4) Install adapter approximately 1/4 inch into gearbox shaft.

(5) Advance insertion tool handle until adapter is 0.100 inch recessed into gearbox shaft.

(6) Remove insertion tool.

(7) Install generator. See DC Generator — Removal / Installation, this Section.

4. DC Generator Spline — Lubrication

NOTE: The following procedure applies to aircraft with standard spline, (not having Vespel spline).

A. Remove electrical power from aircraft.

B. Open accessory gearbox top access panel.

CAUTION: IF AIRCRAFT ELECTRICAL SYSTEM MUST BE ENERGIZED DURING TIME GENERATOR IS REMOVED, PULL APPROPRIATE GENERATOR CONTROL CIRCUIT BREAKER. TAPE AND STOW GENERATOR LEADS BEFORE ENERGIZING ELECTRICAL SYSTEM.

C. Remove generator. See DC Generator — Removal / Installation, this Section.

WARNING: USE CLEANING SOLVENTS IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAME.

D. Clean quillshaft and generator drive shaft with cloth dampened with cleaning solvent and a 1/2 inch stiff nylon bristle brush removing all traces of old grease.

E. Inspect quillshaft and generator drive shaft splines.

Generator drive shaft - spline teeth should show fairly uniform wear. There should be no radial nicks, pieces missing, or knife edges.

Gearbox shaft - spline teeth should show fairly uniform wear. There should be no radial nicks, pieces missing, or knife edges. If quillshaft or quillshaft condition appears abnormal, contact Dowty Rotol or an engine overhaul agency.

NOTE: Splines may be slightly rusty. this is not a cause for rejection if spline condition is satisfactory.

F. Apply a light coat of grease (See Chapter 12 for grease specification) to generator coupling.

G. Install generator. See DC Generator — Removal / Installation, this Section.

NOTE: If brushes must be changed due to wear beyond acceptable limits, the generator(s) must be removed and sent to a certified repair station, or returned to the manufacturer for installation and seating of new brushes.

H. Remove brushes(see note above) and check wear limit lines scribed on each brush (See Figure 201).

NOTE: If brushes are worn beyond limits the generator must be removed and brushes replaced by a certified facility or the manufacturer.

I. Check condition of commutator.

J. Inspect for presents of foreign objects, security of all attachments.

K. Install Brushes .

L. Close access previously opened.

M. Perform DC Generator Paralleling — Adjustment, see Section 24-2-2.

5. DC Generator Brush — Inspection

NOTE: If generator is installed on gearbox it Is permissible to check the accessible two sets of brushes for length and condition and visually check remainder for comparison. If visual inspection remainder of brushes indicates variation from two actual sets which were pulled and measured generator should be removed (See Removal / Installation this section) and each set of brushes given detailed inspection. If generator is already off gearbox or is being removed for some other purpose, check all brushes.

A. Gain access to the dc generator by removing the appropriate gearbox access cover.

B. Remove safety wire, disconnect and remove brush band cover.

CAUTION: USE EXTREME CARE WHEN REMOVING BRUSHES TO PREVENT DAMAGE.

C. Remove brushes (see note above) and check wear limit lines scribed on each set of brushes. (See Figure 201). If brushes are within limits, reinstall brushes.

CAUTION: IF BRUSHES ARE WORN BEYOND LIMITS GENERATOR MUST BE REMOVED AND BRUSHES REPLACED BY CERTIFIED FACILITY OR THE MANUFACTURER.

D. Check condition of commutator.

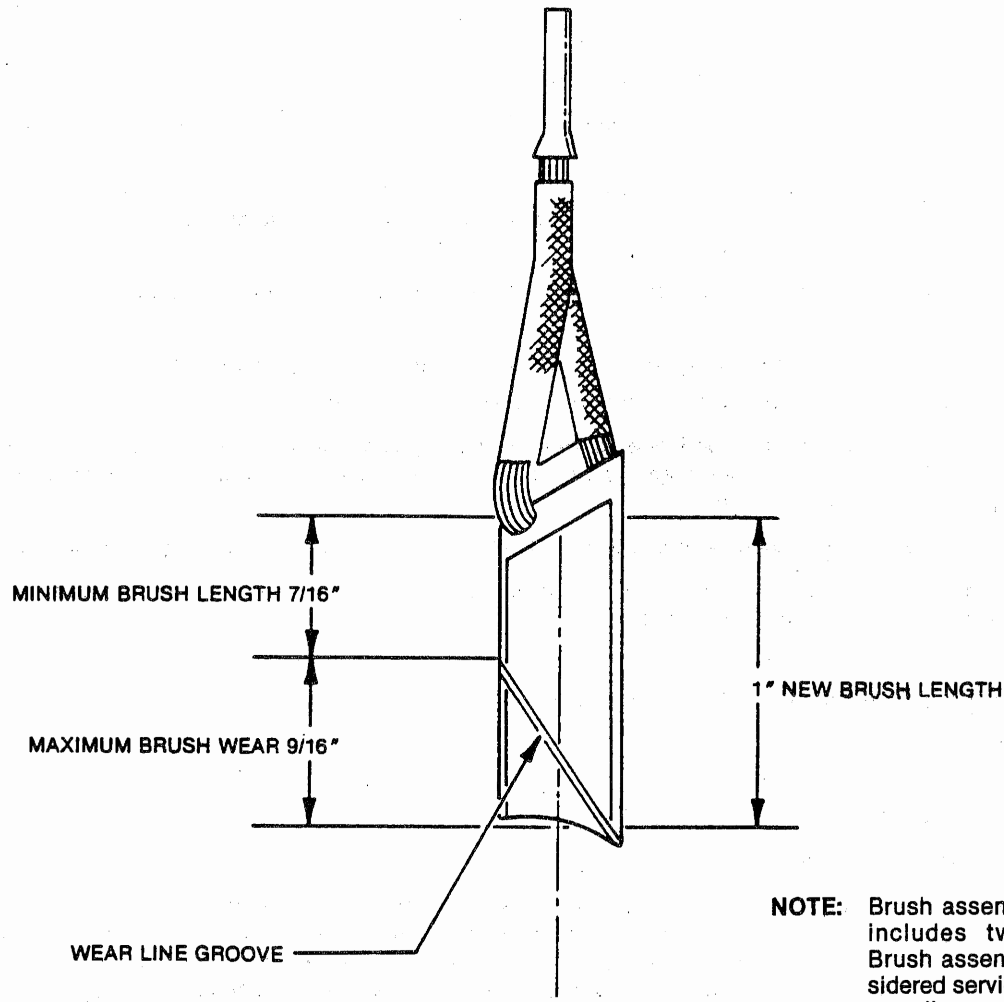
E. Inspect area for presents of foreign objects, security of all attachments.

F. Install brush band cover and safety.

G. Inspect for presence of foreign objects then close access previously opened.

H. Perform DC Generator Paralleling — Adjustment, see Section 24-2-2.

NOTE: Brush assembly shown includes two halves. Brush assembly is considered serviceable until wear line groove on both halves is gone.



NOTE: Brush assembly shown includes two halves. Brush assembly is considered serviceable until wear line groove on both halves is gone.

DC Generator Brush Length
Figure 201.

VOLTAGE REGULATORS — DESCRIPTION / OPERATION

1. General

Since the generator output voltage is determined by the generator field voltage, control of the field voltage provides a regulated output. In this manner, each voltage regulator functions to regulate its generator output under varying engine speed and loading conditions. Since the engine-driven dc generators are paralleled, a bus equalizing system is provided which, working through both generator's voltage regulators, provides an even distribution of the dc load between the generators. The APU generator is not tied in to the bus equalizing system.

2. Description

(See Figure 1)

Three voltage regulators are located in the fuselage below the radio rack. The forward unit is the left generator voltage regulator, the middle unit is the right generator voltage regulator and the aft unit is the APU generator voltage regulator. Each unit consists of a shock-mounted base and a plug-in type carbon pile voltage regulator. The base provides the electrical connections to the aircraft electrical system and the means of mounting and electrically connecting the voltage regulator plug-in assembly. This plug-in assembly consists primarily of a spring-loaded, electromagnetically-operated armature which exerts a pressure on a carbon pile, which is a variable resistor, and a voltage adjusting potentiometer. The equalizing adjusting potentiometer and the voltage test jacks are parts of the base assembly.

3. Operation

The B terminal of the generator cable is connected to the GEN terminal of the reverse current cutout relay. All the generator system control circuits sense and pick up generator voltage at this point. The generator field circuit starts at this terminal, connects to the G + terminal on the voltage regulator, through the carbon pile, out the F + terminal. From there, the current flows through the overvoltage relay, in through pin A, across contact No. 1 and out pin E. Pin E on the overvoltage relay is wired to the A terminal on the generator. The current then flows through the generator field windings ending at the negative brush. The carbon pile voltage regulator is designed to prevent changes in generator output voltage that are caused by changes in generator speed and load conditions. The unit is essentially an automatic generator field variable resistor that maintains a constant output voltage by controlling the generator field current.

The regulator magnet coil shunt winding is connected through an adjustment rheostat across the dc output circuit, from the GEN terminal on the reverse current relay to L+ terminal on the regulator, through the shunt coil and rheostat and to ground through L−. The carbon pile or regulating resistance is connected in series with the generator field circuit. When the generator voltage rises above the voltage at which the regulator has been set, the current in the magnet coil shunt winding increases. This increases the magnetic attraction of the core on the armature and relieves the pressure of the diaphragm armature on the carbon pile. The carbon discs tend to separate and increase the resistance of the field circuit. The result is a decrease in generator field current and a corresponding decrease in output voltage. When the generator voltage falls below the voltage at which the regulator is set, the reverse action takes place. The current in the magnet coil shunt winding decreases, thereby weakening the magnetic attraction of the core on the armature. This permits the diaphragm armature assembly to compress the carbon pile, thereby decreasing the resistance in the generator field circuit to increase the generator field current and output voltage. Normal regulator setting for the engine driven generators is 27.7 $\pm$ 0.2 volts measured at the regulator. Pin jacks for the voltmeter are provided and are coded red for positive and black for negative.

4. Bus Equalizing System (Engine-Driven Generators Only)

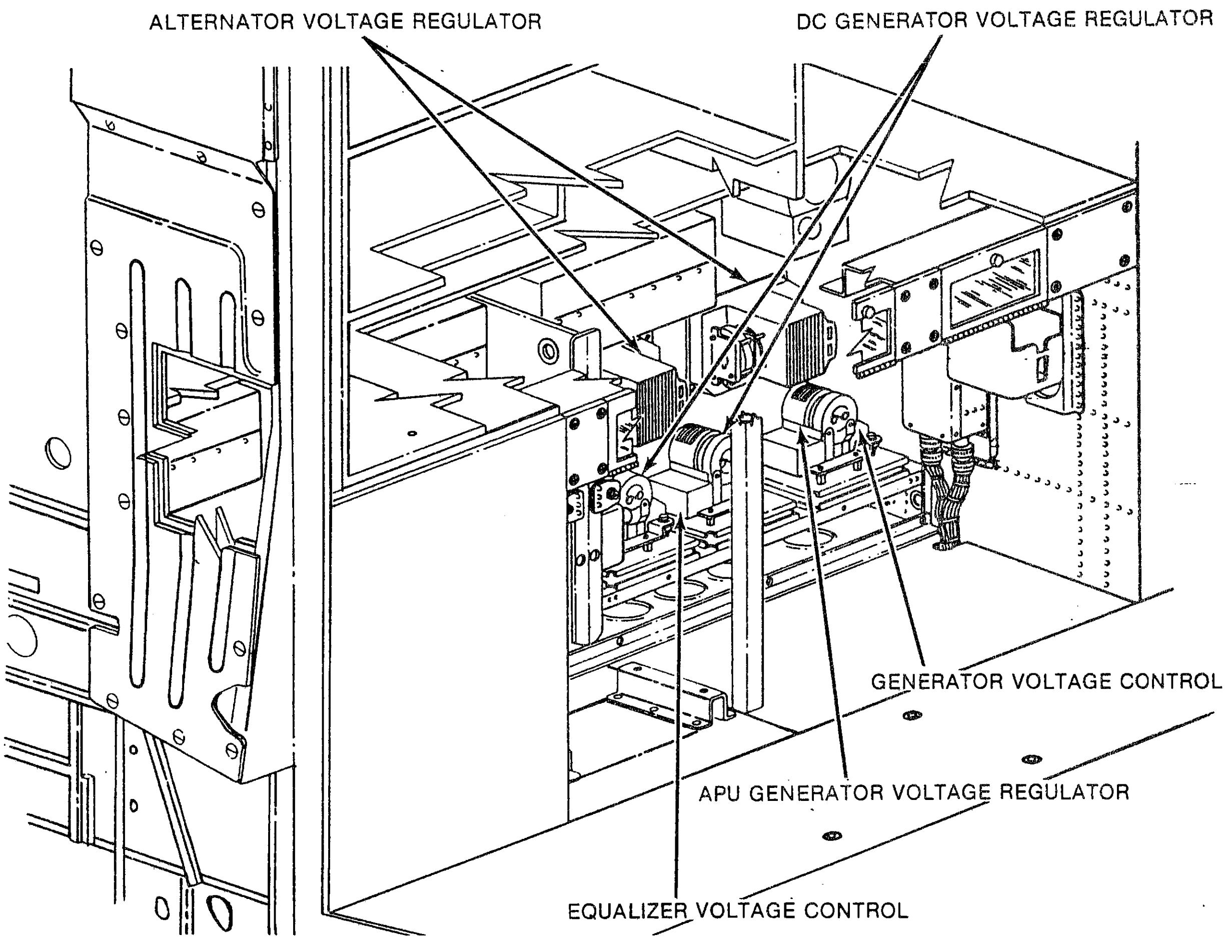
Assume that the left engine and its generator is functioning normally with an output of 27.7 volts at 150 amps. The right engine is running above 8000 rpm, voltmeter reads 27.8 volts. We now set the right generator control switch to ON. Note that the voltage of the right generator is 0.1 volt above bus voltage. The generator voltage must exceed the bus voltage by 0.35 to 0.65 volt before that relay will allow the generator to come on the line. In this case, it does not. At this point even though we have selected both generators, only one is on the line and carrying a load. An equalizer circuit is provided to control both generators in a way which will divide the loads evenly.

The generator control switch in the ON position will energize the equalizer bus relay. There is a relay for each generator. Since the relays are in series, both generators must be selected before the equalizer circuit is completed. The equalizer circuit is between the two negative brushes. The negative brushes are electrically above ground since the interpole and compensator windings and the ammeter shunt are between the brush and ground. The point above ground is determined by the generator current flow. The greater the current flow, the more voltage difference occurs. The potential at the negative brush, of the higher loaded generator will therefore be lowest. If both brushes were connected, current would flow from the generator with the lighter load to that of the heavy load.

The equalizer circuit between the brushes, runs from the brush to the D terminal of the generator which is wired to the EG terminal of the regulator. Current then flows through the equalizer resistor out to ground through L−. The equalizer bus circuit is completed through the equalizer bus relays provided both are energized. The bus is connected to the EB terminal of the voltage regulators at either end. A load equalizing coil, part of the regulator magnet coil assembly, is connected at one side of the bus at the EB terminal, the other end is connected to the equalizer resistor by an adjustable wiper. The resistor-wiper combination is in effect a potentiometer. Current flow from the brush with high potential (no load or lighter load) to the lower potential brush will be through the equalizer coils in the regulators in a direction which will increase the voltage of the no load or lighter load generator. This flow through the regulator of the other generator being in the opposite direction will decrease its voltage. By this action, a differential voltage is produced which will allow the second generator to come on the line. The equalizing system then balances the loads. This generator is now operating in parallel with the first generator.

When two generators are operated in parallel, the equalizer terminals D are connected to paralleling resistors and equalizer coils in the voltage regulators, and to an equalizer bus. If one generator takes too much of the load, there appears across its paralleling resistor, a voltage high enough to force current through its equalizer coil to the equalizer bus, and through the other equalizer coils. The coils then act to reduce the field current and load on the one generator, and to increase the field current and load on the others until loads are equal. Circuit function is the same as described in detail above.

The generator equalizing voltage is regulated by the adjustable slide resistor on the regulator. In parallel operation, increasing equalizing voltage decreases the output current of the generator circuit being adjusted and decreasing the equalizing voltage increases the generator's output.



DC Voltage Regulator Location
Figure 1.

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VOLTAGE REGULATORS — MAINTENANCE PRACTICES

1. Voltage Regulator / Regulator Base — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

NOTE: The Phoenix Aerospace static regulators (P/N VR-1010-24-1A) should not be mixed with carbon pile regulators (P/N 1588-2E or E1588-2E).

A. Removal

(1) Gain access to voltage regulator and/or bases by removing aft access panel below radio rack, entrance compartment, right side.

(2) Release regulator by holding holddown clips outward and removing regulator from base.

NOTE: If regulator is to be replaced, disregard steps (3) thru (5).

(3) Disconnect, tag and insulate leads from base of terminals.

(4) Remove base mounting hardware.

(5) Remove base.

B. Installation:

(1) If base is replaced burnish mating on underside of base to ensure electrical bond to mounting structure.

(2) Install base using hardware previously removed.

(3) If the Phoenix Aerospace static dc voltage regulator is being installed, clean the equalizer voltage control located on each regulator mount and then adjust to 0.4 ohms of resistance between terminals EG and O on regulator base.

(4) Remove tags and insulation and connect leads. Ensure tight connections.

(5) Hold back clips and install regulator; ensure spring clips secure pile firmly in place.

(6) If left or right dc generator regulator was replaced, Perform DC Generator Paralleling — Adjustment, this Section.

(7) Inspect for presence of foreign objects then close access panel previously opened.

2. Engine Voltage Regulator — Adjustment

A. Gain access to left or right generator voltage regulator under aft section of radio rack just forward of Fuselage Station 191 entrance compartment, right side, remove access panel.

B. Start and run engines using normal procedures.

C. Place left and right DC GENERATOR switches on (8000 rpm minimum).

NOTE: 20 minute wait not required if static regulators are installed.

D. Run engines at 11,000 rpm with generators on for at least 20 minutes to allow regulators to warm up and stabilize.

E. Place left DC GENERATOR switch off (DO NOT TRIP).

F. Check voltage at left generator voltage regulator pinjacks with precision voltmeter; if required, set to 27.7 $\pm$ 0.2 V dc.

NOTE: Some operators have set their dc generator regulators to 28.5 $\pm$ 0.2 V dc. On these aircraft, set regulators accordingly.

G. Wait a few minutes after each voltage adjustment to let regulator stabilize until final satisfactory reading is achieved.

NOTE: If voltage regulator fails to hold voltage within limits, perform a stability check. (Carbon Pile only.)

H. Turn Left Generator switch On.

I. Turn Right Generator switch Off (DO NOT TRIP).

J. Repeat Steps 2. F. thru 2. G. for right voltage regulator.

K. Turn Right Generator switch On.

L. Remove test voltmeter and continue with DC Generator Paralleling — Adjustments.

3. DC Generator Paralleling — Adjustment

NOTE: It Is assumed that voltages have been checked/adjusted and regulators are warm and stable.

A. Apply heavy load to dc system, suggested load as follows:

(1) Boost pumps ON (left/right main and aux).

(2) Inverters E and A ON (ESS AC SEL switch to EMER, MAIN AC SEL switch NORM).

B. Compare generator ammeters for load sharing, if unbalanced adjust high generator voltage regulator paralleling rheostat to decrease load and low generator paralleling rheostat to increase its load, until balance is attained.

NOTE: Make adjustments in small increments.

C. Remove load as desired.

D. Shutdown engines, using normal procedures, see Chapter 71.

E. Inspect for of foreign objects then close access panel previously opened.

4. APU Voltage Regulator — Adjustment

A. Gain access to the APU voltage regulator, located under the radio rack - aft section - just forward of Fuselage Station 191, entrance compartment:

B. Remove access panel.

C. Energize essential dc bus.

D. Start and run APU using normal. See procedures this section.

E. Pull No.1 EMER BATT circuit breaker to avoid generator belt stress.

F. Ensure engine-driven dc generators and external power is off.

G. Set APU GEN switch on.

NOTE: 20 minute wait not required if static regulator is installed.

H. Observe cockpit APU ammeter for generator loading when amperage stabilizes and after waiting at least 20 minutes for regulator to warm up, check APU GEN VOLTAGE with PRECISION DC VOLTMETER using pinjacks on regulator. Adjust to 27.0 volts as necessary at REGULATOR POTENTIOMETER.

NOTE: Some operators have engine-driven dc generator voltage set to 28.5 $\pm$ 0.2, in this event set APU generator voltage to 27.8 $\pm$ 0.2..

CAUTION: DO NOT SET APU GENERATOR VOLTAGE TO 27.8 IF ENGINE-DRIVEN GENERATORS ARE SET TO 27.7.

I. Remove test meter.

J. Shut down APU generator and APU using normal procedures, this Section.

K. Reset No.1 EMER BATT circuit breaker pulled in Step E.

L. Set BATT switch off.

M. Inspect for foreign objects then close access.

5. **DC Generator Voltage Regulator Stability — Functional Test**

NOTE: The following test Is not required If static regulators are Installed.

A. Gain access to left or right generator voltage regulator under aft section of radio rack just forward of Fuselage Station 191 entrance compartment, right side; remove access panel.

B. Start and run engines using normal procedures, see Chapter 71.

C. Place left and right DC GENERATOR switches on (8000 rpm minimum).

D. Run engines at 11,000 rpm with generators on for at least 20 minutes to allow regulators to stabilize and warm up.

E. Place left DC GENERATOR switch off (do not trip) and perform the following stability check:

NOTE: The following check is performed with a pair of high Impedance ear phones having a one microfarad capacitor in series with one lead.

(1) Connect earphone leads to terminals F and G of left voltage regulator.

(2) With earphones on gradually advance power lever and listen for a steady, smooth hum or roaring noise Indicating a stable voltage regulator. If a succession of popping noises, Indicating instability, is heard, replace the regulator.

NOTE: If Instability Is encountered with replacement voltage regulator, check generator leads for security of connections, foreign particles In voltage regulator spring contacts, or remove plug-In assembly and check both sides of contacts.

(3) Place left DC GENERATOR switch on and right DC GENERATOR switch off.

(4) Add load to generator by energizing equipment normally not energized. Listening In earphones, check for instability as loads are applied and removed.

NOTE: Instability should not be confused with the click heard each time a load Is applied or removed from the generator.

(5) Disconnect earphone leads when check Is completed.

(6) Shut down engines, using normal procedures, see Chapter 71.

(7) Inspect for presence of foreign objects then close access panel.

6. **APU Generator Voltage Regulator Stability — Functional Test**

NOTE: The following test Is not required If static regulators are Installed.

A. Gain access to APU generator voltage regulator under aft section of radio rack just forward of Fuselage Station 191 entrance compartment, right side; remove access panel.

B. Start APU using normal procedures. (See Chapter 49)

C. Set APU generator switch on.

D. Run APU with generator on ta least 20 minutes to allow regulator to stabilize and warm up.

E. Set generator switch off and perform following stability check:

NOTE: Following check is performed with a pair of high impedance earphones having a one microfarad capacitor in series with one lead.

(1) Connect earphone to terminals F and G of APU voltage regulator and listen for a steady, smooth hum or roaring noise indicating a stable voltage regulator. If a succession of popping noises, indicating instability, is heard, replace regulator.

NOTE: If instability is encountered with replacement voltage regulator, check generator leads for security of connections, foreign particals in voltage regulator spring contacts, or remove plug-in assembly and both sides of contacts.

(2) Set APU Generator switch to on.

(3) Add load to generator by energizing equipment normally not energized.

(4) Listening in earphones, check for instability as loads are applied and removed.

NOTE: Instability should not be confused with click heard each time a load is applied, or removed from generator.

F. Disconnect earphone leads when check is completed.

G. Shut down APU, using normal procedures.

H. Inspect for foreign objects then close access panel.

REVERSE CURRENT CUTOUT (ENGINE-DRIVEN DC GENERATORS — DESCRIPTION / OPERATION

1. Description

(See Figure 1)

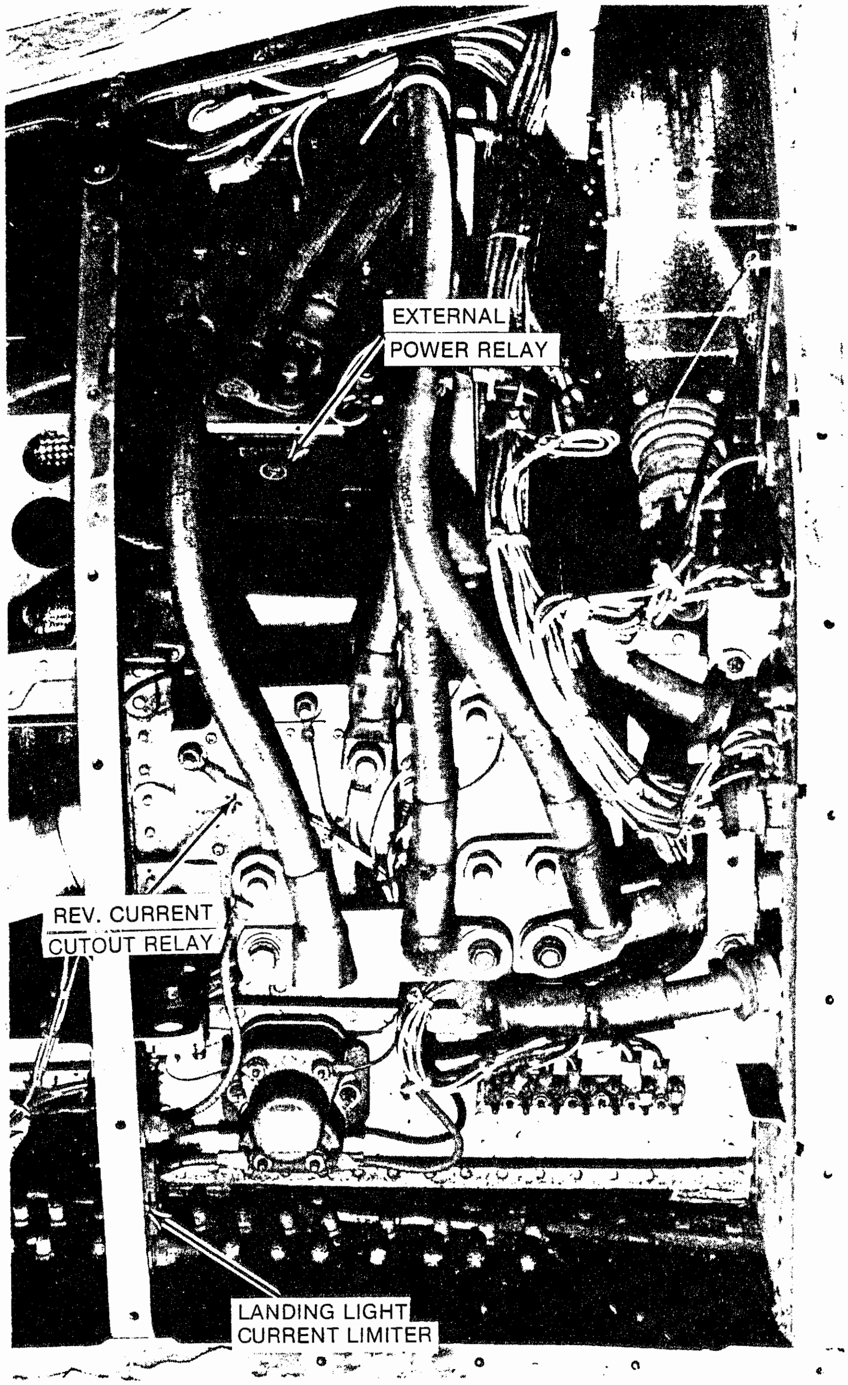
A reverse current cutout for each engine-driven dc generator is located in the nacelle relay box. Each reverse current cutout contains a differential relay. A voltage relay and a main contactor relay. Each cutout automatically prevents current flow from the main dc bus tie cables to the generator and automatically connects the generator to the main dc bus tie cable when generator voltage is of the proper polarity and greater than that of the bus providing the generator control switch is ON. Conversely, the cutout automatically disconnects the generator from the bus when the generator voltage drops to a value lower than that of the main dc bus and a reverse current flows through the cutout.

2. Operation

(See 24-2-1, Figure 3)

The differential relay has three coils controlling its contacts the reverse current coil, the differential coil, and the bias coil. The bias coil and the voltage relay are energized by generator voltage which is routed through the trip relay contacts of the overvoltage relay, the generator control switch, and the T-handle limit switch. When the bias coil and the voltage relay are energized, the differential coil is connected across the open contacts of the main contactor relay. The voltage on this coil is therefore the difference between the generator output and the main dc bus tie cable voltage. When the generator output voltage reaches a predetermined voltage, the differential coil aids the bias coil to close the differential relay contacts. When these contacts are closed, power is routed to the main contactor relay. When the main contactor relay is energized, the generator output is connected to the main dc bus. Power is then available at the "IND" terminal to energize the bus control relay, causing the generator warning light to go out and completing part of the monitor bus relay control circuit.

If generator voltage decreases below that of the main dc bus, a current flows from the bus to the generator through the reverse current coil. This causes the differential relay contacts to open, de-energizing the main contactor relay and disconnecting the generator output from the main dc bus tie cable. The generator warning light comes on, since the bus control relay is de-energized at this time. Setting the generator control switch to OFF, or pulling the T-handle, deenergizes the voltage relay and the bias coil so that the generator will not be connected to the main dc bus tie cable when the generator voltage again increases. If the controls are not moved, the generator output will again be connected to the main dc bus when the generator voltage is sufficient to stop the reverse current flow through the differential coil and the reverse current coil of the differential relay.



Electrical Equipment Location in Left Nacelle Relay Box
Figure 1.

REVERSE CURRENT CUTOUT (ENGINE-DRIVEN DC GENERATORS — MAINTENANCE PRACTICES

1. DC Generator Reverse Current Relay — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to reverse current cutout on nacelle relay box.

(2) Tag, disconnect and insulate electrical leads.

(3) Remove relay mounting hardware and remove relay.

B. Installation

(1) Install relay and secure with mounting hardware.

(2) Connect electrical leads and remove tags.

(3) Close nacelle relay access previously removed.

(4) Run engine and check operation.

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OVERVOLTAGE RELAYS — DESCRIPTION / OPERATION

1. Description

An overvoltage relay is provided for each engine driven dc generator. Both are located under the cabin floor at Fuselage Station 185 under FLR-2. The overvoltage relay contains three relays; an overvoltage sensing relay, a trip relay (with a reset coil and a trip coil) and an anti-cycle relay. A spring.loaded arm holds the trip relay contacts in the position selected. The generator field circuit is routed through one set of contacts of the trip relay. Another set of contacts completes the circuit from the generator to its reverse current cutout via the generator control switch.

This unit also contains a 400 ohm resistor, through which a small current flows from the essential bus to the field windings of the dc generator. This "trickle current" is not of sufficient magnitude to allow generator voltage to build up, but is great enough to ensure that generator polarity does not reverse.

2. Operation

(See 24-2-1, Figure 3)

The generator control circuit is routed through the overvoltage relay, from terminal C across contact number 2 and out terminal D. Note that the overvoltage sensing relay picks up just downstream of the number 2 contact. This sensing relay circuit includes a fixed and a variable resistor in series with the relay coil and terminates through terminal G to ground. The variable resistor controls the trip voltage. It is a bench setting, not to be adjusted in the field. Since this circuit is in parallel with the control circuit, a current directly proportional to the generator voltage will flow through the overvoltage sensing coil. The bias coil shown as part of the overvoltage sensing relay is not utilized in this installation. In normal generator operation, the flow through the sense coil will not build up a magnetic field strong enough to pull in the armature. A generator voltage of 31-33 volts will close this relay.

Suppose that a defective voltage regulator has allowed generator output to reach 36 volts. A proportional current will flow through the sense relay coil, closing the relay. Note that control voltage will now be diverted to the trip coil of the main contactors, across contact number 4 to ground through terminal G. The resulting current will trip the latch, opening all four main contacts. The field circuit across contact number 1 is now broken. Breaking the field current is referred to as tripping the generator. Electrically, in effect, the generator is inoperative. Voltage output at this time is approximately two volts due to the residual field magnetism. In addition, the generator control circuit is broken by contact number 2 opening. This action cuts the power to the reverse current cutout relay, disconnecting the generator from the main dc bus. Meter indication will be no volts-no amps. Since the voltmeter is 15-34 volts the residual voltage will not be indicated on the panel meter. Continuing with the example above, we did not observe the voltmeter during the trip action. The first indication of the trip therefore, was the GEN OFF light coming on in the master warning system. We therefore, do not know what happened to de-energize the generator. Attempt a RESET while observing the voltmeter. The generator control switch for the inoperative system is momentarily depressed in the RESET position. It is spring loaded to OFF. The closed switch in RESET sends voltage from the essential bus through the GEN CONTROL circuit breaker, into the overvoltage relay box through terminal B. In the tripped condition, the four main contacts are open. From terminal B the circuit divides in two ways. First trace the anti-cycle relay circuit which leads through the coil to contact number 3 which is open, therefore, no current flows and the anti-cycle relay remains open. Following the other circuit finds that it leads through the reset coil, through the anti.cycle relay arm to ground through terminal G. The current flow unlatches the tripped main contacts and closes and latches them in the normal position. Several actions take place simultaneously. All of the action so far has taken place in a fraction of a second while depressing the RESET switch and continuing to do so. The anti.cycle relay had a potential impressed upon it, but was waiting for a ground in order to energize and close. Resetting the main contacts, closes contact number 3 providing the ground for the anti-cycle relay thereby shifting the contact. This contact shift breaks the RESET coil circuit by shifting its ground to the anti-cycle relay itself. This relay now has two sources of ground. It will be held in this energized position as long as the RESET switch is depressed. Note that the reset coil itself has lost its ground. This shift of grounds means that regardless of how long we depress the RESET switch, we get only one reset impulse. With the main contacts reset, they again complete the generator field and control circuits. Voltage in the generator will again build up. Recall that during this RESET procedure we are observing the volt-ammeter.

If the high voltage which tripped the generator was a momentary or transient condition, the voltage build up may now be normal, stabilizing at 27.7 $\pm$ 0.2 volts. The RESET switch should then be released and it will spring return to the OFF position. The switch may then be placed in the ON position, putting the generator on the line again. If the high voltage condition still exists, note the build up on the voltmeter. When this voltage reaches the 31-33 volt point, the meter reading will again drop off. This re-tripping will take place even though the generator control is in the RESET position. The anti-cycle relay now prevents an automatic reset. Note that regardless of the time we hold the RESET switch we can only get one reset action. The anti-cycle relay is automatically reset when the GEN switch is returned to the OFF position. If when resetting there is no voltmeter indication, the overvoltage has not been the cause of the no volts-no amperes indication. The malfunction is probably an open field circuit or inoperative generator. In addition to the automatic trip provided by the overvoltage relay, manual tripping is also possible. The manual generator trip is accomplished at crew discretion by depressing the generator TRIP switch, for the desired generator to the TRIP position. Both the left and right generator TRIP switches are actuated when the emergency gang bar is pulled. The TRIP switch routes essential dc bus power to terminal 1 of the overvoltage relay. From there, the power goes directly to the trip coil, tripping the main contacts. Reset after a manual trip is the same as described above. The trip switches receive power from the essential bus through the GEN TRIP circuit breakers.

A generator polarizing circuit is also built into the overvoltage relay. Power for this circuit comes directly from the essential bus through the GEN CONTROL circuit breakers. Feed into the box is through terminal F which is tapped into the field circuit downstream of the number 1 contact and out terminal E after passing through a 400 ohm voltage dropping resistor. With this circuit, a small current will bleed through the generator field coils, setting up a magnetic field in the proper direction. Whenever the essential bus is energized and the generator armature is not rotating, this circuit, which is always on, does away with the necessity of polarizing or flashing the field due to reverse polarity or loss of residual magnetism. In effect, flashing the field occurs each time the battery is switched on. The current downstream of the dropping resistor is so low that its effect on the field circuit during normal operation is negligible.

EXTERNAL POWER RELAY — DESCRIPTION / OPERATION

1. **Description**

The external power relay is located in the left nacelle relay box (See Figure 1). The external power source when plugged into the external receptacle, is connected to the main dc bus tie cable by the external power relay. This unit contains a main contactor relay, a voltage relay, and a differential relay. This relay is similar to the generator reverse current cutout.

On Aircraft 1 - 200, 322 and 323 having ASC 193A, an overvoltage relay panel consisting of a voltage sensing relay and a resettable latching relay has been added under the floor at Fuselage Station 169. This provides protection against excessive voltages and reverse polarity from ground power sources. An external power distribution relay has also been added at Fuselage Station 133 relay panel. (See Figure 2). This relay automatically closes the right and left battery normal relays and both normal bus tie relays with the application of External Power. On Aircraft 1 - 200, 322 and 323 having ASC 193A an engine start interlock relay has been added at Fuselage Station 133 relay box. (See Figure 2) This relay disables the overvoltage protection when starting or cranking engines to prevent disconnection of ground power due to tripping from ground power transients.

2. **Operation**

(See Figure 1)

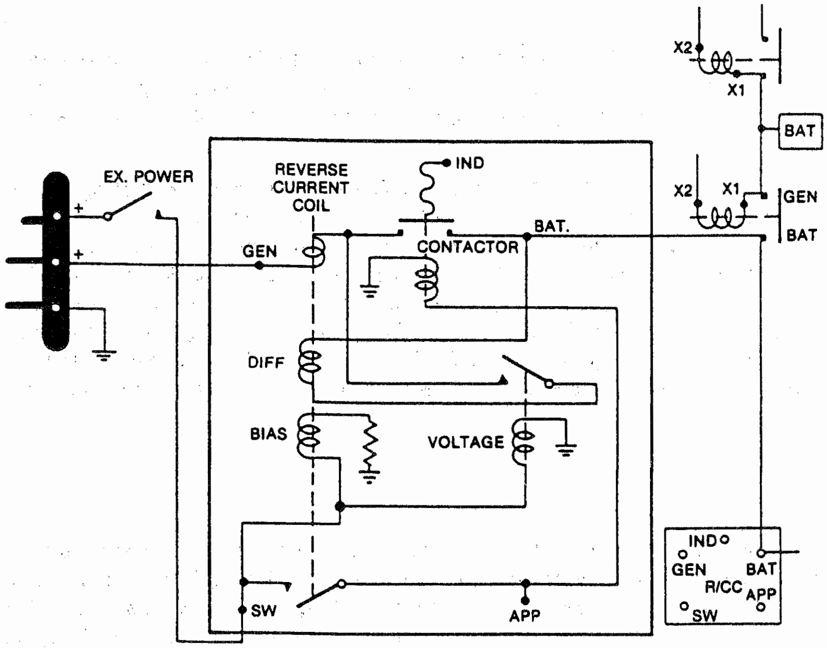
When external power is connected to the aircraft, potential is then applied to the generator side of the large contactor within the relay. Recommended external power voltage is 28 to 29V dc regulated. (On Aircraft 1 - 200, 322 and 323 having ASC 193A, the recommended external voltage should not exceed 28.0V dc regulated.) The short pin, as part of the control circuit, mates with a receptacle in the plug, which is connected to the output source of external power. Whatever voltage Is produced by the external power source is therefore impressed on the control circuit. When the EXT PWR switch is placed in the ON position, it allows the voltage to enter the external power relay through the SW terminal. Voltage from the SW terminal then forces a current to flow through the voltage and bias coils. This flow produces a magnetic field around these coils which is proportional to the voltage and whose polarity is related to the polarity of the external power source. The bias coil forms a part of the differential relay. Its magnetic field is used as a reference of output voltage and direction, both the differential and voltage relays are polarized. A polarized relay is an electromagnetic switch with a permanent magnet for an armature. The opening and closing of this relay, therefore, is dependant upon direction of current flow in the coil. When external power source voltage reaches approximately 22 volts in proper direction, the polarized voltage relay closes. Closing this relay completes a circuit connection across the open main contacts and through the fine wire coil of the polarized differential relay.

Assume 28 volts is on the generator side of contactor and 24 volts on the battery side. Current will then flow through this differential circuit proportional to the difference of the two voltages. The magnetic field developed around the differential coil will also be proportional to this difference and polarity will be related to the direction of flow. This coil is so wound that if the current flows from generator side to battery side, the differential coil magnetic field will be additive to that of the bias coil. If the voltage on the generator side exceeds that of the battery side by 0.35-0.65 volts or more, which in this example it does, the combined magnetic field being in the proper direction will close the differential relay. Control circuit voltage will now be impressed on the main contactor coil, closing it. Current as required by the loads will now flow from the external power source to the normal bus feeder system.

Note that a loop is formed in the line between the GEN terminal and the contactor. This loop is called the reverse current coil. In a flow from GEN to BATT, the field buildup assists the bias coil in keeping the relay closed. If the voltage at the BATT terminal exceeds that at the GEN terminal, current flows in the reverse direction. This builds up a field of opposite polarity which resists the bias field. This results in the opening of the differential relay thereby breaking the circuit to the main contactor, which now opens.

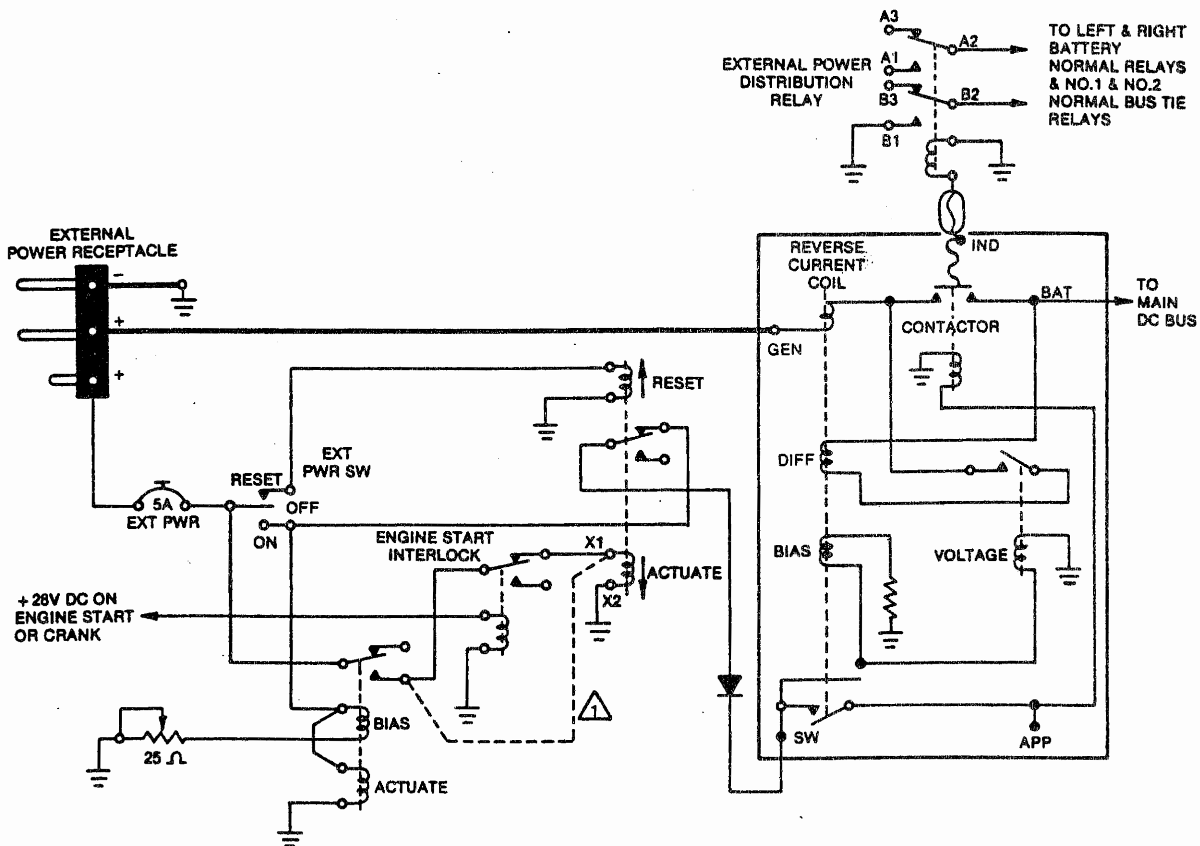
In another condition, assume that a battery on the bus is giving 24 volts at the battery terminal of the external power relay, and external power source voltage is 23.5 volts. In this condition, current flow through the differential coil is in such a direction that the polarity of the field counteracts that of the bias coil. In this condition, the differential relay cannot close, therefore the external power source is not connected to the aircraft bus.

In another condition, assume that the external power source plug is wired opposite the polarity of the aircraft receptacle. Control circuit power to both the voltage and differential relays flows in the wrong direction and the polarized armature does not close. Therefore, the external power is not connected to the aircraft bus.



NOTE: SCHEMATIC TYPICAL OF ALL REVERSE CURRENT RELAYS

External Power Circuit — Simplified Schematic
Figure 1.



1 USED ON AIRCRAFT WITH ASC 193A INCORPORATED

External Power Circuit — Simplified Schematic
Aircraft Having ASC 193A
Figure 2.

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EXTERNAL POWER RELAY — MAINTENANCE PRACTICES

1. Overvoltage — Adjustment

(Aircraft 1 - 200, 322 and 323 having ASC 193A)

A. Close following circuit breakers (pilots circuit breaker panel):

- NUTCRACKER CONT - (Aircraft 1 - 93 and 114 having ASC 108A and Aircraft 94 - 200, 322 and 323)
- LG HORN - (Aircraft 1 - 93 and 114 not having ASC 108A)
- GND START
- EXT PWR - (Copilots circuit breaker panel)

B. Ensure EXT PWR switch and BATT switch (center overhead panel) are OFF.

C. Connect an adjustable dc power source to external power receptacle. Using a precision voltmeter set voltage at exactly 29 volts.

D. Set EXT PWR switch ON. External dc power should be applied to aircraft circuits and automatically disconnect within 30 seconds.

E. Ensure ENG SELECTOR switch (lower overhead panel) is OFF.

WARNING: IN ORDER TO PREVENT PERSONNEL INJURY BY INADVERTENT CRANKING OF ENGINES DURING SUBSEQUENT TESTS THE ENGINE SELECTOR SWITCH MUST REMAIN OFF AND PROPELLERS SHOULD BE CLEARED. TO STOP INADVERTENT CRANKING IMMEDIATELY PLACE START SWITCH TO OFF POSITION.

F. Hold START SELECTOR switch (lower overhead panel) in CRANK.

G. Hold EXT PWR switch momentarily in RESET and return it to ON, external dc power should appear on aircraft circuits and remain on.

H. Set EXT PWR switch and START SELECTOR switches OFF.

I. Using a precision voltmeter, set voltage of external dc power source at exactly 28 volts.

J. Hold EXT PWR switch momentarily in RESET, then ON. External dc power should reappear on aircraft circuits and remain on.

K. Set EXT PWR switch OFF.

L. Failure to obtain an automatic disconnect within 30 seconds as per Step D. above or a failure to remain on as per Step J. above may be corrected as follows:

(1) Set EXT PWR switch and BATT switch (center overhead panel) OFF. Disconnect external power from aircraft.

(2) Remove FLR-5 and locate overvoltage relay panel assembly at Fuselage Station 164.

WARNING: POWER RESISTOR ON OVERVOLTAGE RELAY PANEL ASSEMBLY MAY BECOME HOT ENOUGH TO CAUSE FIRST OR SECOND DEGREE BURNS AFTER A FEW MINUTES OF OPERATION. AVOID DIRECT CONTACT WITH POWER RESISTOR.

(3) On overvoltage relay panel assembly, locate the control resistor and adjust as follows:

NOTE: A small movement of center contact may be all that is necessary.

(a) If aircraft external power does not disconnect as described in Step D. above loosen screw on center contact and decrease the total circuit resistance. This will decrease voltage at which disconnect will occur.

(b) If aircraft external power does not remain on as per Step J. above loosen screw on center contact and increase total circuit resistance. This will increase the voltage at which disconnect will occur.

(4) If proper results are not obtained, repeat procedures in L. above.

2. External Power Relay — Removal / Installation

A. Removal

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

(1) Remove cover of left nacelle relay box.

(2) Remove wires from relay which is located in center/aft side between alternator current balance relay, and generator reverse current cutout relay. Identify and insulate removed wires.

(3) Remove mounting hardware from relay.

(4) Remove relay from chassis.

B. Installation

(1) Install relay on chassis.

(2) Install mounting hardware on relay.

(3) Connect identified wires to relay and check connections.

(4) Connect external electrical power connector.

(5) Set external power switch, located in cockpit, to ON.

(6) Check that relay operation is normal.

(7) Set external power switch off and remove external electrical power connector.

(8) Inspect area for presents of foreign objects, security of all attachments.

(9) Install cover on left nacelle relay box.

BATTERIES — DESCRIPTION / OPERATION

1. Description

A 24-volt, 34-amp, nickel-cadmium battery is installed in the battery compartment of each nacelle. These compartments are located in the lower, aft, outboard section of each nacelle. Each battery is equipped with a quick disconnect type connector. (See Figure 1) The battery consists of 19 replaceable, plastic container cells, each rated at 1.30 volts when fully charged. Each cell cap contains a built-in venting system which prevents spillage when the battery is operated in any position, but which permits the escape of any small amount of gas which may form when the battery is charging. The electrolyte used is potassium hydroxide (KOH), which is an alkali. The cells are vented to the battery case. An overboard vent line is located aft of the battery compartment and is connected to the battery case. The batteries will operate in temperatures from – 65°F to 165°F.

2. Operation

(See Figure 2)

Each battery is connected to the main dc bus by means of a normal battery contactor. The main dc bus in turn is connected to the essential dc bus by means of two normal bus tie relays. In addition, in an emergency, the batteries may be connected to the essential dc bus only, by the emergency battery relays. The battery relays and contactors are mounted in the battery compartment aft of the batteries. The BATT switch in the center overhead panel is provided for control of the batteries. Both batteries are selected simultaneously in either the NORM or EMER position. The NORM position connects the batteries via the normal battery contactors to the main dc bus which in turn ties to the essential dc bus via the two norm bus tie relays. The EMER position connects the batteries via the emergency battery relays to the essential dc bus only. When the battery switch is in the EMER position, the essential dc bus is powered by the batteries and is completely isolated from the main dc bus. Individual battery selection can be achieved by pulling the circuit breaker of the particular battery relay not desired; (i.e., if the right battery is desired on the main dc bus, put the battery switch in the NORM position and pull the left normal battery relay circuit breaker). The BATT CONT circuit breakers are located on the right circuit breaker panel.

On Aircraft 1 - 200, 322 and 323 having ASC 192, individual battery selection can be achieved by pushing the BATT DISC switch for the particular battery not desired: (i.e., if the right battery is desired on the main dc bus, put the battery switch in the NORM position and push the L. BATT DISC switch). An amber light in the switch on the overhead panel will indicate the battery has been disconnected.

3. Electrochemical Action

In the charged state, the positive plates are charged with nickel oxide, and the negative plates are charged with cadmium. During operation, the potassium hydroxide (KOH serves only as an ion conductor, taking no part in the chemical reaction. Specific gravity does not change in relation to the state of charge of the battery. The charge-discharge reactions are chemically reversible with no end products resulting in either direction. This capability of operating without leftovers, plus the plate construction which inhibits shedding of material, are the factors which permit a longer battery life.

The battery can deliver 1000 amps or more if required, without being damaged. The battery does not produce gas during discharge. On charging, gassing is at a minimum and catalytic action by nickel oxide present in the cell returns most of the gas formed, to water. Overcharging will cause gassing. Battery voltage does not vary directly with load or charge state, instead, a plateau characteristic is exhibited. Curves for three different discharge rates are shown in Figure 2 See Figure 3 for information on ground operating time when using the batteries only.

4. **Temperature Effect**

Nickel cadmium batteries will not normally freeze, and can be used at temperatures as low as −65°F. However, the battery will require more voltage to accept a given amount of charge when colder than normal.

The battery will appear to have limited capacity when cold. This capacity reduction may be as much as 40% at −40°F. As the battery becomes warm, its responses return to normal.

High temperatures also limits the amount of capacity available. This is because the use of finish charge (to fully top up the cells) becomes impossible without causing permanent damage. Any temperature above 109°F will cause capacity loss. Approximately 60% a of capacity remains at 120°F. Above 160°F the battery should not be used.

5. **Thermal Runaway**

In a battery being charged under normal conditions, (moderate temperature, charge rate and voltage) the cell internal voltage rises gradually as the electrochemical action takes place. This gradually bucks out the charge voltage, causing the rate charge to gradually decrease until it becomes a mere trickle, just sufficient to equal the continuous cell losses.

When the battery is charged at higher than normal temperatures, the battery efficiency decreases and the cell voltage decreases. The internal cell resistance also decreases as temperature rises. These factors combine to admit more (than normal) charge current, which in turn increases chemical activity producing additional heat. This situation may escalate into a runaway condition.

A battery in good condition will normally reach a point of stability where the trickle current heat will just balance radiated and conducted heat losses. A battery with damage (or deteriorated) cellophane cell separators or a battery wherein other defects exist may not reach stability. Its temperature may seek higher and higher levels until a runaway is in progress.

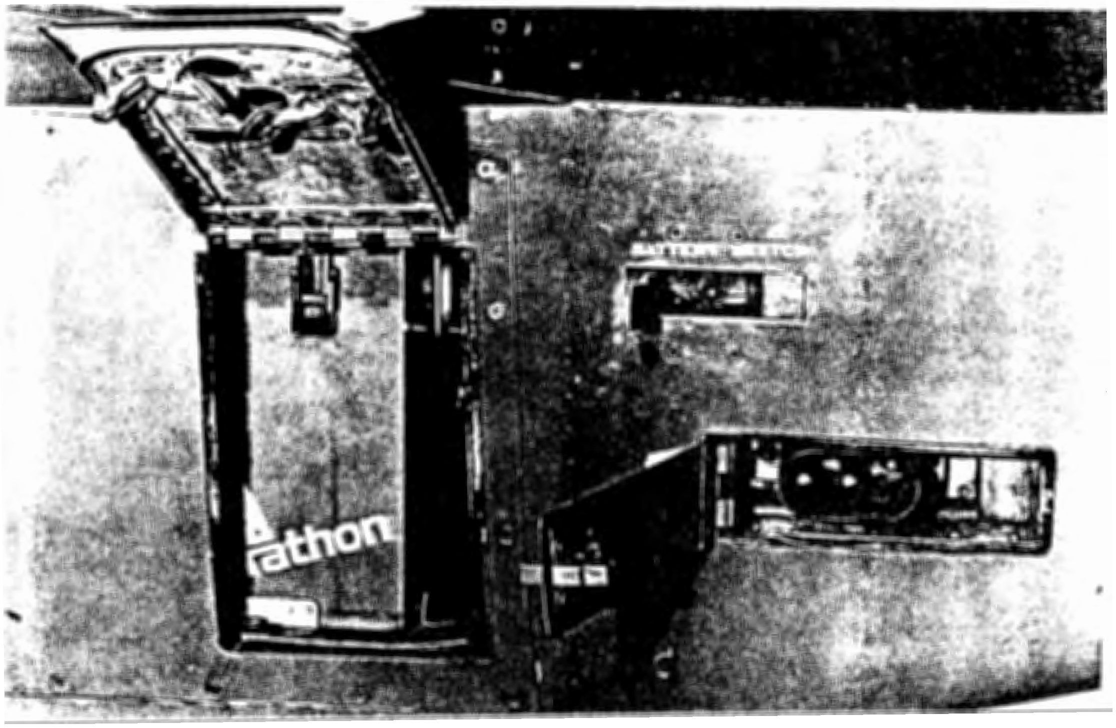
Batteries with internal defects have been known to begin runaway at temperatures of only 120°F. Batteries in good condition have been charged at temperatures as high as 160°F without inducing runaway. It should be noted that operation at elevated temperatures will eventually damage the cellophane separators, thus leading to failure.

A battery can be driven into runaway in one cycle by excessive voltage at high temperature, or it may fail as a result of years in service at not quite critical levels.

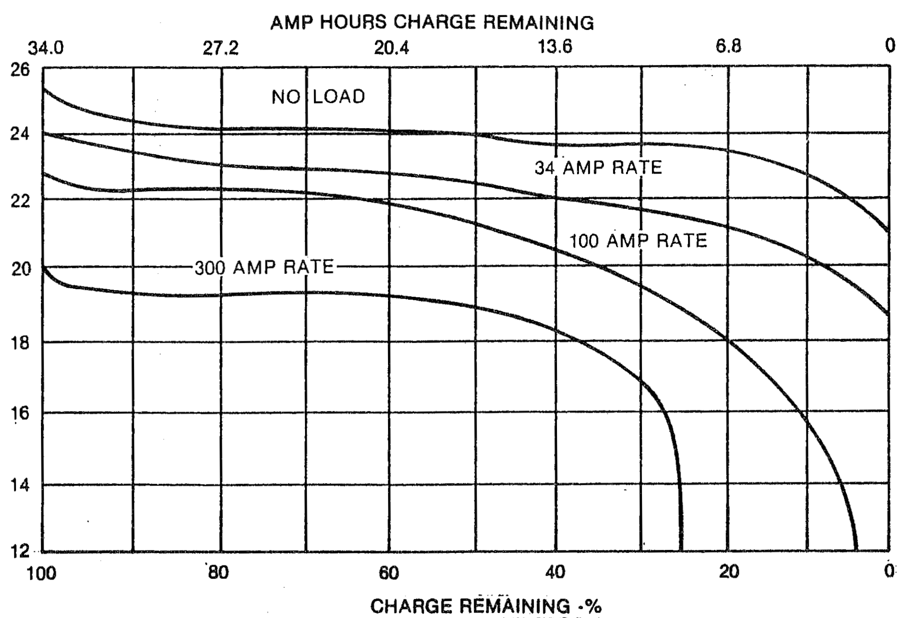
6. **Air Circulator Fan**

(See Figure 4)

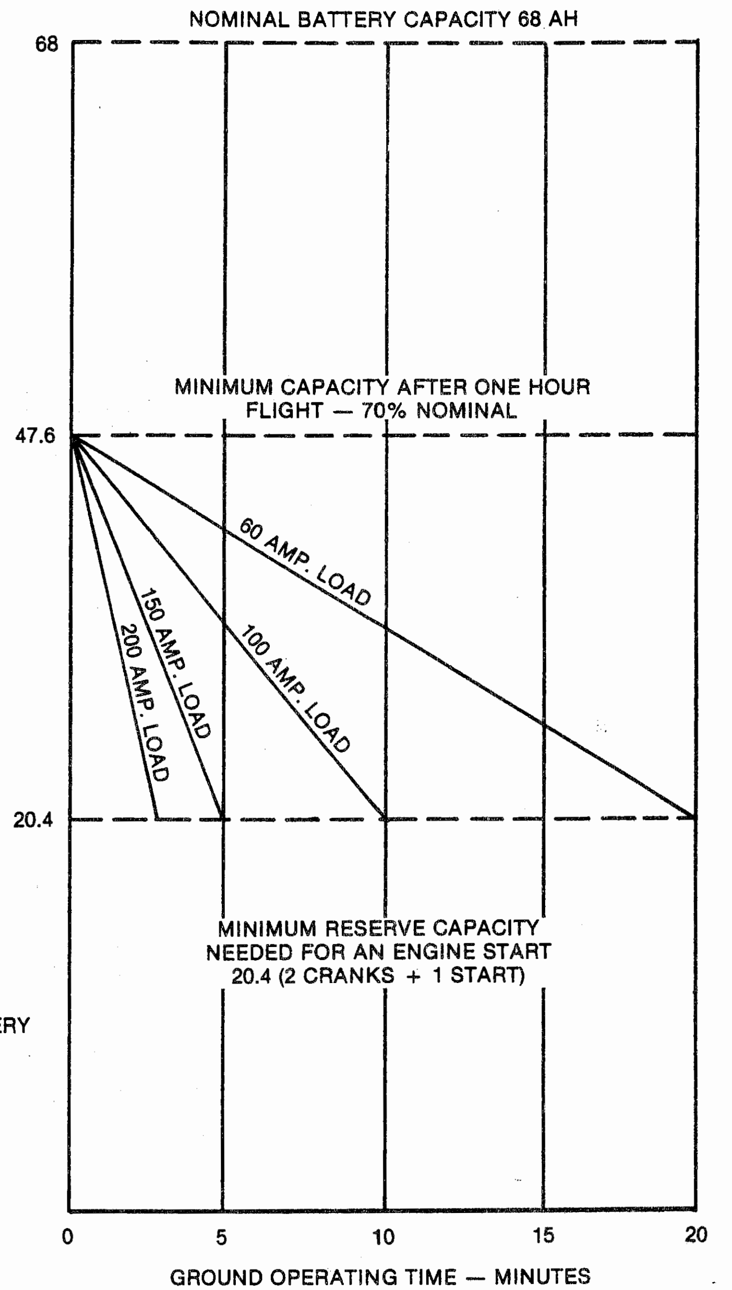
On Aircraft 1 - 200, 322 and 323 having ASC 204, circulation of air through the aircraft's left and right battery compartments is provided by means of air circulator fans. The fans are actuated by means of thermal switches which close when the battery compartment temperature reaches approximately 90° and open when the compartment temperature reaches approximately 75°.



Battery Equipment Location
Figure 1.

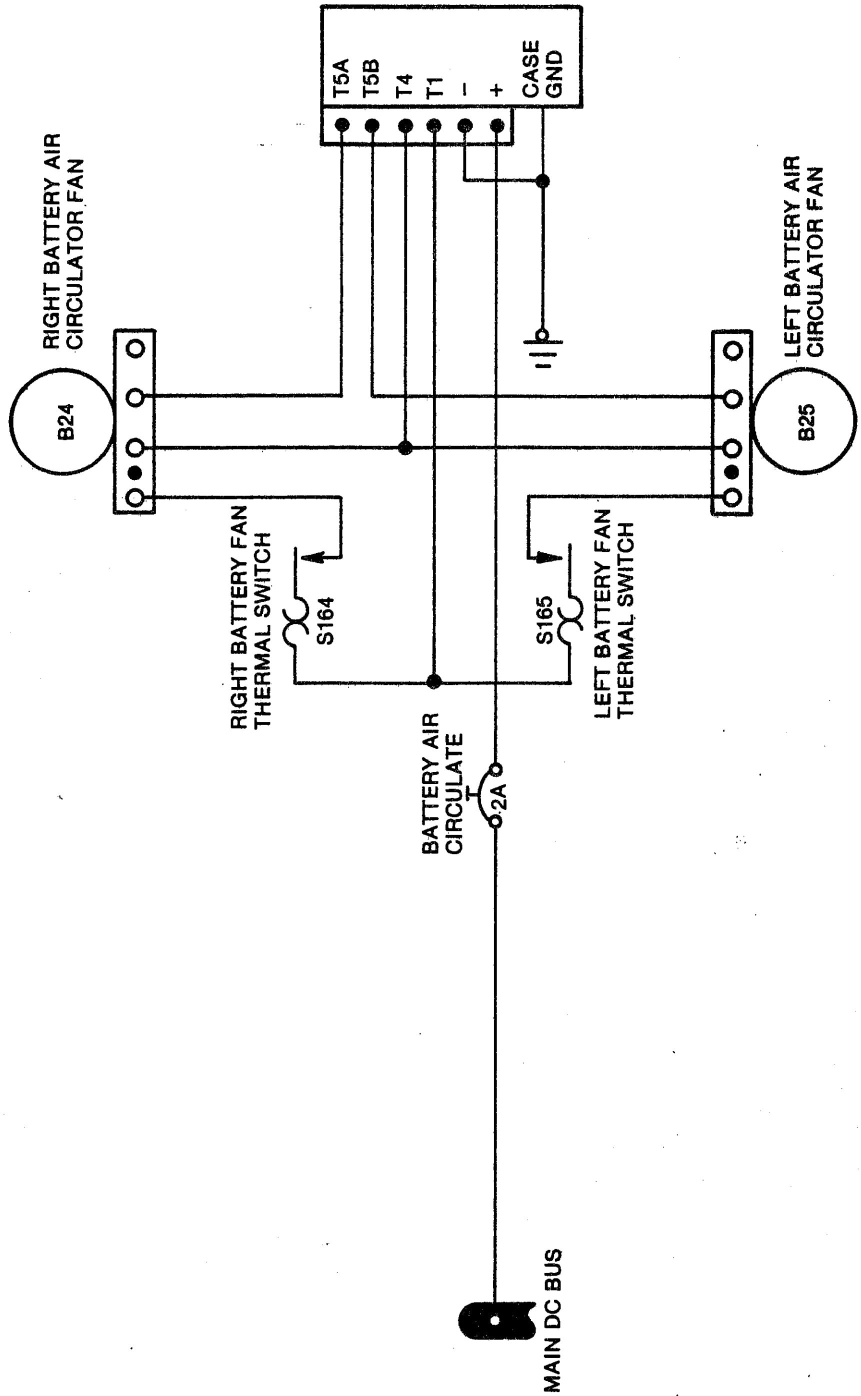


Battery Discharge Curves
Figure 2.



NOTE: SINTERED PLATE NICKEL-CADMIUM BATTERY INSTALLATION IN THE "GULFSTREAM I"

Load Discharge Versus Ground Operating Time on Batteries Only
Figure 3.



Battery Air Circulator System
Figure 4.

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BATTERIES — MAINTENANCE PRACTICES

1. **Safety Practices**

WARNING: A NI-CAD BATTERY IS AN INNOCENT APPEARING CREATION WHICH PACKS HIDDEN PERIL IN SEVERAL FORMS. IT BEHOOVES ANY PERSON USING OR WORKING WITH SUCH A BATTERY TO TAKE CARE AND OBSERVE THE FOLLOWING SAFETY PRACTICES.

A. When a battery exhibits symptoms of distress, disconnect it from charging source without delay.

B. Beware of metal objects in or near a battery. Remove all hand jewelry while working around batteries. Short circuits and sparks can cause serious damage.

C. Wear protective clothing when working around batteries. A face mask (or shield) gloves and apron are advisable.

D. Make due allowances for the weight of the battery.

E. Do not smoke in a battery service area.

F. Keep it clean.

G. Existence of oxygen and hydrogen gasses during a charging cycle makes a battery a potential fire hazard. Operations should be conducted in a well ventilated area, away from sources of ignition.

H. Nickel cadmium batteries use an electrolyte that is strongly alkaline in composition. This will react with violence if contacted by acid materials. Therefore, it is most important to ensure that tools, areas and atmospheres used, not be contaminated by other (lead-acid) batteries.

2. **Battery — Removal / Installation**

A. Removal

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING THIS COMPONENT.

(1) Remove access panels at outboard side of either nacelle to gain access to battery. Retain hardware.

NOTE: On aircraft having ASC 204, disconnect fan connector.

(2) Disconnect battery connector.

(3) Remove vent hose at top outboard side of battery.

(4) Remove battery outboard tiedown rod wing nut. Also remove battery bracket bolts and tiedown rod and bracket.

(5) In wheel well, remove battery inboard tiedown rod.

WARNING: EACH BATTERY WEIGHS APPROXIMATELY 80 POUNDS.

(6) Support battery and slide outboard. Remove battery.

B. Installation

(1) Preform Battery Compartment Visual — Inspection, this Section.

(2) Before installation, ensure unused battery case vent port is capped and clamped with appropriate plastic cap and metal clamp.

(3) Install battery in compartment with connector facing aft.

(4) Install battery outboard tiedown rod and bracket. Do not connect to case at this time.

(5) Slide bracket inboard until firm against battery case. Tighten mounting bolts.

(6) Install inboard tiedown rod in wheel well to battery case cover and tighten nut slightly.

(7) Install outboard tiedown rod to battery case cover and tighten wing nut.

CAUTION: ON AIRCRAFT HAVING ASC 192 THE BATTERY CASES ARE INSULATED FROM THE AIRCRAFT STRUCTURE. INSTALL WING NUT SAFETY WIRE ONLY BETWEEN WING NUT AND INBOARD ROD AND WING NUT AND CLIP ON THE OUTBOARD ROD.

(8) Fully tighten inboard tiedown rod wing nut and safety.

(9) Safety outboard wing nut and battery bracket mounting bolts.

(10) Connect vent hose to battery case vent.

CAUTION: PRIOR TO PERFORMING STEP (10) BELOW, ENSURE THAT THE BATTERY HAS HAD SUFFICIENT TIME SINCE BEING SERVICED, OR CHARGED SO THAT ANY TOXIC OR FLAMMABLE FUMES HAVE BEEN VENTED FROM BATTERY CASE. BATTERY CABLES **MUST NOT** BE CONNECTED FOR THIS TEST.

NOTE: Steps (10) through (17) below apply to aircraft having ASC 192.

(11) Perform a megger check, using a 500V megohm meter, by applying it between ground (aircraft structure) and battery case. The resistance shall not be less than 3.0 megohms.

(12) Connect electrical plug.

(13) In cockpit, depress L or R BATT DISC switch, as applicable.

(14) Place BATT switch to NORM. (Center overhead panel).

(15) Observe essential bus voltmeter for indication of 24 volts (approximately).

(16) Energize one fuel boost pump and E–inverter (ESS AC SEL SW to NORM).

(17) Observe essential bus voltmeter. It must read at least 22 volts.

(18) To check opposite battery, connect battery disconnected in (Step (12) above, and depress opposite disconnected switch. Voltage should read minimum of 22 volts.

NOTE: Step (18) thru (24) below, apply to aircraft not having ASC 192.

(19) Connect electrical plug.

(20) In cockpit, pull L or R NORM BATT, circuit breaker (as applicable).

(21) Place BATT switch to NORM.

(22) Observe essential bus voltmeter for indication of 24 volts (approximately).

(23) Energize one fuel boost pump and E inverter (ESS AC SEL SW to NORM).

(24) Observe essential bus voltmeter. It must read at least 22 volts.

(25) do check opposite battery, push in circuit breaker that was pulled in (Step (19) above, and pull opposite breaker. Voltage should read minimum of 22 volts.

NOTE: The following steps apply to all aircraft.

(26) Turn off boost pump and E-inverter.

(27) Place BATT switch off (center overhead panel).

(28) Disconnect the batteries if no longer required.

NOTE: On aircraft having ASC 204, connect fan connector.

(29) Inspect area for presents of foreign objects, security of all attachments.

(30) Install the battery compartment access cover(s).

3. NI-CAD Battery — Capacity / Deep Cycle

NOTE: The following procedures will be formatted in "STEPS" for ease of maintenance. **Depending on condition of battery**, some or all "STEPS" may be required. Some "STEPS" may have to be repeated i.e. Leakage Test after Deep Cycle for Battery Cleaning.

Note that battery may pass Visual Inspection thru Capacity Check but may still require Deep Cycle and Battery Charging if battery needed Cleaning. Each inspection will differ depending on age, condition and maintenance of your battery.

STEP 1 — VISUAL — INSPECTION / INITIAL CLEANING

A. Open battery access door.

B. Remove batteries from aircraft. Refer to this section.

C. Inspect exterior of case and cover for bulges, evidence of overheating, cleanliness of vent ports and exterior epoxy finish for condition (touch up if required using identical finish).

D. Inspect for evidence of arcing from battery to aircraft structure particularly on lower case area and ends of tie down strap on covers. If arcing signs are evident, determine cause and take appropriate corrective action. Clean as required.

E. Remove covers and inspect tops of cells for signs of spewage. Clean as required.

F. Inspect bus bar connections for evidence of overheating.

G. Check cell polarity markings to ensure proper installation. No. 1 cell (+) should connect to main plus terminal. No. 19 or 20 cell (−) should connect to main minus terminal. All other connections (cell to cell) should be(+) to (−).

H. Inspect rubber lining of cover for damage and condition, and interior of battery case for foreign objects.

I. Inspect main terminal studs for cleanliness, smoothness and evidence of burning or arcing.

J. Check cover latches for operation and ensure they secure cover solidly.

K. If temperature probe (TRAMM P600) is installed, inspect security of attachment. Inspect lead for evidence of chafing or other damage. Inspect connector for security / damage. Perform voltage leak test and resistance check as follows:

NOTE: Refer to probe manufacturers maintenance manual if other type probes are installed.

(1) Connect test meter as shown in See Figure 201. Ensure test leads do not touch metal case as this will give false reading.

(2) Check for voltage between probe connectors A, B, C and negative terminal on battery. If any voltage is present on any terminal, probe is defective and should be replaced.

(3) Check resistance between pins A and B. Reading should be approximately 49.9k.

(4) Check resistance between B and C pins. This reading is affected by temperature, see Resistance Versus Temperature Chart on Figure 201.

L. Perform the following check on thermal sensor if installed:

NOTE: Conduct check Steps (1) and (2) below with battery temperature at 60° to 80°F for optimum results.

(1) Connect an ohmmeter to connector pins 8 and 9 of thermal sensor assembly to check overtemperature thermoswitch. Verify ohmmeter shows closed contacts.

(2) Connect an ohmmeter to connector pins 11 and 12 to check charge control thermistor. Verify ohmmeter reads 2360 to 3050 ohms.

M. Remove any contamination from top of cell such as dirt, oil, grease, potassium hydroxide powder, or corrosion.

N. Inspect battery for following conditions, if any exist, battery should be completely disassembled, See STEP 7.

(1) Evidence of excessive liquid spillage.

(2) Marked corrosion of bus bars or hardware.

(3) Dirty or damaged outer case.

(4) Any knowledge of defective cells.

(5) Encrustation of parts with white (or grey) powder.

NOTE: If none of these conditions exists, and battery is clean, it may be left assembled, however a clean appearance does not ensure batteries will pass subsequent tests. Disassembly at this time may save time later.

O. Check torque on all fittings and connections using torques in battery Maintenance Manual.

STEP 2 — TOPPING CHARGE

CAUTION: NI-CAD BATTERIES MUST NOT BE SERVICED WITH EQUIPMENT WHICH HAS BEEN USED TO SERVICE LEAD ACID TYPE BATTERIES.

ON A PARTLY DRY CELL, VOLTAGE MAY RISE VERY RAPIDLY WHEN CHARGE IS APPLIED. CONTINUED DRY CHARGE WILL CAUSE DAMAGE. CELL VOLTAGES SHOULD BE WATCHED CLOSELY DURING FIRST 10 MINUTES OF CHARGING, AND DISTILLED / DEMINERALIZED WATER ADDED AS NECESSARY IF THIS CONDITION OCCURS. ADD ONLY ENOUGH WATER TO BARELY COVER PLATES.

A. Charge battery as per manufacturers instructions until all cells read between 1.55 and 1.75 volts. Do not exceed the manufacturer's specified max charge rate.

NOTE: This procedure brings all cells to a charged state to enable water level check. A dead battery requires at least 5 hours for water to appear, partly charged batteries reach desired point much sooner.

STEP 3 — ELECTROLYTE CHECK

NOTE: Ensure cells are charged above plateau level (region of final charge) before water level adjustments are made. 1.60 volts per cell is midway in this region and is used as a reference. With reasonable cell balance, when first cell is at 1.60 volts, lowest cell should be at least to top of plateau level.

A. After topping charge is completed, check water level in cells, immediately after charging. See battery Maintenance Manual for proper level.

B. If required, add pure distilled (or demineralized) water up to proper level.

NOTE: Cells that indicate signs of heavy spewage should have specific gravity checked with a hydrometer. Values from 1.240 to 1.320 are acceptable. Addition of potassium hydroxide in field is not advisable. Low specific gravity reading is grounds for removal.

STEP 4 — LEAKAGE TEST

NOTE: Surface contamination anywhere on (or in) battery can provide leak age paths between exposed terminals. All terminals come out the top of cells, making this the prime area of consideration. The following test can be made using a standard electrical shop ammeter instrument.

CAUTION: NEVER CONNECT ANY AMMETER ACROSS THE TWO BATTERY TERMINALS. THE METERS LOW INTERNAL RESISTANCE WILL PERMIT LARGE CURRENTS TO FLOW WITH RESULTING DAMAGE.

A. Adjust meter and input leads for a measurement of 500 milli amps.

B. Connect negative lead from the meter to battery container.

C. Touch positive lead from meter to positive terminal of battery and then to positive terminal of each cell.

D. If, while performing above, there is any deflection of needle from zero, insulation should be considered defective and battery should be thoroughly cleaned, after performing STEPS 5 and 6.

NOTE: If, after cleaning and ensuring that everything is dry, a leakage current is still indicated by a deflection of needle, it should be traced to defective cell or cells which should be replaced.

STEP 5 — CAPACITY CHECK

NOTE: The purpose of this discharge is to determine the electrical characteristics of the battery when it is removed from the aircraft, either for routine maintenance or to investigate a malfunction.

A. Discharge battery as per manufacturers instructions until the battery reaches 20.0 volts. It is important that the discharge current be continually maintained at the selected value, and that the time of discharge be measured accurately.

B. Monitor individual cell voltages periodically during the discharge, and record the time at which the first cell reaches 1.0 volt. It is not a cause for concern if a cell goes to zero volts or reverse polarity during the battery discharge. Simply short out such a cell's terminals for the remainder of the discharge. Stop discharge when the battery voltage reaches 20.0 volts.

C. If the time until the first cell reaches 1.0 volt equals or exceeds the values in manufacturer's manual, it indicates that this cell has at least 85% of its rated capacity: This indicates the battery has been receiving a good charge in the aircraft and that the cells are within an acceptable range of capacity balance. Therefore, after allowing the battery to cool to room temperature (at least 12 hours of rest), it need only be charged STEP 8, and electrolyte leveled STEP 3. It can then be placed back in service. If battery failed Capacity Check, perform Deep Cycle STEP 6, Battery Charging STEP 8, then repeat Capacity Check STEP 5.

STEP 6 — DEEP CYCLE

NOTE: In order to bring all cells to an equal state, they must be discharged to zero and charged evenly until final charge states are approximately equal.

A. Discharge battery as per manufacturers instructions.

CAUTION: BATTERIES SHOULD NEVER BE BROUGHT TO A VERY LOW STATE OF CHARGE UNLESS CELL MONITORING AND SHORTING PROCEDURE IS FOLLOWED.

B. Monitor individual cell voltages, as each cell falls to 0.20 volts, place shorting strap and short circuit it. This procedure is required to prevent cell reversal.

C. Continue discharge until all cells are short circuited.

NOTE: For long term storage and shipping, main terminal shorting bars should be installed.

D. Perform discharge and short circuit cell procedure, when all cells are shorted, main terminal shorting bar is applied.

E. When main terminal shorting bar is in place, cell shorting straps may be removed.

F. Tag battery identifying it as being discharged and shorted.

STEP 7 — BATTERY CLEANING (DISASSEMBLED)

NOTE: Perform the following cleaning procedure when disassembly is required. Ensure 3 hour wait after battery deep cycle before performing maintenance.

CAUTION: COMPLETELY DISCHARGE BATTERY PRIOR TO DISASSEMBLY.

CELL SHORTING JUMPERS MUST BE REMOVED DURING THIS PERIOD. IF REASSEMBLY IS TO BE WITHIN A SHORT PERIOD, LEAVING CELLS OPEN-CIRCUITED IS PERMISSIBLE. IF CELLS ARE TO REMAIN OUT OF THE BATTERY FOR MORE THAN TWO HOURS, REINSTALL SHORTING JUMPERS.

WARNING: OPEN CIRCUITED CELLS WILL RAPIDLY REGAIN VOLTAGE, EVEN THOUGH FULLY DISCHARGED. USE CARE IN HANDLING OPEN CIRCUITED CELLS TO AVOID INJURY.

A. Disassemble battery, retain hardware and tag each cell as to location within battery. Ensure cells are identified as to which battery they were removed from.

B. Clean battery with distilled (or demineralized) water and bristle brush and air dry.

C. On rubber lined case, wash gently and air dry.

D. Inspect rubber for deterioration and pulling away from can, replace or recement as required. Use Neoprene rubber of 40-60 durometer hardness. Secure to can with rubber-to-metal cement.

E. Clean cover using same procedures as on can. Repair or replace Neoprene as required.

F. Ensure latches operate properly and secure cover firmly.

G. Any conformal coating around top of cells should be removed for proper cleaning of cell.

WARNING: DO NOT USE METALLIC BRUSHES OR TOOLS, SHORT CIRCUITS MAY RESULT.

CAUTION: ENSURE VENT CAP ASSEMBLY IS IN PLACE AND VENT CLOSED TO PREVENT CLEANING PRODUCTS FROM ENTERING INSIDE CELL.

H. Clean cells with bristle brush and adequate supply of distilled (or demineralized) water.

I. Inspect cell outer case. No holes, thin spots or leakage when squeezed, should be evident.

CAUTION: IT IS IMPORTANT THAT ALL CELLS USED IN ONE BATTERY BE IDENTICAL.

J. Remove vent cap assembly, keeping cell upright. If vent cap is not immediately replaced, cover opening.

NOTE: Some vent assemblies have a rubber collar in a groove and some have spring loaded plungers incorporated to release pressure build up.

K. Inspect vent assembly for condition and operation, replace rubber parts and metal springs as required.

L. A pressure check of vent assemblies is recommended. Using a pressure gage and regulated air source, ensure vents function between 2 to 10 psig.

NOTE: If pressure checking means are unavailable, vent fitting replacement is recommended at least once per year.

M. Install cleaned vent caps or replace with new assembly if required.

N. Clean bus bars and hardware, using care not to remove plating material. If this occurs, replace part.

O. Ensure cleaning solutions and emery dust residue are cleaned from hardware.

NOTE: Contact surfaces of cell terminals may be polished with fine file or emery paper. Do not remove excessive amount of material. Clean metal particles and dust from cells after cleaning.

CAUTION: DO NOT USE TOOLS TO CHECK CONTACT PIN LOOSENESS, THIS WILL DAMAGE FINISH.

P. Inspect disconnect receptacle in battery case, contact pins should be clean and unmarred both inside and outside case, pins should not rotate or be loose in sockets when turned with fingers.

Q. Disconnect receptacle plastic material should be clean and free of damage, area where receptacle attaches to case should be free of moisture, corrosion and foreign material. If receptacle requires disassembly for cleaning, replace any damaged gaskets during assembly.

R. Assemble battery. Place cells back in case, using care to place polarity marks in correct position.

S. Inspect cells for slight mechanical wear marks, if marks are evident, locate cells differently from original positions.

T. Add replacement cells as required to fill out 19/20 positions and identify cell location on data sheet.

U. Install bus bars. Torque bus bar connections as per battery manual.

V. Install individual cell shorting straps. Check cell voltages first and if necessary drain cells until each is less than 0.20 volts, do not short cells having higher voltage than this, it may cause heating or possible internal damage.

W. When all cells have been shorted for 1 hour, install main terminal shorting bar, then remove individual shorting straps.

X. Perform torque check on fittings and connections as per battery manual

Y. Charge battery STEP 8 prior to installation in aircraft.

STEP 8 — BATTERY CHARGING (AFTER MAINTENANCE)

A. Ensure all cell shorting straps and terminal shorting bars are removed.

B. Charge battery as per manufacturer's instructions. At end of charge, all cells should read between 1.55 and 1.75 volts with charge still in progress. Do not exceed the manufacturer's specified max charge rate. Any cell with reading outside of these limits should be rejected.

C. Perform Electrolyte Check STEP 3 and Leakage Test STEP 4.

D. Install battery and perform Battery — Check, this section.

4. Battery — Capacity Check (20 Cell Saft)

NOTE: The purpose of this discharge is to determine the electrical characteristics of the battery when it is removed from the aircraft, either for routine maintenance or to investigate a malfunction.

A. Discharge the battery at one of the current rates below until the battery reaches 20.0 volts. It is important that the discharge current be continually maintained at the selected value, and that the time of discharge be measured accurately.

DISCHARGE RATE	MINIMUM TIME FOR 1ST CELL TO REACH 1.0 VOLT
9 amps	3 Hrs. 24 Minutes
18 amps	1 Hr. 42 Minutes
36 amps	51 minutes

B. Monitor individual cell voltages periodically during the discharge, and record the time at which the first cell reaches 1.0 volt. It is not a cause for concern if a cell goes to zero volts or reverse polarity during the battery discharge. Simply short out such a cell's terminals for the remainder of the discharge. Stop discharge when the battery voltage reaches 20.0 volts.

C. If the time until the first cell reaches 1.0 volt equals or exceeds the values shown in the table above, it indicates that this cell has at least 85% of Its rated capacity. This indicates the battery has been receiving a good charge in the aircraft and that the cells are within an acceptable range of capacity balance. Therefore, after allowing the battery to cool to room temperature (at least 12 hours of rest), it need only-be charged, the electrolyte leveled and leak tested. It can then be placed back in service.

NOTE: During charging of the battery, the cover shall be removed and all cell relief valves loosened but not removed.

Check individual cell voltages at the beginning of charge. If any cell indicates an immediate voltage rise above 1.5 volts, inject into it about l0cc of distilled or de-mineralized water.

D. To charge battery, use one of the following procedures:

(1) Charge at 3.6 amps until battery voltage reaches 30 volts. Then continue charging at this same rate for 4 additional hours. The total charge time must be at least 14 hours but no more than 15 hours.

(2) Charge at 18 amps for at least 2 hours. If after this time the battery voltage has not yet reached volts, continue to charge at this rate until the voltage rises to this level. However, Go not charge at this rate for more than 2 hours and 30 minutes. Then charge at 3.6 amps for 4 additional hours.

(3) Charge at 36 amps for at least 1 hour. If after this time the battery voltage has not yet reached 31.4 volts, continue to charge at this rate until the voltage rises to this level. However, do not charge at this rate for more than 1 hour and 15 minutes. Then charge at 3.6 amps for 4 additional hours.

NOTE: During the last 15 minutes of charge at 3.6 amps adjust electrolyte level while the battery is still being charged.

CAUTION: THE ADDITION OF WATER BY ANY METHOD OTHER THAN THAT GIVEN BELOW IS PROHIBITED AS IT MAY CAUSE SPEWING AND LOSS OF ELECTROLYTE DURING OVERCHARGE.

E. Before adjustment of electrolyte level, remove relief valves with the special plastic wrench. Immerse valves and their brings in distilled or deionized water and let them soak to dissolve any salts. Cover cells with a protective sheet to prevent entry of foreign matter.

F. To check liquid level in the cell, insert the syringe into the cell opening until the shoulder of the nozzle rests on the valve seat. Withdraw plunger and check for any liquid in the syringe. If level is too low. syringe will remain empty, indicating that the end of the syringe nozzle did not reach the liquid in the cell. Any excess liquid in the cell will be drawn into the syringe until the level of electrolyte corresponds to the end of the nozzle. This is the correct level for the electrolyte.

G. If electrolyte is low, the procedure outlined below should be followed using only distilled or demineralized water.

(1) Draw a measured amount of distilled water, such as 5cc, into the syringe and inject it into the cell.

(2) With syringe nozzle resting on the valve seat, fully withdraw plunger on the syringe.

(3) If syringe remains empty, repeat Steps (1) and (2), counting the number of 5 cc injections required to achieve correct level.

(4) At the point in Step (2) when some excess liquid is drawn into the syringe, correct level for that cell has been reached. Expel the excess liquid into a container for later disposal.

(5) The number of cc's of distilled water required to fill the cell to the correct level will serve as an approximate guide to amount required for the remaining cells. The water in each cell, however, must be adjusted individually to correct level. It is important to check that the quantity of water added per cell does not exceed 25 cc. If consumption is too high, check setting of charging system or regulator. If this setting is correct, then shorten the period between servicing.

NOTE: Surface contamination anywhere on (or in) battery can provide leakage paths between exposed terminals. All terminals come out the top of cells. making this the prime area of consideration.

The permissible leakage current in a SAFT battery should not exceed 10 microamps. However. taking into consideration the precision instrumentation required to measure such a small current. the following test can be made using a standard electrical shop instrument:

CAUTION: NEVER CONNECT ANY AMMETER ACROSS THE TWO BATTERY TERMINALS. THE METERS LOW INTERNAL RESISTANCE WILL PERMIT LARGE CURRENTS TO FLOW WITH RESULTING DAMAGE.

H. Adjust the meter and input leads for a measurement of 500 milliamps.

I. Connect the negative lead from the instrument to the battery container.

J. Touch the positive lead from the instrument to the positive terminal of the battery and then to the positive terminal of each cell.

K. If. while performing the above, there is any deflection of the needle from zero. the insulation should be considered defective and the battery should be thoroughly cleaned.

L. If. after cleaning and assuring that everything is dry, a leakage current is still indicated by a deflection of the needle. It should be traced to the defective cell or cells which should be replaced.

M. If the battery is in good electrical condition, dry off the cell tops with compressed air. if necessary. and lightly grease its terminals and connections. Use only neutral (nonacid) petroleum jelly and apply it with a small paint brush.

5. Battery Shop and Storage Areas

A. Batteries should be stored and worked on in clean, well ventilated areas. There should be no acid petroleum vapors present. Lead acid batteries should not be used, stored or be allowed to pass through these areas.

B. Work areas should have temperatures kept within limits of 55° to 80°F. Storage areas should never get hotter than 120°F and may be kept as cool as 32°F.

C. Batteries in transit and storage (more than two weeks) lose energy and become electrically unbalanced if kept open circuited. It is therefore recommended that all storage and shipment be accomplished with batteries fully discharged. Either all individual cells should be kept shorted, or else the main terminals

fitted with a shorting bar. Such a bar should not be fitted until each cell has been separately drained of energy.

D. To best be able to achieve cell match, all the cells in a given battery should be alike. This means the same number of plates, the same amount of plate area, the same amount of active plate material, the same plate thickness, the same age and the same internal condition. All these factors are best achieved by insisting on the cells being from the same manufacturer, with the same part number. Cells which have been "rebuilt" have undergone handling, twisting and stretching which weakens their mechanical integrity. They may pass initial tests easily, but they are more prone to early failure than new cells. It is therefore desirable to keep rebuilt cells together and not to intersperse them among new cells.

E. The following schedule of checks is recommended to prevent operational battery difficulties.

(1) A spot check is required weekly, or more often if water use is excessive.

(2) A deep cycle check is required every 100 hours. This will restore cells to a near uniform condition.

(3) A complete check is required every 400 hours. This evaluates cells for gradual degradation and restores them to approximately equal status.

(4) A unit experiencing operating difficulties (hot spots, spewing, overheat, loss of voltage, etc.) should be withdrawn from service and given a complete check to weed out defective parts. Operation with a unit prone to erratic behavior may subject the aircraft to hazardous condition.

NOTE: The batteries will share their work most evenly if they are identical. Since different makers rate their units differently, matching nameplate ratings is not good enough. Use batteries from the same maker, with the same part number, and with the same type of cells wherever possible. Mismatched batteries should be used only as temporary measures, and upon being restored to proper duty, they should pass tests verifying their condition.

6. Ground Cable — Inspection

(Battery/Starter/External Power/Generator)

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Gain access to starter/external power/battery/generator ground bus, located in respective main gear wheel well. Ground bus is attached to wing lower panel and front wing beam.

NOTE: On aircraft having ASC 196, it is impossible to inspect battery cables and their tie points in nacelle battery relay boxes without removing aft end of thermo barrier box.

On aircraft having ASC 196, remove aft end of thermo barrier box and inspect cables and their tie points in accordance with the following steps.

On aircraft having ASC 194, the battery ground circuit has been changed. Inspect battery cables and each tie point in accordance with the following steps. Inspect starter cables from starter through the nacelle firewall feed-throughs to ground pads in accordance with the following steps.

B. Inspect battery, starter and external power, ground cables as follows:

(1) Unbolt ground connection.

(2) Inspect wires, terminals, washers, bolts, and bus bar plates for signs of overheating, deformation, corrosion, or dirt.

(3) Clean or polish surface as necessary to ensure electrical contact between wire terminals and bus bar.

(4) Reassemble, torque to 95 - 110 inch-pounds above normal drag of nut.

CAUTION: DO NOT ENLARGE BOLT HOLE BEYOND 7/16 INCH MEASURED ACROSS GREATEST DIAMETER.

(5) If limited corrosion is found in bus bar, remove by enlarging bolt hole.

C. Using a mirror and strong light inspect the generator ground cable for:

(1) Chafing

(2) Deteriorated insulation

(3) Corrosion build up at ground bus. If corrosion is evident, the cable must be unbolted and cleaned or polished as necessary to ensure positive contact with ground bus.

(4) Signs of overheating

(5) Security of attachment. If cable appears loose, it must be unbolted, cleaned, inspected and reinstalled or replaced.

7. Battery Plug Cable — Inspection

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Gain access to battery plug, located just aft of each battery in the battery compartment in the nacelle.

B. Disconnect quick disconnect plug from battery cables.

C. Slide tubing from terminal.

D. Inspect cable for fraying, broken wire strands, burn marks. (37 strands max. allowable.)

E. Slide tubing over terminal and secure.

F. Connect quick disconnect plug to battery cables.

G. Inspect for foreign objects then close battery compartment access panel.

8. Battery Compartment Visual — Inspection

A. Disconnect electrical connector from battery.

(1) Battery disconnects should be clean, contact springs in good condition, have no corrosion and have snug uniform fit on battery terminals.

(2) Battery cables should be clean, unfrayed and not showing any evidence of chafing fairleads.

B. Disconnect vent lines from battery.

(1) Venting systems should be clean, free of traps, and with ports in aircraft skin open. Moisture drain cans should be empty.

(2) Compartment ventilation provisions should be clean and in good working order.

C. Disconnect mounting hardware from battery.

(1) There should be no evidence of arcing from the battery to the aircraft structure. The area near the bottom of the case (next to restraint angles) and the slotted ends of the tiedown strap (on the cover) are the likely areas for this condition.

(2) If arcing signs are present, check aircraft installation to determine what insulation provisions have failed and correct the condition. The battery should then be cleaned up as necessary.

D. Remove battery from aircraft. The compartment should be cleaned and no foreign substances inside.

9. Battery Charging

A. Battery Charging with External Power Source.

(1) Turn off all unnecessary loads from main and essential dc buses.

(2) Ensure that BATT and EXT PWR switch in aircraft are OFF.

(3) Plug external power into external power receptacle located in left engine nacelle. Turn unit power switch on.

(4) Place EXT PWR switch in ON.

(5) Note ammeter reading on external power unit. This reading is current required by aircraft main dc bus.

(6) Place aircraft battery switch in NORM and note rise in current on external power cart ammeter.

NOTE: If the batteries are completely discharged, this rise in value may be as high as 400 amps.

(7) Observe dc bus voltmeter on upper overhead panel. It should indicate external power voltage (28 to 29 volts).

(8) Continue charging until ammeter has dropped to about the same reading recorded before the aircraft battery switch was closed. (Time required, approximately 10 to 15 minutes.)

(9) When current has dropped, batteries are charged and ready for service. Place aircraft battery switch in OFF. Place EXT PWR switch OFF. Turn external power unit off. Disconnect external power source from aircraft.

(10) If battery power is to low to energize the battery relay, perform the following:

(a) Ensure EXT PWR switch is ON. Ensure BATT switch is in NORM.

(b) Locate normal battery relay for associated battery, aft of battery, in battery compartment.

WARNING: ENSURE BATTERY SWITCH IS IN THE NORM POSITION BEFORE JUMPER WIRE IS PLACED ON RELAY. OTHERWISE, BATTERY WILL ATTEMPT TO TAKE A CHARGE THROUGH THE JUMPER WIRE.

(c) Momentarily place jumper wire between the GEN and BATT terminals of the normal battery relay. Relay should close allowing battery to charge through heavy duty contacts of relay.

B. Charging Batteries with APU Generator.

(1) Remove all unnecessary loads from main and essential dc buses.

(2) Place BATT switch in EMER.

CAUTION: MONITOR APU GENERATOR CURRENT. DO NOT EXCEED GENERATOR RATINGS AS GIVEN IN OPERATION SECTIONS OF 24-2-7 OR 24-2-8.

(3) Start the APU gas turbine unit and connect the APU generator output to the essential dc bus by placing the APU generator switch to ON postion. Batteries are now being charged from APU generator. A charge of one hour will provide sufficient battery capacity for one engine start sequence.

NOTE: One start sequence is defined as three crankovers of a maximum of 30 seconds duration.

(4) If load exceeds the generator limitations, return battery switch to OFF. Depress one BATT DISC switch (on Aircraft 1 - 200, 322 and 323 not having ASC 192, pull L or R BATT EMER circuit breaker). Again place battery switch to EMER. If generator current still exceeds limit, pull L NORM FUEL PUMP circuit breaker. Since this action forced APU to suck its own fuel. Listen to APU to confirm it is still running.

NOTE: When the batteries are known to be low, start the APU as follows:

(a) Switch battery switch and all individual circuit switches to OFF.

(b) Set battery switch to EMER.

(c) Quickly set APU MASTER SW to ON and PRESS APU START button. Amber START light should come on.

NOTE: Step (d) and (e) apply to Aircraft 1 - 200, 322 and 323 having ASC 192.

(d) When green OPER RPM light comes on immediately depress one BATT DISC switch.

(e) Turn APU GEN switch ON. If load still exceeds limitations, see Step (4) above.

NOTE: Step (f) and (g) apply to Aircraft 1 - 200, 322 and 323 not having ASC 192.

(f) When green OPER RPM light comes on, turn APU GEN switch ON.

(g) Immediately pull one emergency battery control circuit breaker, located on the right circuit breaker panel. If load still exceeds limitations, see step (4) above.

(5) Upon completion of 1 hour charge, APU generator should be removed from bus by placing APU GEN switch to OF. Gas turbine unit should be shut down.

(6) Check bus voltage on dc bus voltmeter. This is battery voltage. The charged batteries should indicate minimum of 24 volts.

NOTE: If a battery drops off the line during the charging procedure, it is an indication of too much bus loading. With the APU running and the fuel pump circuit breaker pulled, the ammeter should show only 15 amps with the batteries disconnected. The APU generator is connected to the essential bus, regardless of battery switch position, and the 15 amps is the APU load (relays solenoids, etc.). Wait about two minutes so the battery can recover its voltage. Then throw the battery switch to EMER, which allows the battery to close its own relay and the emergency bus-tie relay, which places the battery on the line.

C. Charging Batteries in Battery Shop

Battery voltage (open circuit) behaves oddly during and after charging. During most of the charge this will vary from 26.2 to 27.6 volts. As the charge is completed it may rise to as high as 30.4 volts. Shortly after charge, it will subside to 24.7 volts or slightly below. There are several methods of bench charging, however Gulfstream Aerospace only recommends the constant current method. A recommendation for this procedure is to set a current of 8 amps and maintain it for 7 hours.

NOTE: Manufacturing tolerances produce individual battery cells all of which do not accept charge at the same rate. Thus in a charging battery, some cells are at a higher potential than others unless charging time is prolonged. Under normal (short charge) conditions the battery does not actually contain the energy it is supposed to. With successive charges and discharges the percentage energy present becomes less and less. Eventually the low cells begin to reverse polarity, and thereafter capacity is sharply limited. Also, after this occurs, the battery no longer holds a charge, but will run down just sitting idle. The corrective action for this condition is to "deep cycle" the battery every 100 hours. This procedure amounts to discharging the battery, eventually using shorting bars, so each cell is absolutely dead. The battery is then overcharged, using constant charging. Fast charging methods do not resolve cells to their proper conditions, although they will give apparently good results, the effect will not last. For Gulfstream I batteries, it is recommended that the slow charge by at 8 amps for 7 hours. This is the only way to regain full capacity from the nickel cadmium battery.

D. Water Level Check Procedure

NOTE: Water level only rises to high levels as a cell becomes charged above its plateau level. Optimum operations is to have as much liquid in the cell as is possible without spewage in normal use. It is therefore necessary to get the cell into the region of final charge before water level adjustments can be made. 1.60 volts per cell is midway in this region and is used as a bench mark. With reasonable cell balance, when the first cell is at 1.60 volts, the lowest should be at least to the top of its level.

(1) Check liquid level in all cells. Immediately after charging, the level should stand 1/4 inch above the top of the plates (up to the step of the spacer). After a 3 hour stand, the level should be 1/8 inch above the top of the plates.

NOTE: Do not delay longer than 3 hours after charging before making this check.

(a) Add pure distilled (or demineralized) water to bring the level to the proper point. Note the quantity added on the report card.

(b) If a cell has a record of frequent water additions, it should be kept under surveillance. Continued high use of water is grounds for rejection.

(c) In a complete battery, figure the average water added per cell. Any cell requiring more than 10 cc above the average should be rejected.

(d) Charge at 8 amps rate until all cells register between 1.55 volts and 1.75 volts. Any cell failing to make 1.55 volts within a total of 7 hours charge time should be rejected. Any cell reading more than 1.80 volts should be rejected.

(e) If a cell is to be replaced, discharge the battery at 34 amp rate (or 17 amp rate). As each cell falls below 0.20 volts, short circuit it with a shorting strap. This action is required to prevent cell reversal. When all cells are shorted, replace the faulty cell (or cells). Then repeat all preceding steps required by the check being performed.

NOTE: Any cell showing signs of heavy spewage should have its specific gravity checked using a suitable hydrometer. Values between 1.240 and 1.320 are acceptable. Addition of potassium hydroxide in the field is not advisable. Therefore, a low specific gravity reading is grounds for removal. Proper material to be used is potassium hydroxide, KOH SP. GR. 1.320 - Reagent Grade.

E. Discharge and Short Circuit Cell Procedure

NOTE: The only way to get cells to an equal state is to discharge them to zero. If they are then brought up evenly, their final charge states are approximately equal. The safest time to perform mechanical work on the battery is with it fully discharged and shorted.

(1) Discharge the battery at 34 amp rate (or 17 amp rate).

CAUTION: A BATTERY SHOULD NEVER BE BROUGHT TO A VERY LOW STATE OF CHARGE UNLESS THIS CELL MONITORING AND SHORTING PROCEDURE IS FOLLOWED.

(2) Monitor individual cell voltages. As each cell falls to 0.20 volts, place a shorting strap so as to short circuit it. This procedure is necessary to prevent any of the cells from becoming reversed.

(3) Continue the discharge until all cells are short circuited.

(4) After all individual cells are shorted, the main terminal shorting bar is applied.

(5) Once the main terminal shorting bar is inplace, the individual cell shorting straps may be removed.

(6) The battery should be tagged identifying it as being "discharged and shorted".

(7) Voltage recovery check procedure

NOTE: Cell recovery voltage, taken a given time after shorting bars are removed, provides a ready means of detecting high resistance short circuits and damaged internal connections.

(a) Start out with cells shorted (or main terminal shorting bar in place) and the battery in a clean condition.

(b) Install one ohm cell shorting jumpers, removing shorting straps and main terminal shorting bars.

(c) Let sit for 16 to 17 hours.

(d) Check individual cell voltages with the one ohm resistors still in place. No cell should show more than 0.20 volts. Now remove the one ohm jumpers.

(e) After 24 hours, read and record (on the battery report card) all cell voltages. Any cell failing to attain 1.08 volts should be rejected.

10. Replacing Individual Cells

If a defective or damaged cell is found, the battery should be removed from the aircraft and the cell replaced. To accomplish this, proceed as follows:

A. Discharge the battery first through a loading resistor and then with a short-circuiting bar applied to each cell in turn.

B. Remove all vent caps on all cells. This relieves the pressure in the cells which might be caused by gas. Other cells may expand when the defective cell is removed if gas pressure is not relieved.

C. Tighten all vent caps. This prevents any spilling of electrolyte.

D. Remove intercell connectors (bus bars) from the defective cell.

E. Replace the defective cell with discharged, serviceable cell. Be sure the replaced cell is installed with the same polarity as the defective cell. A red + sign is located at one of the intercell connector points.

F. Connect the intercell connectors and torque to 35 - 50 inch-pounds.

G. Change battery prior to installing it in the aircraft.

11. Cell Puller Assembly Fabrication

Detailed information required for fabrication of a cell puller assembly is contained in Figure 202.

12. Battery Check

NOTE: Steps A through G apply to Aircraft 1 - 200, 322 and 323 not having ASC 192.

A. Connect left and right batteries.

B. In cockpit, pull L or R NORM BATT circuit breaker (as applicable).

C. Place BATT switch to NORM.

D. Observe essential dc bus voltmeter for indication of 24 volts (approximately)

E. Energize one fuel boost pump and E-inverter (ESS AC SEL SW to NORM).

F. Observe essential bus voltmeter. It must read at least 22 volts.

G. To check opposite battery, push in circuit breaker that was pulled in Step B above, and pull opposite circuit breaker. Voltage should read minimum of 22 volts.

NOTE: Steps H thru N apply to aircraft 1 - 200, 322 and 323 having ASC 192.

H. Connect battery cables to quick disconnect.

I. In cockpit, depress L or R BATT DISC switch, as applicable.

J. On center overhead panel, place BATT switch to NORM.

K. Observe essential bus voltmeter for indication of 24 volts (approximately).

L. Energize one fuel boost pump and E-inverter (ESS AC SEL SW to NORM).

M. Observe essential bus voltmeter. It must read at least 22 volts.

N. To check opposite battery, connect the battery disconnected in Step I. above, and depress opposite disconnect switch. Voltage should read a minimum of 22 volts.

NOTE: The following steps apply to all aircraft.

O. Turn off boost pump and E-inverter.

P. On center overhead panel, place BATT switch to OFF.

Q. Disconnect batteries if no longer required.

13. Battery Circulator Fan — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF.

A. Removal

(1) Open battery access cover.

(2) Disconnect electrical connectors to circulator fan and remove battery access door.

(3) Remove fan electrical connectors.

(4) Remove fan.

B. Installation

(1) Install circulator fan.

(2) Connect electrical connectors to fan.

(3) Install battery access door.

(4) Connect electrical connectors to circulator fan.

(5) Perform Battery Cooling Fan / Inverter — Functional Test, this Section.

14. Battery Cooling Fan / Inverter — Functional Test

A. Ensure batteries are connected.

B. In right circuit breaker panel close L BATT CONT and R BATT CONT circuit breakers. In inverter junction box circuit breaker panel close BATTERY AIR CIRCULATE circuit breaker.

C. Place battery master switch to NORM.

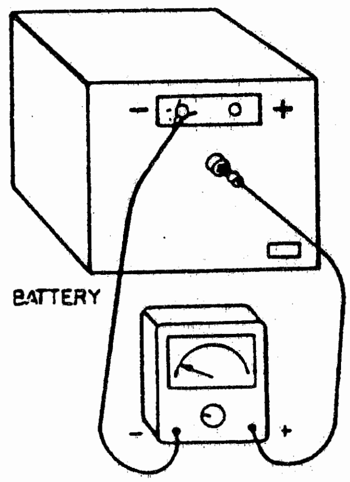
D. Heat right battery compartment thermal switch using a heat gun or some other safe method of heating at a temperature of approximately 90°F, the switch should close and right battery fan will start operating.

NOTE: Direction of airflow must be inboard.

E. Cool thermal switch with ZERO-MIST or some other method of cooling. At a temperature of approximately 75°F, the switch should open and fan will stop circulating.

F. Repeat above Steps D. and E. on left battery compartment components.

G. Place battery master switch off.

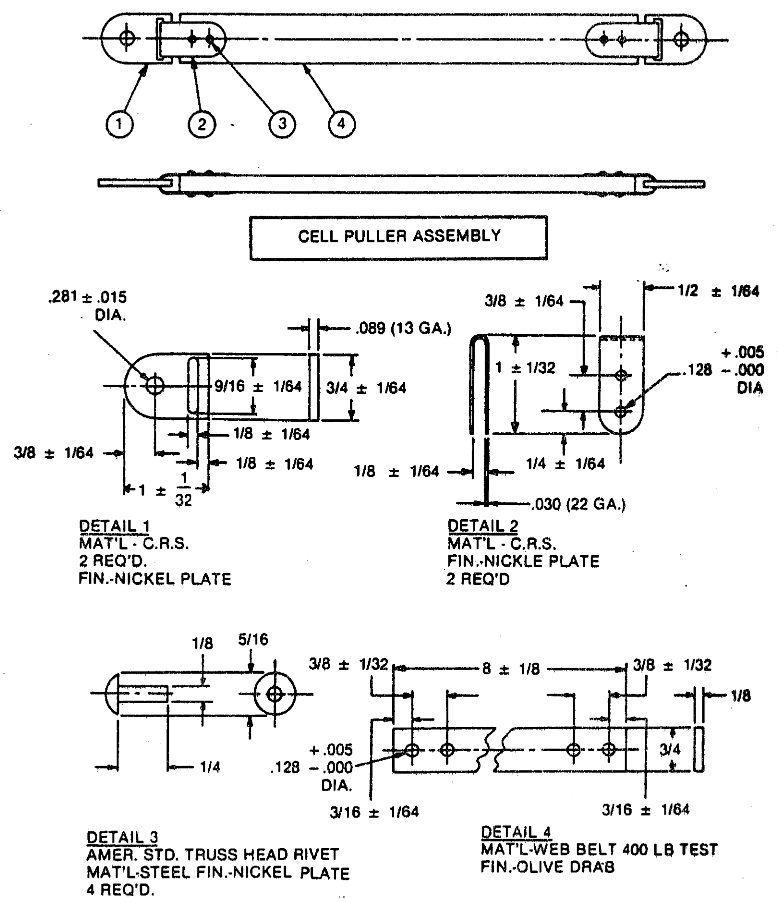


Resistance Versus Temperature
(For pins B and C)

− 20°F	5.5 Meg.
0°F	3.05 Meg.
+ 32°F	1.08 Meg.
+ 75°F	300.0 k
+120°F	51.1 k
+170°F	33.2 k
+190°F	22.1 k

Simpson 260 or Triplett
20.000 Ohm Per Volt Test Meter

Simpson 260 or Triplett
20,000 Ohm Per Volt Test Meter
Figure 201.



NOTE: THE ABOVE PART CANNOT BE PURCHASED. IT MUST BE HAND FABRICATED.

Cell Puller Assembly
Figure 202.

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50 AMP APU GENERATOR — DESCRIPTION / OPERATION

1. General

Unless subsequently altered, Aircraft 1 - 60 have APU generators rated at 50 amps, 27 volts, at 4000 generator rpm.

The dc generator is mounted on the APU (Auxiliary Power Unit). This generator is shunt wound and is driven by the accessory gear train of the APU by a belt. Cooling air is supplied to the generator from the compressor inlet duct. A voltage regulator, reverse current cutout, a control switch, and a generator warning light are provided for this generator.

The generator is rated at 150 amps for one second, 75 amps for five minutes and 50 amps for continuous operation. The reverse current cutout relay is rated at 100 amps. Otherwise it is similar to the units used in the engine-driven generator systems. The voltage regulator in this system is set at 27.0 $\pm$ 0.2 volts. This low voltage forestalls the possibility of this generator being on the line with an engine-driven or external power generator, the reverse current cutout acting as the protective device. The low voltage also acts to limit the maximum load. The voltage regulator is interchangeable with that used in the engine-driven generator.

The APU ammeter is one-half of a volt-ammeter placarded APU GEN AMPS, the voltmeter half being connected to the essential dc bus and is placarded DC BUS VOLTS. It is located in the cockpit on the upper overhead panel. The shunt for this ammeter is in the ground leg of the generator and is physically located in the APU relay box. The ammeter enables the crew to monitor the APU generator loads when this source of power is being used. The ammeter scale is 0-150 amps. In operation, the APU generator switch is used similarly to the generator control switch of the engine-driven generator system with two basic differences:

A. To prevent loading the APU engine before sufficient APU rpm is attained, the control circuit is routed through the APU loading relay. This loading relay is energized when APU speed reaches 95 percent (34,200 rpm). At this point and above, the unit can carry its normal load. The loading relay is located in the APU relay box.

B. On aircraft 1 - 100 and 114 not having ASC 158A, the generator output, through the APU generator control switch, provides the power to the APU generator warning relay contact to illuminate the APU GEN OFF light in the APU control section of the center overhead panel.

2. Major Components and Locations

A. Center Overhead Panel

- APU GEN. ON-OFF switch.
- APU generator ammeter.

B. Right Fuselage Station 133 Panel

- APU generator warning relay.
- APU generator circuit breaker (circuit breaker panel).

C. APU generator voltage regulator, below radio rack, Fuselage Station 169 and 193.

D. 200 amp APU current limiter, inverter fuse panel Fuselage Station 193.

E. APU Relay Box

- APU generator reverse current cutout.
- APU starter relay.
- APU ammeter shunt.

3. Operation

(See Figure 1)

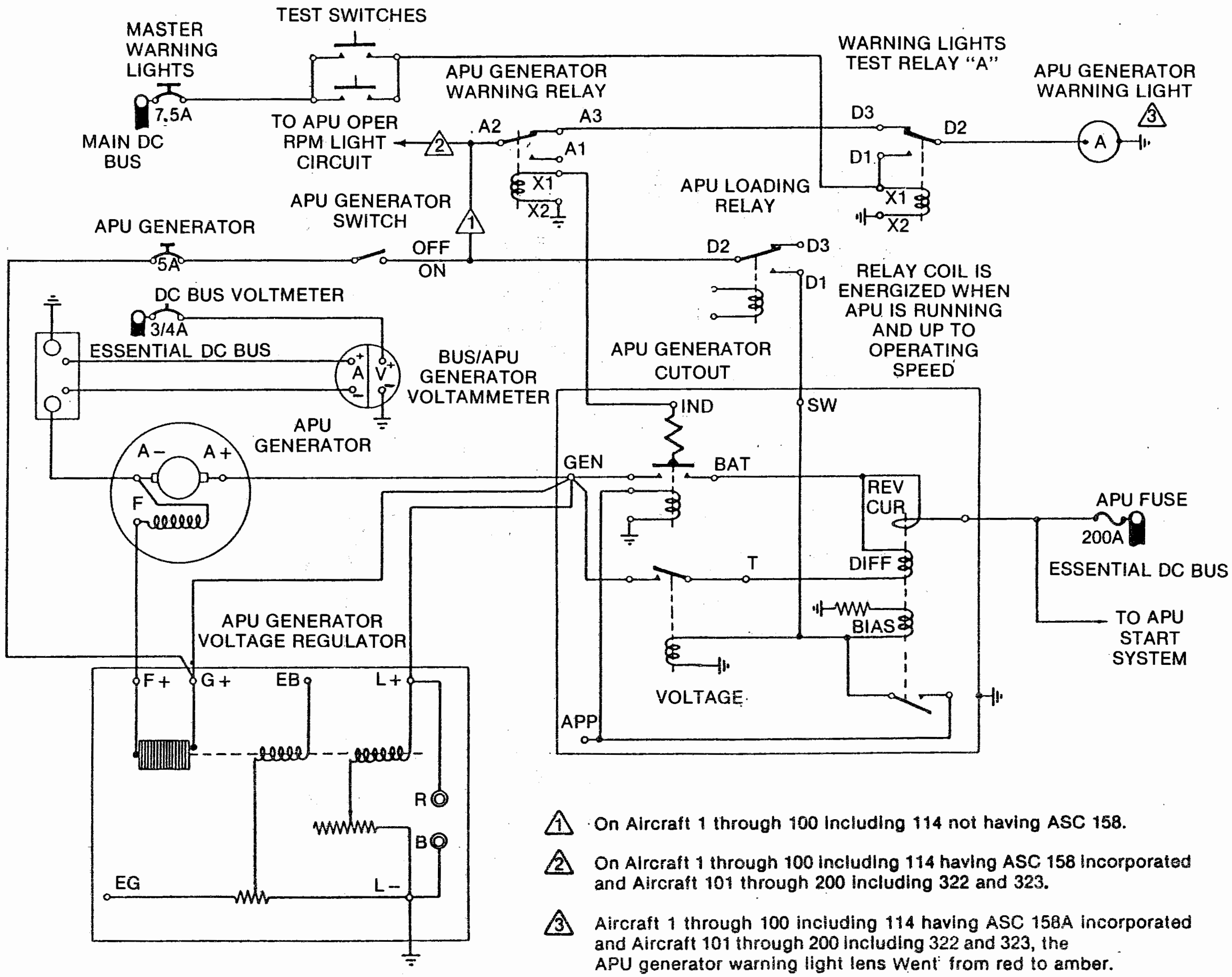
The APU, through its accessory section gear train and a drive belt, drives the 50-amp dc generator. Generator output is regulated at 27.0 $\pm$ 0.2 volts. The APU must be up to its proper operating rpm before the generator can supply power to the essential dc bus. Switching circuit power is taken directly from the self-excited generator, therefore the generator must be putting out and the APU up to operating rpm before the generator can be placed on the line. Switching circuit operation is as follows: power is taken from generator output, routed to the APU generator circuit breaker, then to the APU generator control switch located on the center overhead panel. When the switch is placed in the ON position, power continues on through the closed contacts of the APU loading relay (energized when APU is up to operating rpm - 95%) to the voltage relay coil of the reverse current cutout. Providing that the generator output is of correct polarity, and is 0.35-0.65 volts above the voltage of the essential dc bus, the main contactor of the reverse current cutout will close, connecting generator output to the essential dc bus.

4. APU Generator/Main Engine Starter Interlock Circuit

(See Figure 2)

An interlock circuit may be added, at the option of all operators, which will make it impossible to overload the APU generator during the start of an engine.

The introduction of one small relay in a very simple manner acts to prevent either cranks or ground starts as long as the APU generator is delivering an output. This interlock circuit has no effect on air start ignition, and is therefore incapable of causing any difficulty in flight. The relay to be added may be any 28 volt coil relay with a set of normally closed contacts having a capacity of at least two amps. The three wires to be cut into run from the upper overhead panel in the cockpit, back through the ceiling to Fuselage Station 133, then down through the right Fuselage Station 133 relay panel, and then aft under the floor. The relay can be added anywhere along the line.

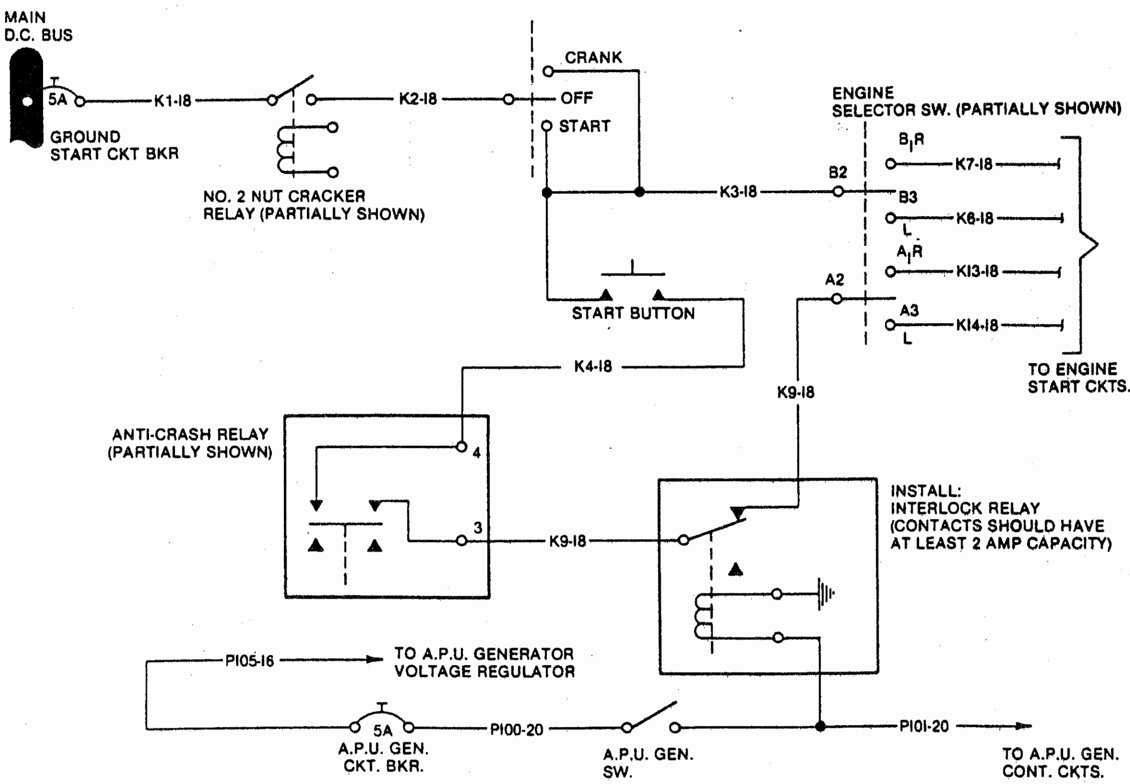


1 On Aircraft 1 through 100 including 114 not having ASC 158.

2 On Aircraft 1 through 100 including 114 having ASC 158 incorporated and Aircraft 101 through 200 including 322 and 323.

3 Aircraft 1 through 100 including 114 having ASC 158A incorporated and Aircraft 101 through 200 including 322 and 323, the APU generator warning light lens Went from red to amber.

50-Amp APU Generator Control Circuit — Schematic
Figure 1.



Interlock Circuit — Schematic
Figure 2.

50 AMP APU GENERATOR — MAINTENANCE PRACTICES

1. APU (50 Amp) Generator / Generator Belt — Removal / Installation

A. Removal

(1) Gain access to APU generator on left side of APU in tail compartment.

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

NOTE: For Generator Belt — Removal/Installation only, comply with Step A(4) and Steps B.(3) - B.(7) below.

(2) Remove attaching clamps, then remove cooling air duct.

(3) Disconnect electrical lead at generator terminal block.

(4) Remove cover from accessory section covering accessory section generator drive gear, then loosen bolts and nuts to relieve tension on belt. Remove belt and check for breaks or deterioration.

(5) Remove tension bracket adjusting bolt and washers. Remove attaching bolts, washers and nuts securing generator to mounting bracket, then remove generator assembly and tension bracket.

(6) Do not remove or disassemble mounting bracket unless damage is evident.

NOTE: Leave belt attached to accessory case drive.

(7) If new or replacement generator is to be installed, remove shaft drive gear from generator.

CAUTION: OBSERVE POSITION OF DRIVE GEAR WHEN REMOVING AND INSTALL IN SAME MANNER ON NEW GENERATOR. (SHALLOW 1/8 INCH RECESS TOWARD GENERATOR; DEEP 7/8 INCH RECESS TOWARD NUT AND WASHER).

NOTE: If generator is removed for brush inspection.

B. Installation

(1) Position generator in position on mounting bracket and secure with attaching bolts, washers and nuts. Adjust washers, as required, to provide radial alignment between driven gear on generator and drive gear in accessory section.

(2) Install tension bracket and secure with bolt, washers and nut. Tighten nut hand tight. Coat threads of adjusting bolt with anti-seize compound and install with washers as required to maintain a parallel between bracket and generator; then tighten screw sufficiently to allow generator movement.

NOTE: When installing adjusting bolt, washers must be installed so that 1/8 inch washer is against bolt head and 1/4 inch washer is between tension bracket and lug on generator. Remaining washers should be used to fill gap between 1/4 inch washer and generator lug, and to maintain gear alignment and proper drive belt tracking when tightening adjusting bolt.

(3) Check drive belt for indication of breaks or deterioration and proper belt tracking.

(4) Check drive gear and driven gear for condition and deterioration.

(5) Install drive belt over drive gear in accessory section and driven gear of generator.

(6) Loosen adjusting bolt and move generator until belt is snug.

(7) Place AiResearch tool 281959 gage over end of drive gear and belt. Tighten/loosen belt until 8.0 $\pm$ 0.010-inch is indicated between center of accessory drive and center of generator drive. (See Figure 201)

(8) Torque adjusting bolt to 19 - 21 foot-pounds. Remove gage. Safety-wire bolt.

(9) Tighten all bolts and nuts to ensure generator security.

(10) Install air duct and secure with clamps. Tighten all electrical leads.

(11) Inspect area for presents of foreign objects, security of all attachments.

(12) Install access covers.

(13) Start and run APU using normal procedures.

(14) Check for bearing noise and/or vibration.

(15) Check drive belt tracking and adjust radial adjustment of driven gear on generator and drive gear on accessory section as required.

(16) Perform APU voltage check.

2. **APU Reverse Current Relay — Removal / Installation**

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING THIS COMPONENT.

A. Removal

(1) Gain access to relay, located in the APU relay box, located on left side of APU enclosure, and accessible through double access opening on left side of tail compartment (from outside).

(2) Remove relay box cover.

(3) Disconnect, identify and insulate electrical leads.

(4) Remove mounting hardware and retain.

(5) Remove relay.

B. Installation

(1) Install relay using mounting hardware previously removed.

(2) Connect electrical leads as identified. Ensure tight electrical connections.

(3) Start and operate APU.

(4) Pull one battery circuit breaker to prevent belt stress before energizing APU generator.

(5) Check operation of relay with APU generator.

(6) Shut down APU.

(7) Depress battery circuit breaker previously pulled.

(8) Inspect area for presents of foreign objects, security of all attachments.

(9) Install relay box cover and close access panels.

3. **APU Generator Fuse — Removal / Installation**

A. Removal

(1) Gain access to inverter fuse panel (on aft side of Fuselage Station 193 bulkhead, right side).

(2) Locate APU generator fuse (current limiter-bolt downtype).

(3) Loosen holddown nuts at each end of fuse and slide fuse out from under bus bar, it is not necessary to remove nuts.

(4) Note fuse value.

B. Installation

(1) Install fuse of same value as one removed.

(2) Tighten down holddown nuts and torque to 65-95 inch-pounds.

(3) Inspect area for presents of foreign objects, security of all attachments.

(4) Close access previously opened.

(5) Start and operate APU using normal procedures.

(6) Shut down APU using normal procedures.

4. **APU Drive Belt/Accessory Drive Gear/Generator Driven Gear — Inspection**

   A. Access drive belt through left hand access covers.

   B. Check drive belt for indication of breaks or deterioration and proper belt tracking.

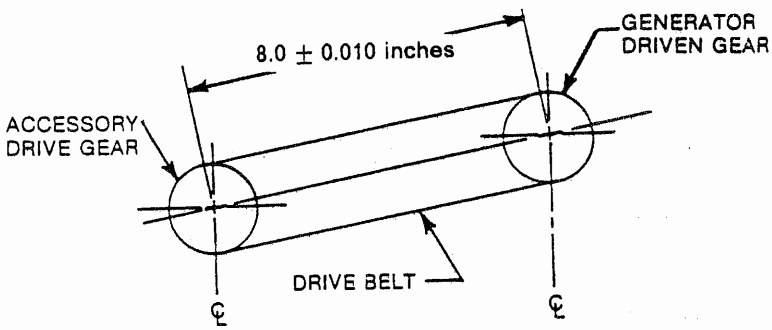
   C. Check drive gear and driven gear for condition and deterioration.

   D. Check distance (8.0 ± 0.010 inches on 50 AMP generator) See Figure 201

   E. Adjust if necessary.

   F. Inspect area for presents of foreign objects, security of all attachments.

   G. Install access covers.



Generator Driven Belt Adjustment (50 AMP)
Figure 201.

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200 AMP APU GENERATOR — DESCRIPTION / OPERATION

1. General

Aircraft 61 - 200, 322 and 323, excluding 114 have APU generators rated at 200 amps. 27 volts, at 4000 generator rpm.

The dc generator is mounted on the APU (Auxiliary Power Unit). This generator is shunt wound and is driven by the accessory gear train of the APU by a belt. Cooling air is supplied to the generator from the compressor inlet duct. A voltage regulator, a reverse-current cutout, an overvoltage protector, an overload control, a control switch, and a generator warning light are provided for this generator.

The reverse current cutout relay is similar to the units used in the engine driven generator systems. The voltage regulator in this system is set at 27.0 $\pm$ 0.2 volts. This low voltage forestalls the possibility of this generator being on the line with an engine-driven or external power generator, the reverse current cutout acting as a protective device. The low voltage also acts to limit the maximum load. The voltage regulator is interchangeable with that used in the engine-driven generator.

The overvoltage protector is a multi-contact circuit breaker that is tripped by a small solenoid. It is designed to prevent damage to electrical and electronic equipment from excessively high voltage developed by the generator under full field fault conditions. The solenoid sensing coil is connected in parallel with the generator field. When the field voltage reaches 28 $\pm$ 1 volt, the solenoid trips the protector causing the main and all auxiliary contacts to open. The main contacts L + and G + are connected to open the generator field circuit, thereby tripping the generator. Surge voltages will not actuate this unit. A low coefficient wire wound resistor is connected in series with the sensing coil to minimize changes in resistance caused by temperature variation. The unit is mounted on the bulkhead just forward of the dc voltage regulator below the radio rack. It is reset manually.

The overload control consists of a thermal actuated disc in series with the load terminals and an auxiliary single-pole double throw switch. The load terminals L-1 and L-2 are connected in the ground leg of the generator, in series with the ammeter shunt. It is located in the APU relay box. The auxiliary switch, through terminals S-1 and S-2, is in the APU generator control switch circuit. In the event of an overload, the heat created by the excessive flow will expand the thermal actuating disc, which in turn will actuate the auxiliary switch, breaking the control circuit. Breaking this circuit will open the reverse current relay cutting the generator load. The thermal disc is self-resetting. When the disc cools, it will reset itself, closing the control circuit to the reverse current relay, again putting the generator on the line. The nominal rating of the overload control is 175 amps, and it has continuous current capacity of 115 percent nominal rating or 201 amps at 77°F.

The APU ammeter is one-half of a volt-ammeter placarded APU GEN AMPS, the voltmeter half being connected to the essential dc bus and is placarded DC BUS VOLTS. It is located in the cockpit on the upper overhead panel. The shunt for this ammeter is in the ground leg of the generator and is physically located in the APU relay box. The ammeter enables the crew to monitor the APU generator loads when this source of power is being used. The ammeter scale is 0-300 amps. In operation, the generator switch is used similarly to the generator control switch of the engine-driven generator system with two basic differences.

A. To prevent loading the APU engine before sufficient APU rpm is attained, the control circuit is routed through the APU loading relay. This loading relay is energized when APU speed reaches 95 percent (34,200 rpm). At that point and above, the unit can carry its normal load. The loading relay is located in the APU relay box.

B. On Aircraft 1 - 100 and 114 not having ASC 158A, the generator output, through the APU generator control switch, provides the power to the APU generator warning relay contact to illuminate the APU GEN OFF light in the APU control section of the upper overhead panel.

2. Major Components and Locations

A. Upper Overhead Panel

(1) APU GEN. ON-OFF switch.

(2) APU generator ammeter.

B. Right Fuselage Station 133 Panel

(1) APU generator warning relay.

(2) APU generator circuit breaker (circuit breaker panel).

C. Below Radio Rack Fuselage Station 169-193.

(1) APU generator voltage regulator.

(2) APU generator overvoltage control.

D. 200 amp APU current limiter, inverter fuse panel, Fuselage Station 193.

E. APU Relay Box

(1) APU generator reverse current cutout.

(2) APU starter relay.

(3) APU ammeter shunt.

(4) APU generator overload control.

3. Operation

(See Figure 1)

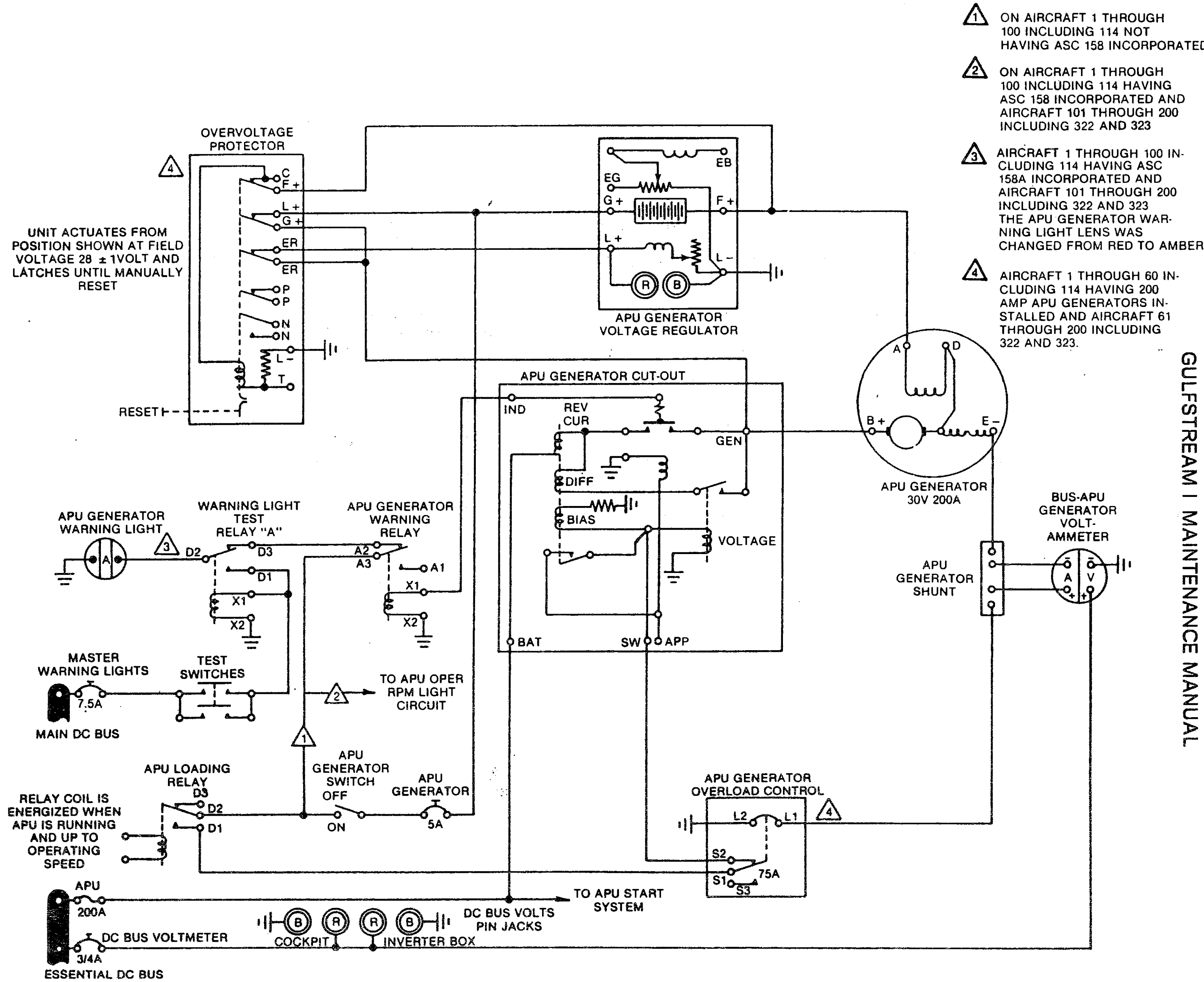
The APU, through its accessory section gear train and a drive belt, drives the 200-amp dc generator. Generator output of the unit is regulated at 27.0 $\pm$ 0.2 volts. The APU must be up to its proper operating rpm before the generator can supply power to the essential dc bus. Switching circuit power is taken directly from the self-excited generator, therefore the generator must be putting out and the APU up to operating rpm before the generator can be placed on the line. Switching circuit operation is as follows: power is taken from generator output, routed to the APU generator circuit breaker, then to the APU generator control switch located on the upper overhead panel. When the switch is placed in the ON position, power continues on through the closed contacts of the APU loading relay (energized when APU is up to operating rpm - 95%) to the voltage relay coil of the reverse current cutout. Providing that the generator output is of correct polarity, and is 0.35-0.65 volts above the voltage of the essential dc bus, the main contactor of the reverse current cut-out will close, connecting generator output to the essential dc bus. It also passes through a set of closed contacts of an overvoltage relay (set to actuate at 28.0 $\pm$ 1.0 volt field voltage) and through an overload control (a thermal switch, which when overheated, due to an overload condition breaks the control circuit).

4. APU Generator / Main Engine Starter Interlock Circuit

(See Figure 2)

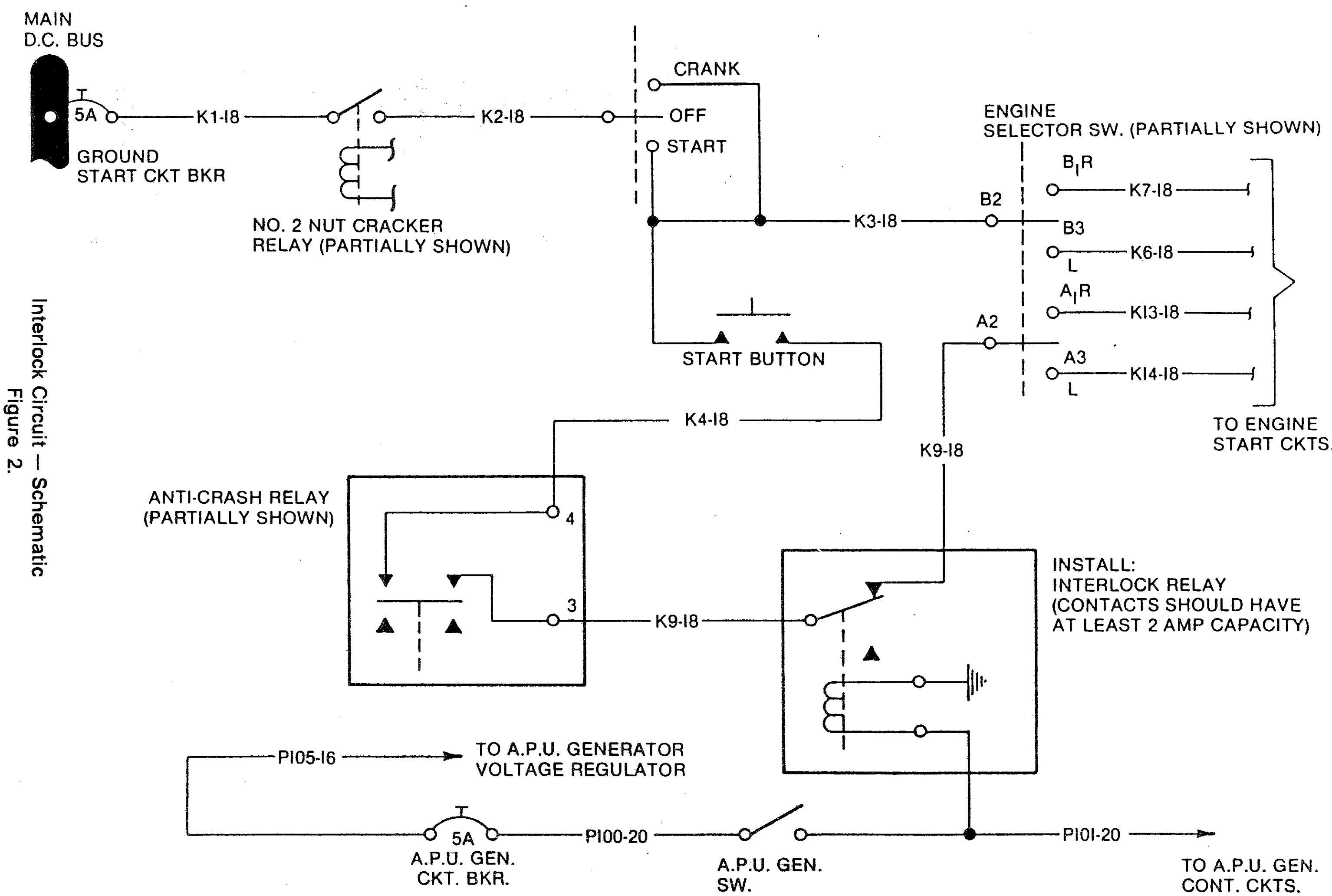
An interlock circuit may be added, at the option of all operators, which will make it impossible to overload the APU generator during the start of an engine.

The introduction of one small relay in a very simple manner acts to prevent either cranks or ground starts as long as the APU generator is delivering an output. This interlock circuit has no effect on air start ignition, and is therefore incapable of causing any difficulty in flight. The relay to be added may be any 28 volt coil relay with a set of normally closed contacts having a capacity of at least two amps. The three wires to be cut into run from the upper overhead panel in the cockpit, back through the ceiling to Fuselage Station 133, then down through the right Fuselage Station 133 relay panel, and then aft under the floor. The relay can be added anywhere along the line.



200 Amp APU Generator Control Circuit — Schematic
Figure 1.

"Reference to manufacturers in this Tech. Pub. are for identification purposes only and are neither specified nor furnished as a source for obtaining such parts."



Interlock Circuit — Schematic
Figure 2.

200 AMP APU GENERATOR — MAINTENANCE PRACTICES

1. APU (200 Amp) Generator / Generator Belt — Removal / Installation

A. Removal

(1) Gain access to APU generator on left side of APU in tail compartment.

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

NOTE: For Generator Belt — Removal / Installation only, comply with Step A(4) and Steps B(3)-B(7) below.

(2) Remove attaching clamps, then remove cooling air duct.

(3) Disconnect electrical lead at generator terminal block.

(4) Remove cover from accessory section covering accessory section generator drive gear, then loosen bolts and nuts to relieve tension on belt. Remove belt and check for breaks or deterioration.

(5) Remove tension bracket adjusting bolt and washers. Remove attaching bolts, washers and nuts securing generator to mounting bracket, then remove generator assembly and tension bracket.

(6) Do not remove or disassemble mounting bracket unless damage is evident.

NOTE: Leave belt attached to accessory case drive.

(7) If new or replacement generator is to be installed, remove shaft drive gear from generator.

CAUTION: OBSERVE POSITION OF DRIVE GEAR WHEN REMOVING AND INSTALL IN SAME MANNER ON NEW GENERATOR. (SHALLOW 1/8 INCH RECESS TOWARD GENERATOR; DEEP 7/8 INCH RECESS TOWARD NUT AND WASHER).

NOTE: If generator is removed for brush inspection.

B. Installation

(1) Position generator in position on mounting bracket and secure with attaching bolts, washers and nuts. Adjust washers, as required, to provide radial alignment between driven gear on generator and drive gear in accessory section.

(2) Install tension bracket and secure with bolt, washers and nut. Tighten nut hand tight. Coat threads of adjusting bolt with anti-seize compound and install with washers as required to maintain a parallel between bracket and generator; then tighten screw sufficiently to allow generator movement.

NOTE: When installing adjusting bolt, washers must be installed so that 1/8 inch washer is against bolt head and 1/4 inch washer is between tension bracket and lug on generator. Remaining washers should be used to fill gap between 1/4 inch washer and generator lug, and to maintain gear alignment and proper drive belt tracking when tightening adjusting bolt.

(3) Check drive belt for indication of breaks or deterioration and proper belt tracking.

(4) Check drive gear and driven gear for condition and deterioration.

(5) Install drive belt over drive gear in accessory section and driven gear of generator.

(6) Loosen adjusting bolt and move generator until belt is snug.

(7) Place AiResearch tool 281959 gage over end of drive gear and belt. Tighten/loosen belt until 9.5 ± 0.010-inch is indicated between center of accessory drive and center of generator drive. See Figure 201

(8) Torque adjusting bolt to 19 - 21 foot-pounds. Remove gage. Safety-wire bolt.

(9) Tighten all bolts and nuts to ensure generator security.

(10) Install air duct and secure with clamps. Tighten all electrical leads.

(11) Inspect area for presents of foreign objects, security of all attachments.

(12) Install access covers.

(13) Start and run APU using normal procedures.

(14) Check for bearing noise and/or vibration.

(15) Check drive belt tracking and adjust radial adjustment of driven gear on generator and drive gear on accessory section as required.

(16) Perform APU voltage check.

2. APU Drive Belt/Accessory Drive Gear/Generator Driven Gear — Inspection

A. Access drive belt through left hand access covers.

B. Check drive belt for indication of breaks or deterioration and proper belt tracking.

C. Check drive gear and driven gear for condition and deterioration.

D. Check distance (9.5 $\pm$ 0.010 inches on 200 AMP generator)

E. Adjust if necessary.See Figure 201

F. Inspect area for presents of foreign objects, security of all attachments.

G. Install access covers.

3. APU Generator (200 Amp) Overvoltage Relay — Removal / Installation

A. Removal

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING THIS COMPONENT.

(1) Gain access to the APU generator overvoltage relay, located on the aft side of the Fuselage Station 169 frame, under the radio rack, entrance compartment, right side.

(2) Disconnect, identify and insulate electrical leads.

(3) Remove and retain relay mounting hardware.

(4) Remove relay.

B. Installation

(1) Mount relay using hardware previously removed.

(2) Connect electrical leads as previously identified.

(3) Ensure reset button (under rubber boot) is pushed fully in.

(4) Start and operate APU using normal procedure.

(5) Pull one battery circuit breaker to prevent belt stress.

(6) Place APU generator switch to ON and check for proper generator operation.

(7) Turn off APU generator. Reset battery circuit breaker previously pulled.

(8) Shut down APU using normal procedure.

(9) Inspect area for presents of foreign objects, security of all attachments.

(10) Close access previously opened.

4. APU Generator (200 Amp) Overload Control — Removal / Installation

A. Removal

(1) Gain access to overload control, located in APU relay box, located on left side of APU enclosure, and accessible through double access opening on left side of tail compartment (from outside).

(2) Remove relay box cover.

(3) Remove, identify and insulate overload control leads.

(4) Remove mounting hardware and retain.

(5) Remove overload control.

B. Installation

(1) Install overload control using hardware previously removed.

(2) Connect leads as identified. Ensure tight electrical connections.

(3) Start and operate APU using normal procedures.

(4) In cockpit, pull L or R NORM BATT circuit breaker (on Aircraft 1 - 200, 322 and 323 having ASC 192, depress L or R BATT DISC switch) to prevent belt stress.

(5) Turn on APU generator and check for proper operation of APU generator.

(6) Shut down APU generator.

(7) Shut down APU using normal procedure.

(8) Reset L or R NORM BATT circuit breaker. (Aircraft 1 - 200, 322 and and 323 having ASC 192, depress L or R BATT DISC switch.)

(9) Inspect area for presents of foreign objects, security of all attachments.

(10) Install APU relay box cover and close access panels.

5. APU Generator Fuse — Removal / Installation

A. Removal

(1) Gain access to inverter fuse panel (aft side of Fuselage Station 193 bulkhead, right side).

(2) Locate APU generator fuse (current limiter-bolt-down type).

(3) Loosen hold-down nuts at each end of fuse and slide fuse out from under bus bar, it is not necessary to remove nuts.

(4) Note fuse value.

B. Installation

(1) Install fuse of same value as one removed.

(2) Ensure that forked end of fuse is securely under bus bar.

(3) Tighten down hold-down nuts and torque to 65 to 95 inch pounds.

(4) Inspect area for presents of foreign objects, security of all attachments.

(5) Close access previously opened.

(6) Start and operate APU using normal procedures.

(7) Shut down APU using normal procedures.

6. APU Generator Brush (200 AMP) — Inspection

A. Gain access to APU generator.

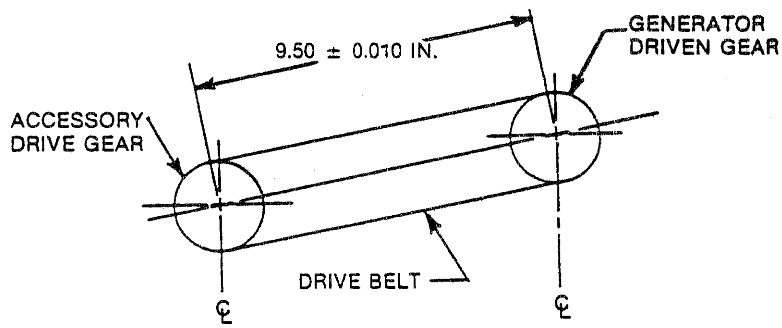
B. Disconnect and remove generator band cover.

C. Pull brushes and inspect for wear.

NOTE: Brushes are to be considered worn beyond limits if lower angle of brush is touching wear mark. If brushes are found to be worn beyond limits, generator must be returned to manufacturer or certified repair station for installation and seating of brushes.

D. Inspect condition of commutator and for presents of foreign objects, security of all attachments.

E. Install brushes and cover band.



Generator Drive Belt Adjustment (200 AMP)
Figure 201.

APU GENERATOR WARNING LIGHT CIRCUIT — DESCRIPTION / OPERATION

1. **Description**

The APU generator warning light, labeled GEN OFF, is located with the APU controls on the center overhead panel. When illuminated, the light glows red (Aircraft 1 - 100 and 114 having ASC 158A and Aircraft 101 - 200, 322 and 323 the light glows amber). The light provides the crew with an indication that the APU generator is supplying current up to its reverse current cut-out but is not connected to the essential DC bus when the APU generator control switch is in the ON position.

On Aircraft 1 - 100 and 114 having ASC 158A and Aircraft 101 - 200, 322 and 323, the light provides the crew with a indication that the APU generator is not connected to the essential dc bus regardless of APU generator control switch position (ON or OFF).

2. **Operation**

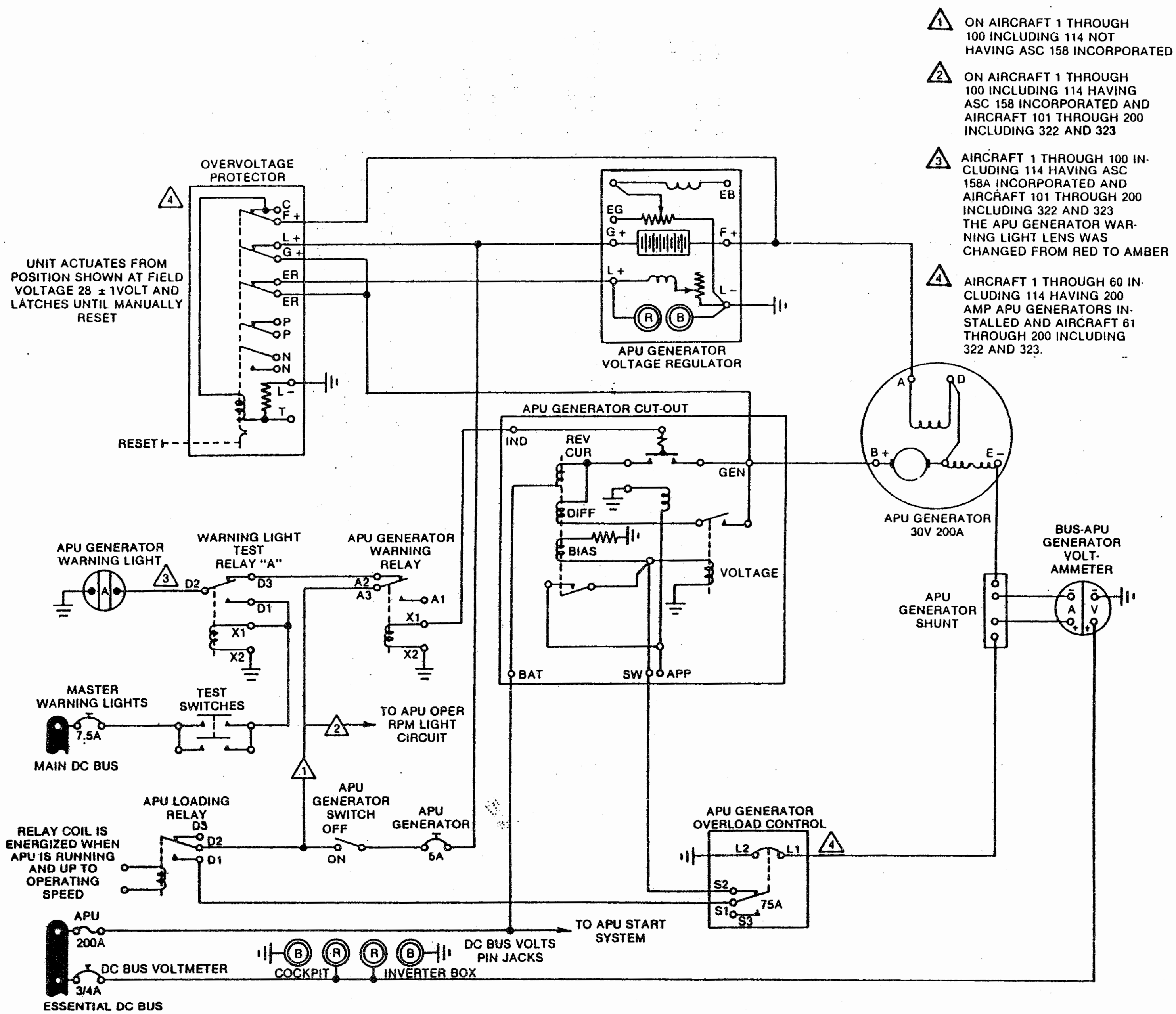
(See Figure 1)

The generator output, through the APU generator control switch, provides the power to the APU generator warning relay contact to light up the APU GEN OFF light. The IND terminal of the reverse current cutout relay feeds the APU generator warning light relay coil. When energized, this relay breaks the generator warning light circuit and the light should go out. When the reverse current cutout relay opens, the warning light circuit is again complete and the light will come on, provided the generator switch is still on.

NOTE: Since power for the illumination of the light is derived directly from the generator output, it will not light up if the drive belt is snapped, if the generator is not supplying power due to a malfunction, or if the overvoltage relay is tripped (200 amp installation).

On Aircraft 1 - 100 and 114 having ASC 158A and Aircraft 101 - 200, 322 and 323 power for the APU generator warning light is provided by the essential DC bus through the APU run indicating light circuit. Therefore, if for any reason the APU generator is not supplying power to the essential dc bus, the GEN OFF warning light will be on, provided that the APU is running at its operating rpm and some other source of power is supplying the essential dc bus.

The IND terminal of the reverse current cutout relay feeds the APU generator warning light relay coil. When energized this relay breaks the generator warning light circuit and the light should be out. When the reverse current cutout relay opens, the warning light circuit is again complete and the light will come on. This system gives a true indication of a malfunction since the warning light is not dependent upon the APU generator for power to light up.



200 Amp APU Generator Control Circuit — Schematic
Figure 1.

APU GENERATOR REVERSE CURRENT CUTOUT — DESCRIPTION / OPERATION

1. **Description**

The reverse current cutout for the APU generator is located in the APU relay box. It contains a differential relay, a voltage relay, and a main contactor relay. The differential relay has three coils controlling its contacts - the reverse current coil, the differential coil, and the bias coil.

2. **Operation**

With the APU GEN circuit breaker in closed APU GEN switch positioned to ON and APU running and up to operating speed, power from the APU generator is routed through the circuit breaker and closed contacts of the APU GEN switch and APU loading relay and energizes the bias coiled and voltage relay. This connects the differential coil and the reverse current coil across the open contacts of the main contactor relay. When the APU generator output is sufficient to produce current flow through these coils to the essential dc bus, the differential relay contacts close, energizing the main contactor relay. When the main contactor is energized, the APU generator is connected to the essential dc bus and power is available at the IND terminal to energize the APU generator warning light relay (Aircraft 1 - 100 and 114 not having ASC 158A). This causes the APU generator warning light to go out.

If the APU generator voltage falls below that of the essential DC bus, a current flows from the essential dc bus to the generator, through the reverse current coil and the differential coil. This reverse current causes the differential relay contacts to open, deenergizing the main contactor relay. The general output is then disconnected from the essential bus and the APU generator warning light come on (Aircraft 1 - 100 and 114 not having ASC 158A). When the APU generator voltage is sufficient to cause a current to flow from the generator to the essential dc bus, the differential relay contacts close, connecting the generator output to the essential dc bus and causing the warning light to go out (Aircraft 1 - 100 and 114 not having ASC 158A).

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APU GENERATOR REVERSE CURRENT CUTOUT — MAINTENANCE PRACTICES

1. APU Reverse Current Cutout Relay — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING THIS COMPONENT.

A. Removal

(1) Gain access to relay, located in the APU relay box, located on left side of APU enclosure, and accessible through double access opening on left side of tail compartment (from outside).

(2) Remove relay box cover.

(3) Disconnect, identify and insulate electrical leads.

(4) Remove mounting hardware and retain.

(5) Remove relay.

B. Installation

(1) Install relay using mounting hardware previously removed.

(2) Connect electrical leads as identified. Ensure tight electrical connections.

(3) Start and operate APU.

(4) Pull one battery circuit breaker to prevent belt stress before energizing APU generator.

(5) Check operation of relay with APU generator.

(6) Shutdown APU.

(7) Depress battery circuit breaker previously pulled.

(8) Inspect area for presents of foreign objects, security of all attachments.

(9) Install relay box cover and close access panels.

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BATTERY FAILURE MONITOR SYSTEM — DESCRIPTION / OPERATION

1. General

(Aircraft 1 - 200 and 322 and 323 having ASC 194 And Amendments)

The battery failure monitor system provides a simple digital readout of charge/discharge rate for each battery, and provides a warning light indication of abnormally high charge rate, so that a failing battery may be switched out of the system. The system consists of two dual-channel monitor warning amplifiers, two battery indicators, three battery failure warning lights, and two battery shunts, one in series with each nicad battery ground lead.

One current level monitor channel in each amplifier drives its associated digital readout. This digital readout or charge level monitor allows the operator to recognize excessive or abnormal charge rates, and if observed periodically during an elapsed time period, will allow determination of increasing charge current, an indication of thermal runaway.

The di/dt channel in each amplifier monitors charge current flow in its associated battery, and provides an automatic visual alarm signal (flashing light) should an abnormal or gradual increase in charge rate over a period of time occur. The flashing light is also activated if the charge rate is in excess of 15 amps for a significant time period.

The design of the system is such that a signal amplifier failure will not result in complete loss of ability to monitor either battery. Monitor warning amplifier A drives the current level readout for right battery, and the flashing warning light for the left battery. Monitor warning amplifier B drives the current level readout for the left. battery, and the flashing warning light for right battery. BATT FAIL light, located on the master caution lights display panel, will flash if either warning light is flashing. Each amplifier has a self-test lamp generator which substitutes a simulated rising current input for both channels, causing the two channels to respond in the same manner as to an actual thermal runaway. The self test is initiated by momentarily pressing the TEST button, and is automatically terminated in approximately 50 seconds.

2. Operation

(See Figure 1)

The battery failure monitor system derives power from the essential 28V dc bus through BATTERY MONITOR A and BATTERY MONITOR B circuit breakers and is operational when this bus is energized. The BATT MONITOR TEST momentary pushbutton switch, when actuated, initiates an internal automatic self test of the system.

A. Current level channel

Operation of the current level channel (Figure 1) is initiated by an internal free-running clock which provides synchronizing pulses for the A/D (analog to digital) converter and for the digital display system counter. Clock pulses to the BCD (binary-coded decimal) counter cause the counter to increment one count for each pulse, and the output develops a step voltage that increases with each successive clock pulse. This step voltage is sealed to the proper value at the bias amplifier. The output of the bias amplifier is fed to the A/D Comparator.

The voltage signal from the battery shunt, proportional to and of the same polarity as current flow in the battery, is fed to the preamp. One output of the preamp is fed directly to an A/D converter input, and the other through an inverter to reverse its polarity.

The A/D converter compares input from the preamp and inverter and causes the sign logic flip-flop to energize the plus segments of the plus-minus LED display if the inputs indicate a charging current through the battery shunt. The A/D converter also continuously compares the preamp inverter input with the increasing step voltage from the BCD counter. At such time that the step voltage equals that of the input from the preamp inverter, an output signal from the A/D comparator causes a strobe pulse output to the display latches. The count then present in the BCD counters is read and stored in the display batches. The latch outputs are fed through the tens and units drivers to provide an LED display equal to current flow through the battery shunt.

Clock pulses continue to advance the A/D BCD counter and the tens and units BCD counters until a count of 99 is reached. A synchronization pulse sent from the A/D BCD counter to the tens and units BCD counter nines input ensures equal counts in both counters. The next clock pulse resets all counters and the step voltage to zero, and the process begins again

B. di/dt Channel

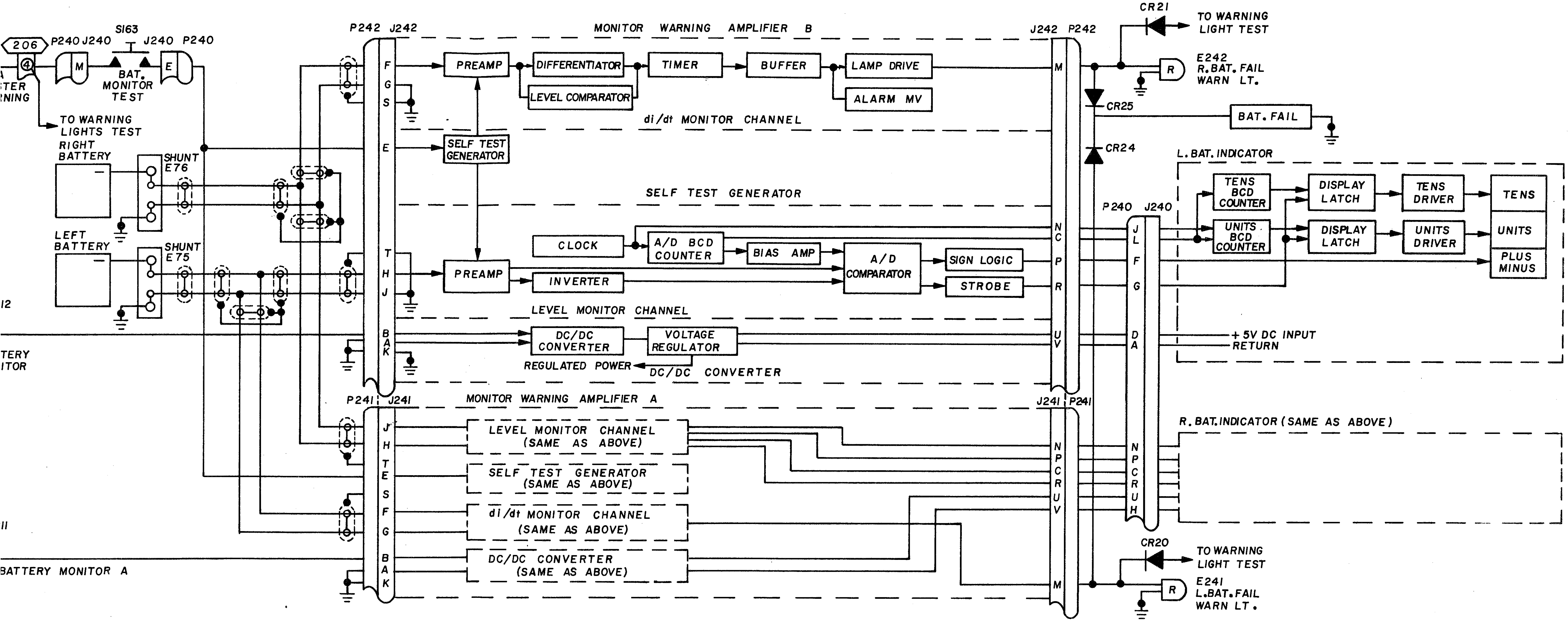
Operation of the di/dt channel will cause the warning lamps to be flashed at 80 pulses per second if current signals from the battery shunt indicate increases in charging rate in accordance with those shown below or points on an exponential curve drawn from these points.

Rate of Increase (Amps per Minute)	Duration
1.0	5 minutes
10.0	30 seconds
20.0	15 seconds

In addition, a steady state charge above 15 amps will energize the warning lamps in 4 to 5 minutes.

To avoid false alarms due to surges causes by changes in ambient temperature, minor changes in generator output voltage and other insignificant events, increases of less than 0.1 amp per minute at the low end of the curve are biased out of the circuit. In addition, sudden high current surges, sometimes occurring when a generator or APU is connected to a partially discharged battery, will not trigger the system warning lights.

Input from the battery shunt (See Figure 1) is scaled by the preamplifier and applied to the high-stability differentiator. The preamplifier is polarized and will not respond to discharge currents. The output of the differentiator is zero if the output of the preamplifier is steady or decreasing, and positive if the preamplifier output is increasing. The level of positive differentiator output, being dependent upon rate of increase, causes voltage level applied to the timer to vary with the rate of current increase. The output of the timer (actually an intergrator) will reach given level in a period of time determined by the value of the voltage applied. The timer output is applied through the buffer amplifier to the lamp drive when the timer output reached a fixed predetermined level. Input to the lamp drive is modulated by the alarm multivibrator at a rate of 80 pulses per minute causing the warning lamp to flash.



Battery Failure Monitor System — Partial Schematic
Figure 1.

BATTERY FAILURE MONITOR SYSTEM — MAINTENANCE PRACTICES

1. **General**

The procedures outlined in Steps 2. and 3. below provide a ground operational test of battery failure monitor system. Quick checks may be performed as required during system operation by momentarily pressing the TEST button. Both digital ammeters will register a charge (+) current that increases at a steady rate. At 30 ± 5 seconds after TEST button is depressed, L BATT FAIL, R BATT FAIL, and BATT FAIL lights will start blinking. Test cycle will then be automatically terminated within 40 seconds. Both fail lights will go out and ammeters will return to normal battery load current readings.

NOTE: This test may only be accomplished when digital ammeter readouts indicate discharge (−) values or charge (+) values less than 10 amps.

2. **Warning / Caution Lights — Operational Test**

A. On right circuit breaker panel, open BATT MON A and BATT MON B circuit breakers.

B. Ensure all other normally closed circuit breakers are closed.

C. Connect aircraft batteries.

D. Connect an external 28V dc power source to aircraft.

E. Select external power and batteries for normal connection to dc bus system.

F. On eyebrow panel, press and hold WARN TEST switch. Ensure following lights are ON:

L BATT DISC R BATT DISC L BATT FAIL	R BATT FAIL BATT FAIL

G. Replace any defective lamps. Repeat Step F. as required.

3. **Battery Monitor Indicator — Operational Test**

A. Connect external 28V dc power source to aircraft.

B. Connect aircraft batteries.

C. Select external power and batteries for normal connection to dc bus system.

D. Ensure that BATT MON A and BATT MON B circuit breakers, located on right circuit breaker panel, are closed. Ensure that all other normally closed circuit breakers are closed.

E. L BATT and R BATT digital ammeters lights should come on and display some value of positive (+) current (charging).

NOTE: If either digital ammeter readout is + 10 amps or more due to discharged condition of batteries, wait until readout is less than + 10 amps before continuing system test.

F. On upper overhead panel, momentarily depress BATT MONITOR TEST switch. L BATT and R BATT digital ammeters should indicate 00 + 1 amp, and begin increasing steadily in a positive direction. (Self-test simulated increasing charging current.)

G. After 30 ± 5 seconds, the R BATT FAIL, L BATT FAIL lights (located on upper overhead panel) and BATT FAIL light (located on master caution panel) should start to flash. (Lights will remain steady if R227 has been removed from monitor amplifier.)

H. R BATT FAIL, L BATT FAIL, and BATT FAIL lights should go out within 40 seconds and both digital ammeters should decrease from test value to less than + 10 amps (actual battery charge rate).

I. Disconnect external dc power source from aircraft. Both digital ammeters should display some value of negative (−) current (discharging).

J. Reconnect external dc power source to aircraft. Both digital ammeters should display some value of positive (+) current (charging).

K. Using the battery disconnect switches, disconnect the left and right batteries.

L. Both digital ammeters should display 00 $\pm$ 1 amp, and the L BATT DISC and R BATT DISC lights should come on.

M. Reconnect the batteries using the battery disconnect switches.

N. Remove electrical power from aircraft and disconnect external dc power source.

4. Start Bypass Relays (K132 and KI33) — Test

A. On right circuit breaker panel, close BATT MON A and BATT MON B circuit breakers. L BATT and R BATT digital ammeters should light up and display some value of positive (+) current.

B. On left circuit breaker panel, close NUTCRACKER CONTROL circuit breaker.

C. Turn on a relatively heavy dc load such as an inverter.

D. Disconnect external dc power source. L BATT and R BATT digital ammeters should display negative (−) (discharging) values of current.

E. On left circuit breaker panel, close GROUND START circuit breaker.

F. On lower overhead panel, place ENG SELECTOR to OFF.

WARNING: IN ORDER TO PREVENT INJURY TO PERSONNEL, BY INADVERTENT CRANKING OF ENGINES DURING PERFORMANCE OF STEPS (G) THROUGH (L) BELOW, ENSURE AREA AROUND PROPELLERS IS CLEAR OF PERSONNEL AND ENG SELECTOR SWITCH REMAINS IN OFF POSITION.

G. On lower overhead panel, place START SELECTOR switch to CRANK.

H. While observing L BATT and R BATT digital ammeter, press ENGINE START button switch. Both current readings should decrease to less than 50% of value before pushing switch. (Decrease should be to approximately 1/10 of former reading.)

I. Release ENGINE START button switch. Current readings shall increase to approximately their former values.

J. On lower overhead panel, place START SELECTOR switch to OFF.

K. On left circuit breaker panel, open GROUND START circuit breaker.

L. Return all switches to OFF or NORMAL and close all circuit breakers.

FIXED FREQUENCY AC POWER SUPPLY AND DISTRIBUTION SYSTEM — DESCRIPTION / OPERATION

1. General

Certain equipment in the aircraft requires fixed frequency ac energy. These units require an extremely stable source of voltage and frequency. Frequency straying from normal tolerances will make key instrumentation inaccurate, and lead to abnormal indications which can become dangerous. Gyro operated flight instruments, integrated flight systems, etc., fall in this class, as well as certain key engine instruments, such as oil pressure, fuel flow, and fuel quantity.

2. Description

The fixed frequency ac power supply and distribution system consists of three ac inverters (identified as A, B, and E), three ac distribution buses (main, instrument, and equipment), and the control switches, relays, and warning lights associated with the inverters and buses. (See Figure 1) All three inverters are dependent upon the dc power system for power.

Power for the A and B-inverters is supplied by the main dc bus. Power for the E-inverter is supplied by the essential dc bus. The three inverters are connected to produce 115 volt, 400 cycle, single-phase ac power. The control circuitry of inverters A and B derives power from the monitor dc bus so that failure of both generators (monitor dc bus de-energized) automatically disconnects these loads from the dc system. Control power for the E-inverter is supplied by the essential dc bus. Two auto-transformers are connected to the instrument ac bus to supply 26 volt, 400 cycle, single-phase power for engine instruments operation. These units are mounted beneath FLR-8.

Control over the inverters is as follows:

A. To give the crew control of essential ac circuits, an ESS AC BUS switch, located on the upper overhead panel, is provided. This switch has three positions: NORM, OFF, and EMER.

- In the NORM position, circuits are set up to power the instrument ac bus from the main ac bus if the latter has power. Should this condition not be met, the main ac bus sense relay will be relaxed, completing circuits which cause inverter E to run and feed the instrument ac bus.
- With the switch set to OFF the instrument ac bus will not be powered.
- Setting this switch to EMER will cause E-inverter to run and feed the instrument ac bus. This will occur no matter what other inverters are operating.
- The equipment ac bus always ties to the largest inverter on the line. It ties to the main ac bus if that bus is energized, otherwise it ties to the instrument ac bus.

B. A MAIN AC BUS switch is provided to control the main ac bus power. This switch, which is located on the upper overhead panel, has three positions: NORM, OFF, and STANDBY.

- The NORM position operates A-inverter, and connects it to the main ac bus via protected feeders.
- In the OFF position, no energy is delivered.
- STANDBY position operates B-inverter, and connects it to the main ac bus.

NOTE: Transfer from A to B-inverter is entirely controlled by the pilot. No automatic devices are employed.

C. A voltmeter (AC VOLTS) and an AC VOLTS SELECTOR switch is provided so that the crew may observe conditions on the instrument or main ac buses.

D. The RADAR AC BUS switch on the upper overhead panel is not wired. It is a three position switch of the same type as the MAIN AC BUS switch. It is installed as a facility for the installation of a third large inverter (radar or C-inverter). Most operators, however, do not find it necessary to include this third large inverter and its associated circuitry, since the 2500VA inverter supplies sufficient current for all requirements, and the weight savings utilizing two 2500VA inverters versus three 1500VA inverters is approximately 36 pounds. Facilities for the mounting of the radar C-inverter and mounting for associated relays, etc., are incorporated in the aircraft. No relays or wiring are supplied for this inverter. This instal-

lation will be disregarded for the remainder of this section, since it is not a part of the delivered aircraft. The RADAR position of the AC VOLTS SELECTOR switch is also inoperative and should be disregarded.

Aircraft 1 - 60 and 114 were equipped with 1500VA inverters. Aircraft 61 - 200, 322 and 323 were equipped with 2500VA inverters. Aircraft 1 - 200, 322 and 323 having ASC 222 (solid state inverters) are equipped with 2500VA A and B-inverters.

All inverters installed on Aircraft 1 - 119 were provided with "keep alive" circuits. This was designed to keep the tubes in the regulating section warm, and permit regulation of both speed and voltage to commence as soon as the inverter is started. Should an inverter be started up cold without the "keep alive" having been activated, acceleration will not bring the unit up to speed for several seconds. During the period, the output voltage will be very low and the input dc power will be more than normal. Aircraft 120 - 200, 322 and 323 have inverters using solid state regulation. These inverters do not require a "keep alive" voltage. The A and B-inverters on Aircraft 1 - 200, 322 and 323 having ASC 222 are solid state, therefore, requiring no "keep alive" voltage. All aircraft are still wired with the "keep alive" circuit in the event that the older, tube type regulated inverter is used as a replacement.

3. Operation

(See Figure 2 and Figure 3)

With the main and monitor dc buses energized, setting the MAIN AC BUS switch to NORM brings inverter A into operation. The output of inverter A is then fed to the main ac bus. The equipment ac bus relay is energized by main ac bus power and connects the equipment ac bus to the main ac bus, through the closed contacts of the equipment bus relay. With the ESS AC BUS switch set to NORM and the essential dc bus energized, the output of inverter A is fed to the instrument AC bus through the NORM contacts of relay E-2. (The relay is a double coil relay with a set of contacts for the NORM coil and a set for the EMER coil. A mechanical interlock device prevents both relay coils from operating their respective contacts at the same time. Since both sets of contacts are electrically common, this would connect two separate ac power sources to a common point.) The NORM coil of relay E-2 is energized by a circuit which is routed, from the essential dc bus, through the contacts of the energized main ac bus sensing relay and the ESS AC BUS switch NORM contacts.

Each bus has an ac bus sensing relay connected to it which functions to light its corresponding bus failure indicating light when the bus is de-energized.

In the event of failure of A-inverter, the main ac bus is de-energized, causing the main ac bus sensing relay to be de-energized. This completes a circuit from the 28V dc warning lights bus through the relaxed contacts of the main ac bus sense relay causing the main ac bus warning light to light up. De-energizing the main ac bus sense relay also breaks the circuit to the NORM coil of relay E-2 and completes a circuit from the essential dc bus in the EMER coil of relay E-2 through the NORM contacts of the ESS AC BUS switch. Relay E-1 is also energized by this circuit applying essential dc bus power to inverter E, bringing it to operation. The output of inverter E is then fed to the instrument ac bus through the EMER contacts of relay E-2. Since the main ac bus is de-energized at this time, the ac equipment bus relay is also de-energized, and connects the equipment ac bus to the instrument ac bus.

Setting the MAIN AC BUS switch to STANDBY removes the power from the inoperative A-inverter, energizes B-inverter. B-Inverter takes over the function of A-inverter of feeding all three buses. When power is reapplied to the main ac bus, the main ac bus sense relay is re-energized, breaking the control circuit to the E-inverter and the main ac bus warning light. This causes the warning light to go out and de-energizes the E-inverter, which is now standing by once more in the event that the B-inverter should fail.

If the ESS AC BUS switch is set to EMER, E-inverter is brought into operation to feed the instrument ac bus, regardless of the operating status of A or B-inverters. If either A or B-inverter is not operating at the time, the ac equipment bus relay will be de-energized, connecting the equipment ac bus to the instrument ac bus, through its relaxed contact. If either A or B-inverter is operating, the main and equipment ac buses are energized by A or B-inverter. E-inverter will only power the instrument ac bus.

4. **Component Location**

A. All three inverters are located under the radio rack between Fuselage Stations 133 and 193.

B. All relays associated with the inverter system except the ac equipment bus relay are located in the inverter relay panel.

C. The ac equipment bus relay is located in the cockpit at the copilots circuit breaker panel on the lower outboard side.

D. Inverter A and B current limiters (fuses) are located in the inverter fuse panel, aft side of Fuselage Station 193, right side of the aircraft.

E. All other circuit protective devices are located in the cockpit circuit breaker panels, or in the circuit breaker section of the inverter relay panel.

5. **Emergency DC Operation**

A. The A and B-inverters are inoperative in emergency.

B. The E-inverter is operative in emergency.

C. The ac bus warning lights on the master warning system will be operative within the limitations of the emergency dc operation of the master warning system (capsules only).

6. **Gulfstream Supplied AC, 400 Cycle Equipment**

The following 400 cycle ac equipment is supplied by Gulfstream as delivered to the customer. The buses to which they are connected are indicated.

A. AC Instrument Bus

(1) 115 to 26V ac, 400 cycle instrument transformer (left)

- Left fuel flow indication
- Left oil pressure indication

(2) 115 to 25V ac, 400 cycle instrument transformer (right)

- Right fuel flow indication
- Right oil pressure indication

(3) Left engine zones I and II (Graviner) fire detection system

(4) Right engine zones I and II (Graviner) fire detection system

(5) Left fuel quantity system

(6) Right fuel quantity system

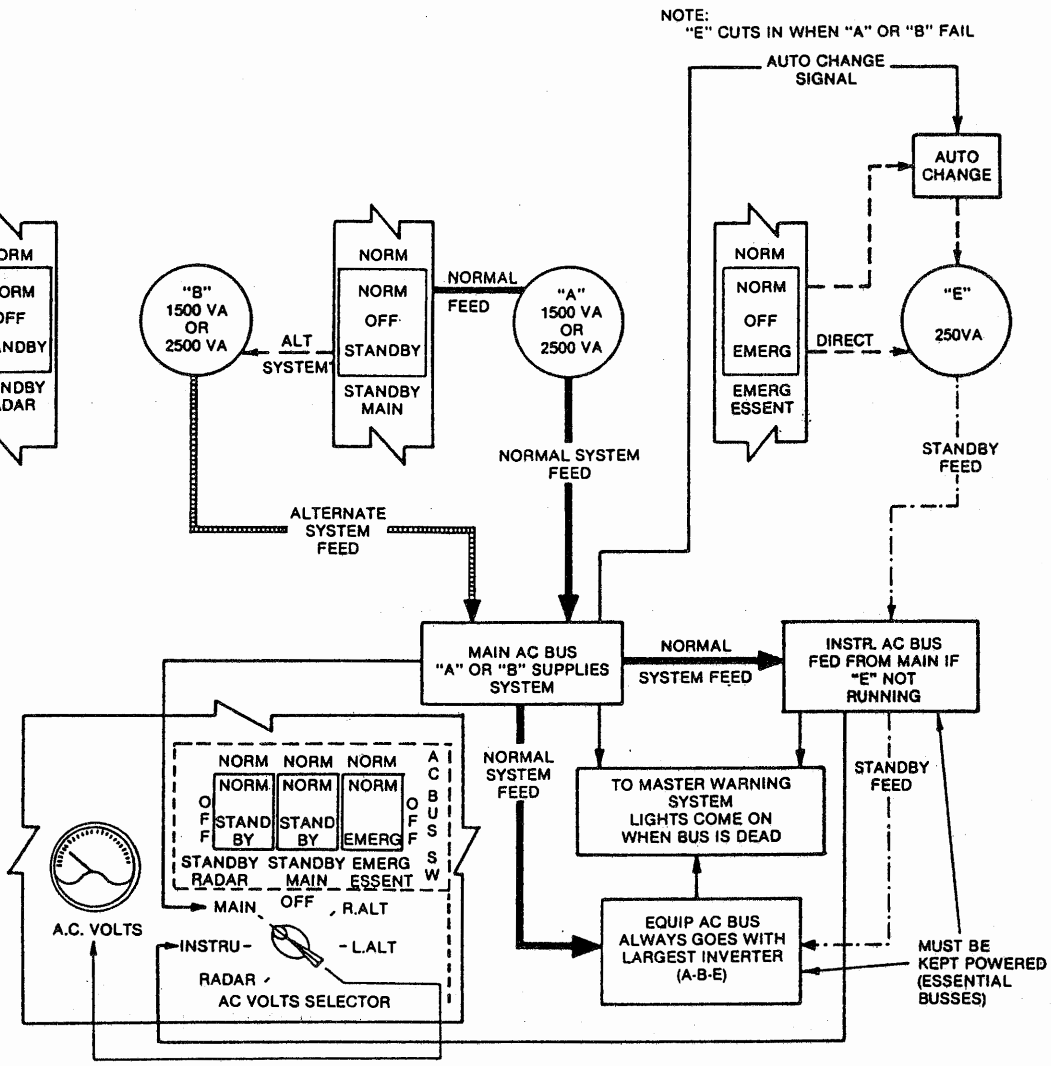
B. AC Main Bus

C. Windshield heat control ac.

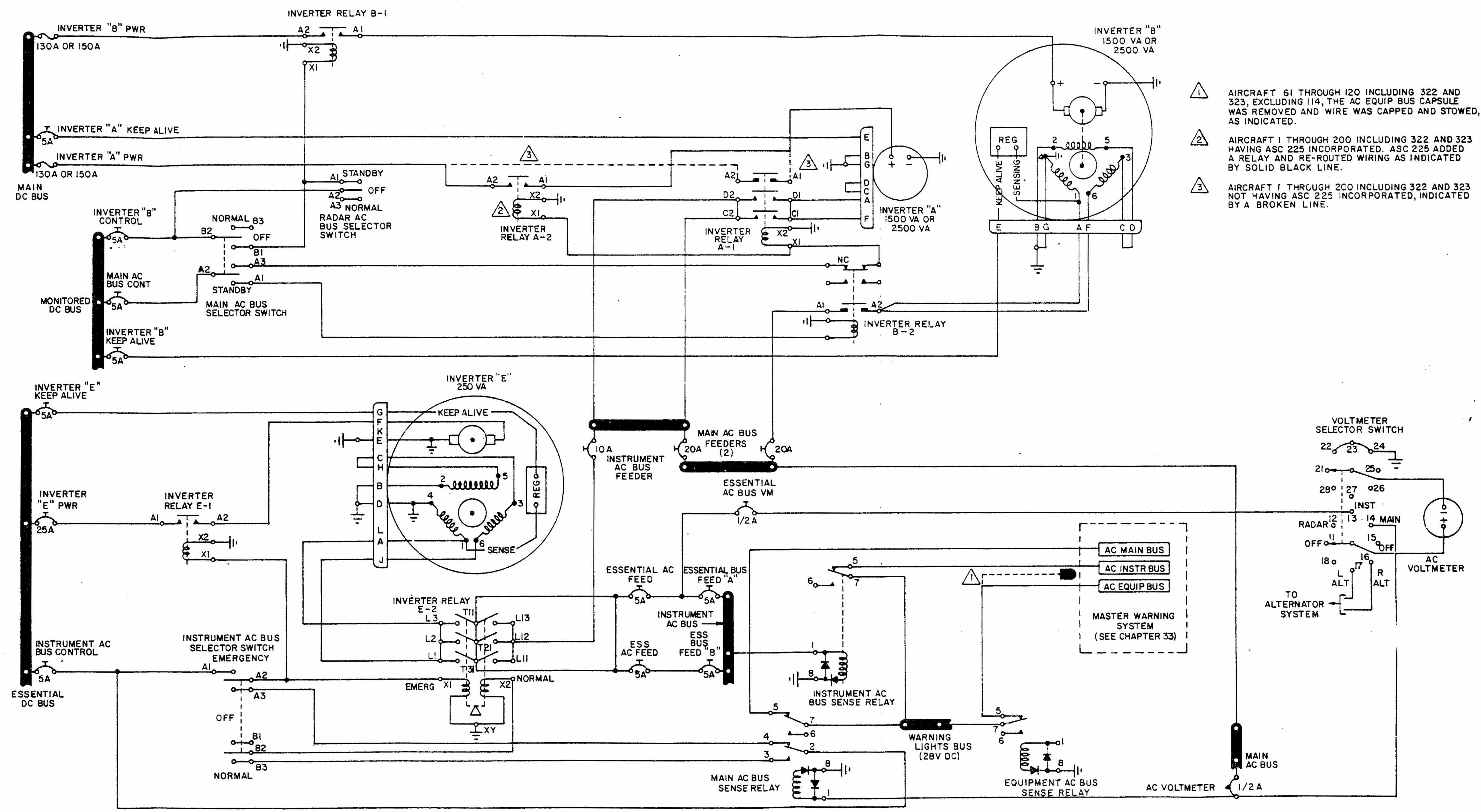
D. AC Equipment Bus

Facilities only are supplied for installation of this bus in Gulfstream delivered aircraft. No wiring is furnished. It will be wired and necessary connections made when the aircraft is furnished as predicated by the furnishing agency and customer preference.

NOTE: Other ac operated gear is added by the outfitting agency. It will be connected to appropriate ac bus as determined by the loads, etc. See AC Load Analysis for details as to how each individual aircraft is configured.



Inverter System — Block Diagram
Figure 1.

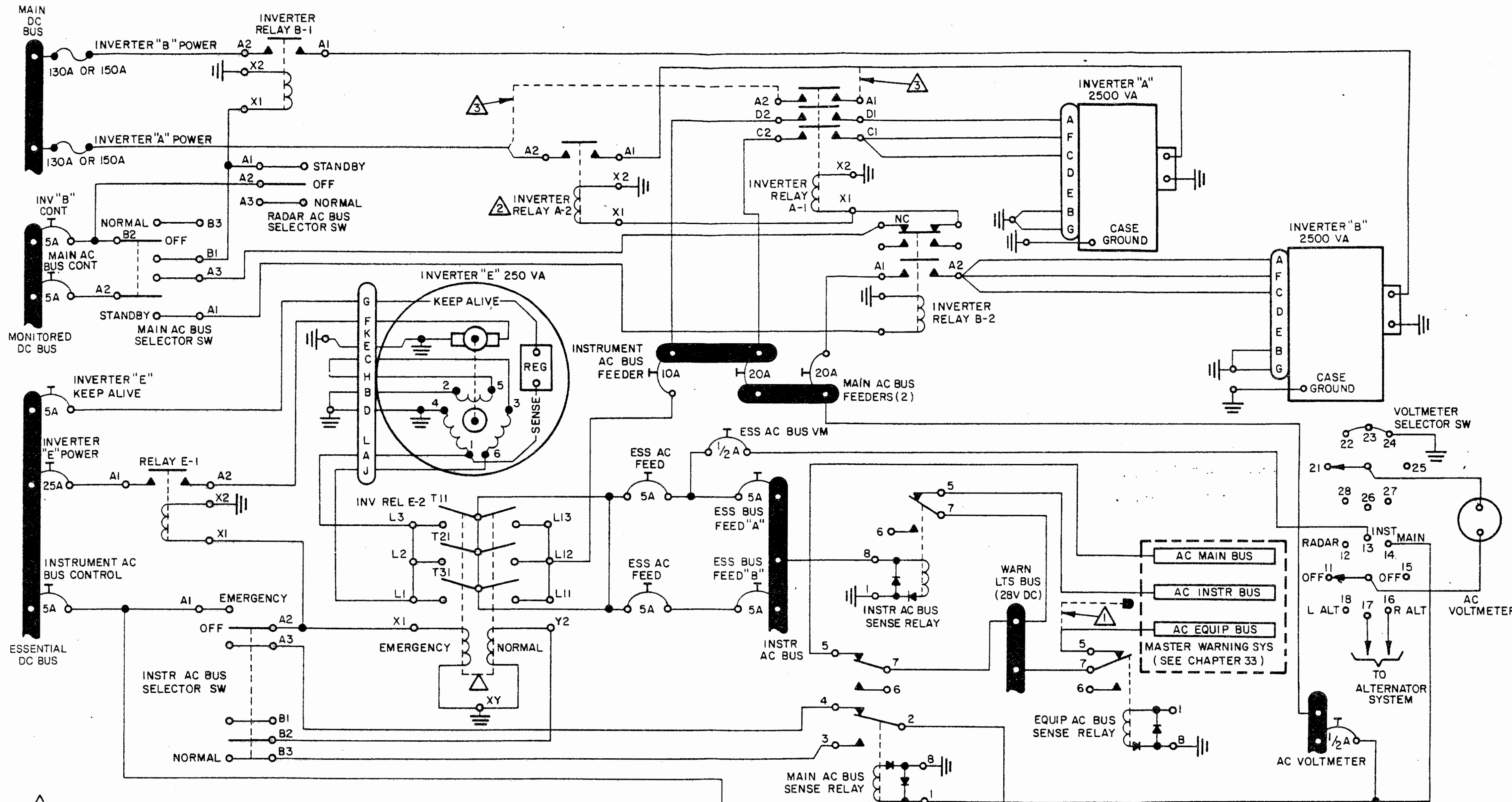


1. AIRCRAFT 61 THROUGH 120 INCLUDING 322 AND 323, EXCLUDING 114, THE AC EQUIP BUS CAPSULE WAS REMOVED AND WIRE WAS CAPPED AND STOWED, AS INDICATED.

2. AIRCRAFT 1 THROUGH 200 INCLUDING 322 AND 323 HAVING ASC 225 INCORPORATED. ASC 225 ADDED A RELAY AND RE-ROUTED WIRING AS INDICATED BY SOLID BLACK LINE.

3. AIRCRAFT 1 THROUGH 200 INCLUDING 322 AND 323 NOT HAVING ASC 225 INCORPORATED, INDICATED BY A BROKEN LINE.

Fixed Frequency AC Power Supply and Distribution System Circuit — Schematic
(1500VA and 2500VA Inverters)
Figure 2.



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3. AIRCRAFT 1 THROUGH 200 INCLUDING 322 AND 323 NOT HAVING ASC 225 INCORPORATED. INDICATED BY A BROKEN LINE.

Fixed Frequency AC Power Supply and Distribution System Circuit — Schematic
(Aircraft 1 - 200, 322 and 323 Having ASC 222)
Figure 3.

FIXED FREQUENCY AC POWER SUPPLY AND DISTRIBUTION SYSTEM — MAINTENANCE PRACTICES

1. AC Voltmeter — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to ac voltmeter in center overhead panel of cockpit as follows:

(a) Remove knobs from AC VOLTMETER selector switch and lighting rheostats.

(b) Remove center lighting panel from overhead panel.

CAUTION: DO NOT TWIST OR BEND THIS PANEL. PULL STRAIGHT OUT FROM SOCKETS.

(c) Remove hardware securing overhead panel to structure and support panel to prevent straining of plumbing and electrical lines.

(2) Disconnect, identify and insulate electrical leads to meter terminal.

(3) Remove mounting screws.

NOTE: AC voltmeter is back mounted.

(4) Remove AC voltmeter.

B. Installation

(1) Mount meter in panel.

(2) Connect electrical connectors as previously identified to terminals; ensure good electrical connections.

(3) Reinstall overhead panel to structure.

(4) Energize ESSENTIAL AC and INSTRUMENT AC buses.

(5) Select INSTRUMENT BUS on AC VOLTMETER SELECTOR.

(6) Meter should read 115V ac, if necessary adjust needle adjusting screw on face of indicator to this reading.

NOTE: If desired, alternators may be used or other inverters with proper switch selection. If alternator is used, set to actual alternator voltage using test meter and ac pinjacks in inverter relay panel. One adjustment of needle will affect all selector switch settings.

(7) De-energize electrical power.

(8) Install lighting panel.

(9) Install knobs on AC voltmeter selector switch and lighting rheostats.

(10) Energize main dc bus and check lighting panel for illumination with rheostat in overhead panel; leave rheostat OFF.

(11) De-energize electrical power.

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INVERTERS — DESCRIPTION / OPERATION

1. General

Three AC inverters are mounted on the equipment rack. The forward unit is B-inverter. The middle unit is A-inverter. The aft unit is E-inverter. A and B-inverters are identical units rated at either 1500VA or 2500VA. Aircraft 2 - 60 and 114 were originally fitted with 1500VA inverters. Aircraft 1 and 61 - 200, 322 and 323 are fitted with 2500VA inverters. E-inverter is similar to the others; however, it is smaller and rated at 250VA. Each inverter is composed of two major subassemblies - a control box, and a dc motor/ac generator base assembly. The control box contains a voltage regulator a noise filter, relays and an externally adjustable rheostat. The voltage regulator combined with the externally adjustable rheostat, controls the inverter output to 115 + 3 volts. An electronic control circuit in the unit provides for combined regulations of voltage and 400-cycle output with temperature compensation. The control circuit, in conjunction with the carbon pile voltage regulator, regulates the motor shunt field to provide automatic speed control and constant frequency output of the inverter. The A and B-inverters on Aircraft 1 - 200, 322 and 323 having ASC 222 are 2500VA solid state inverters. Voltage and frequency adjustment cannot be made on these inverters.

All inverters used on the aircraft are capable of supplying either single or three phase WYE/DELTA output. Either type of output can be attained by rearranging jumper wires within the plug. All inverters in the aircraft are wired for single phase, 115 volt, 400 cycle ac output. This has been done to avoid problems of phase unbalance (loading) which would be encountered utilizing a three phase output. Where three phase output voltage is required in the aircraft, a phase adapter is utilized. This adapter unit converts a single phase input to a three phase output.

A-inverter receives DC power from the main dc bus tie cable through the A INV PWR fuse and relay A-l (relay A-2, on Aircraft 1 - 200, 322 and 323 having ASC 225). A-inverter "keep alive" voltage (which is provided to keep the tubes in the electronic control unit heated) is derived from the main dc bus through the A INV K/A circuit breaker. The control circuit of A-inverter derives its power from the monitor dc bus through the MAIN AC BUS CONTROL circuit breaker and the MAIN AC BUS switch. When this switch is set to NORM, relay A-l (relay A-2, on Aircraft 1 - 200, 322 and 323 having ASC 225) is energized and inverter A is in operation.

B-inverter derives dc power from the main dc bus tie cable through the B INV PWR fuse and relay B-1. B-inverter "keep alive" voltage is obtained from the monitor bus through the B INV K/A circuit breaker. The control circuit of B-inverter derives its power from the monitor dc bus through the B INV CONT and MAIN AC CONT circuit breakers via the MAIN AC BUS switch. When this switch is set to STANDBY, relays B-1 and B-2 are energized. Relay B-1 brings B-inverter into operation while relay B-2 interrupts the control circuitry of A-inverter and connects B-inverter to the distribution buses.

NOTE: On Aircraft 120 - 200, 322 and 323, the K/A circuits are installed but not required by the inverters.

Aircraft 1 - 200, 322 and 323 having ASC 222, the A and B-inverters are solid state inverters and do not need a "keep alive" circuit.

On Aircraft 1 - 200, 322 and 323 having ASC 225 an additional relay is installed in the A-inverter control circuit. The additional relay separates the input and output power of the A-inverter. Separation of the input and output power is to prevent an uncontrolled A-inverter remaining on line in case of fused contacts in the single existing control relay.

2. Use of Larger Capacity Breakers in the Main and Instrument AC Feeder Circuits

The main and instrument ac feeder circuit breakers in the inverter relay box have been known to trip under certain conditions of loading. Larger capacity breakers can now be installed because it has been determined that the increase in capacity can be safely handled by the existing circuitry. The following charts provide information relative to the installation of larger capacity circuit breakers.

MAIN AC FEEDER CIRCUIT:			
Aircraft	**Inverter**	**In Use**	**Substitute**
2 - 60 and 114	1500VA	Spencer D6761-1-10	Spencer D6761-1-20
1, 61 - 200, 322 and 323	2500VA	Spencer D6761-1-20	No Change

INSTRUMENT AC FEEDER CIRCUIT			
Aircraft	**Inverter**	**In Use**	**Substitute**
2 - 60 and 114	1500VA	Spencer D6761-1-7	Spencer D6761-1-10
1, 61 - 200, 322 and 323	2500VA	Spencer D6761-10	No Change

3. **Inverter Interchangeability**

When replacing 1500VA inverters with 2500VA inverters, it is important to consider the electrical requirements. Output rating, unit rating, effects of input current, and input wire and fuse size must be evaluated. Because installation of 2500VA inverters may cause current input to reach or exceed 130 amps, the current limiter (fuse) to be used must have also amp rating and the two wires in the power feed to the B- inverter must be changed to No. 2 gage (copper). See Wiring Diagram Manual.) Aircraft 1, 61 - 200, 322 and 323 have had this change in production.

Typical steady-state input current requirements (in amps) for 1500VA and 2500VA inverters are as follows:

Inverters	No Loaded	400	600	800	1000	1200	1400	1600	1800	Output VA
1500VA	41	55	62	68	74	84	94	Overloaded		Input (Amp)
2500VA	63	74	81	90	99	108	117	126	135	

NOTE: For output circuit breaker size, see Paragraph 2 of this section.

On Aircraft 120 - 200, 322 and 323, the A and B-inverters are Leland MGE-23-400 using solid state regulation, thereby eliminating the need of the "keep alive" circuits. The K/A circuits are installed in all aircraft in the event that the MGE-23-3 inverters are used as replacements.

On aircraft having ASC 222, the A and B inverters are so lid state inverters and do not need the "keep alive" circuits.

Bendix 2500VA inverter No. 32B56-11-A is an approved alternate interchangeable with Leland 2500VA inverter.

To prevent nuisance blowing of fuses with MGE-23-400 or MGE-23-3 modified with solid state regulator, 150 amp current limiters (ANL150) may be replaced with 175 amp current limiters (ANL175).

INVERTERS — MAINTENANCE PRACTICES

1. **A, B, C and E-Inverter Voltage and Frequency — Adjustment / Check**

CAUTION: EXTERNAL DC POWER IS REQUIRED FOR THIS CHECK.

NOTE: Check is given for one inverter. Repeat for others as required.

A. Voltage/Check

(1) Gain access to Inverter(s) under radio rack, entrance compartment.

(2) Apply external dc power.

(3) Energize inverter to be checked.

(4) Allow inverter to warm up for at least 5 minutes under normal loading conditions.

(5) Using precision voltmeter (0-150V ac) and appropriate test leads, check inverter voltage at pinjacks located on inverter. Voltage must be 115 $\pm$ 3 volts. If adjustment is required, unlock jam nut on voltage adjustment rheostat and adjust accordingly. Lock jam nut when completed.

CAUTION: AIRCRAFT AC VOLTMETER IN THE OVERHEAD PANEL IS NOT RECOMMENDED AS ITS ACCURACY IS LIMITED FOR VOLTAGE ADJUSTMENT PURPOSES.

(6) Proceed with Frequency Adjustment/Check, Step B. or C. below.

(1) Frequency Adjustment/Check (Aircraft not having ASC 222)).

(a) Using precision frequency meter, check frequency of inverter using same pinjacks as were used for voltage check. Frequency must be 400 $\pm$ 8 Hz. If adjustment is required, the inverter must be shutdown and electrical power deenergized. Remove the cover from the regulator section of the inverter to expose the frequency adjustment rheostat, located on the printed circuit board. Using small insulated screwdriver adjust frequency after reenergizing electrical power and inverter until the 400 $\pm$ 8 Hz required is obtained. Recheck voltage after adjustment.

(b) De-energize electrical power and Inverter, and tighten jam nut.

(c) Inspect for foreign objects and install regulator section cover.

(d) Inspect area for foreign objects, security of all attachments.

(e) Reinstall any components removed for access and secure access covers.

B. Frequency/Check (Aircraft having ASC 222)

(1) Connect aircraft batteries and apply external dc power.

(2) Place BATT switch to NORM.

(3) Place Monitor BUS TEST switch on.

(4) Ensure MAIN AC BUS selector switch is OFF and ESSENTIAL AC BUS switch is in NORMAL

(5) Ensure MAIN AC BUS warning light is on and ESSENTIAL AC BUS and INSTRUMENT BUS lights are out. (Master Caution Panel).

(6) Place MAIN AC BUS selector switch to MAIN. Check that MAIN AC, ESSENTIAL AC and INSTRUMENT

(7) AC Warning lights in the Master Caution Panel are all OUT.

(8) Turn MAIN AC BUS selector switch off.

(9) Ensure main ac light is on and E-inverter comes on line.

(10) Ensure essential and instrument lights are out.

(11) Place MAIN AC BUS selector switch to STANDBY.

(12) Ensure MAIN AC light goes out. Check B-inverter for 115 $\pm$ 5V ac and 400 $\pm$ 5 Hz.

(13) Place MAIN AC BUS and ESSENTIAL AC BUS selector switches off (MAIN, ESSENTIAL and INSTRUMENT ac warning lights should be on.

(14) Place MONITOR BUS TEST switch off.

(15) Place BATT switch off.

2. AC Sensing Relays — Replacement

If replacement of one of the ac sensing relays is necessary, it is imperative that it be replaced with the same type removed, namely - Leach 937-85H. The coil of this relay operates on 115 volt, 400 cycle ac and is ad justed to close the contacts at 105 volts or greater and opening the contacts at 70 volts or less.

3. A, B and C-Inverter Fuse — Removal / Installation

CAUTION: ENSURE THAT ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT, DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to inverter fuse panel (on aft side of Fuselage Stations 193 bulkhead; right side).

(2) Locate inverter fuses (current limiters - boltdown type).

(3) Loosen holddown nuts at each end of fuse and slide fuse out from under bus bar, do not remove nuts.

(4) Note fuse value.

B. Installation

(1) Install fuse (same value as removed one).

(2) Ensure that forked end of fuses are securely under bus bar.

(3) Tighten down holddown nuts and torque to 65 to 95 inch-pounds.

(a) Using torque wrench, tighten stop-nut until just short of making contact with terminal or bus bar (note torque to move nut).

(b) Add this torque to value that stud requires (65 to 95 inch-pounds) and tighten to final torque.

(4) Inspect area for foreign objects and security.

(5) Close access previously opened.

(6) Operate inverters involved and check operation using ac voltmeter in cockpit overhead panel with pro per AC VOLTMETER SELECTOR switch position.

4. A, B, C and E-Inverters — Removal / Installation

NOTE: The C-inverter was installed in Aircraft 2 - 60 and 114 in production.

A. Removal

NOTE: Removal procedures are common to all inverters except where noted.

(1) Gain access to inverter involved by removing the following access panels under the radio rack, entrance compartment:

(a) A or E-inverter - aft lower panel (Fuselage Stations 169-191)

(b) B or C-inverter - forward lower panel (Fuselage Station 133-169)

(2) For E or A-inverter, remove voltage regulators after marking for reinstallation.

(3) For B or C-inverter, remove two electrical connectors to pressurization dump solenoids, and also any pneumatic lines to pressurization components under forward section of radio rack. Cap all lines.

(4) Disconnect electrical connections to inverter involved. Identify leads where applicable. Disconnect bonding straps.

(5) To remove A, B or C-inverter remove four mounting bolts and rubber isolators between inverter and mounts. Inverter will remain in position. Support inverter, lift up to unseat and remove inverter.

NOTE: To remove A or B-inverter on Aircraft 1 - 200, 322 and 323 having ASC 222 (solid state inverters), remove 2 nuts securing inverter to inverter mounts. To remove, pull inverter out.

CAUTION: SUPPORT E-INVERTER BEFORE LOOSENING MOUNTING BOLTS, AS IT WILL DROP IF NOT SUPPORTED.

(6) If E-inverter is being removed, support E-inverter and loosen mounting hardware as it is mounted directly to structure.

B. Installation

(1) Install A, B or C-inverter using hardware previously removed. Tighten mounting bolts until rubber isolators compress.

NOTE: To install A or B-inverter on Aircraft 1 - 200, 322 and 323 having ASC 222, engage pins mounted on rear of inverter into holes on rear fitting assembly and slide inverter front fitting assembly onto front mounting pins. Install 2 nuts and tighten until inverter is secure to structure.

(2) Install E-inverter support inverter in position and tighten down hardware until it is secured to structure.

(3) Connect leads (as applicable) as previously identified, and/or reconnect electrical plug. Ensure good electrical connection.

(4) Connect bonding strap ensuring good connection.

(5) Energize following buses (as applicable):

(a) A, B or C-inverter - main, essential and monitor buses.

(b) E-inverter - essential dc bus.

NOTE: External power is recommended.

(6) Check inverter for operation, noise and vibration.

(7) Perform A, B, C and E-Inverter Voltage and Frequency — Adjustment / Check, this Section.

NOTE: Inverter voltage and frequency adjustments cannot be made on Aircraft 1 - 200, 322 and 323 having ASC 222 (solid state inverters).

(8) Install voltage regulators previously identified, or reconnect electrical plugs and pneumatic lines disconnected previously in Steps A(2) or A(3) above.

5. Brush Inspection Period

NOTE: Does not apply to A and B solid state inverters (Aircraft 1 - 200, 322 and 323 having ASC 222).

See Chapter 5.

6. Brush Wear Limits

NOTE: Does not apply to A and B solid state inverters (Aircraft 1 - 200, 322 and 323 having ASC 222).

All inverter brushes have a wear limit line scribed on the outside face of the brush. The minimum permissible length as shown by these lines corresponds to the limits noted in the table below. If a brush is worn beyond the limits given in the table below, the inverter must be removed and sent to a certified repair station, or returned to the manufacturer for installation and seating of the brushes.

INVERTERS "A" AND "B"			INVERTER "E"
LELAND INVERTERS			
AC Brushes	**15OOVA**	**25OOVA**	
Straight	SA-9238-7/16"	SA-9238-7/16"	SA-I 0215-5/16"
Angle	SA-9236-7/16"	SA-9236-7/16"	SA-10214-5/16"
DC Brushes	A-10303-9/16"	A-10372-9/16"	A-10007-7/16"
BENDIX INVERTERS			
	25OOVA	**NORMAL LENGTH**	**ALLOWABLE WEAR**
AC Brushes	1102025	9/16"	1/4"
DC Brushes	1540124	1.047	0.360

VARIABLE FREQUENCY, THREE PHASE, AC POWER SUPPLY AND DISTRIBUTION SYSTEM -- DESCRIPTION / OPERATION

1. Description

(See Figure 1)

There are two alternators on the aircraft, which are mounted on and driven by their respective accessory gearbox. Each alternator has two functions:

- To supply three phase ac power to its related engine and prop deicing system.
- To supply three phase ac power to part of the cockpit windshield heat system. Both left and right alternator systems are identical to one another, but completely independent of each other. Since it is an ungrounded system, all voltage detecting is done line to line.

The main dc bus provides field excitations for the alternators through the L ALT FLD PWR and R ALT FIELD PWR circuit breakers. Control circuit power is taken from the main dc bus through the L ALT CONT and R ALT CONT circuit breakers. Control over each alternator system is by means of a three position (on-off-momentary reset) switch. These are the two outboard switches under the gang bar located on the upper overhead panel. (See Figure 1) Associated with each system is an AC GEN OFF warning 1light which is part of the master warning system. The alternators are ram air cooled and have a thermal switch in the air outlet duct which controls an AC GEN HOT warning light (also part of the master warning system).

Each alternator system is provided with automatic voltage regulating, and automatic tripping, which occurs when there is an unbalance condition between phases, and overvoltage condition, or an undervoltage condition.

2. Operation

(See Figure 2 and Figure 3)

With the alternator driven by the gearbox, and the alternator switch set to OFF, the field bias resistor is inserted into the field circuit so that the alternator field is effectively not excited. However, a small amount of current does flow in the alternator field to prevent polarity reversals. Setting the alternator switch to ON, energizes the field power relay by means of a series circuit through the contacts of the reset relay and the current balance relay. The field power relay shorts out the field bias resistor, allowing the alternator field to be excited through the voltage regulator carbon pile. However, since the undervoltage relays are set to de-energize when there is a line-to-line voltage of less than 108-131 volts, the current balance relay is immediately energized by the undervoltage relays, which de-energizes the field power relay. This necessitates resetting the alternator. This is accomplished by depressing the alternator control switch to its momentary RESET position, which energizes the reset relay. Thereafter the following sequence takes place. By energizing the reset relay, a circuit is completed which allows the anti-cycle relay to energize. This relay in turn removes the ground from the reset relay coil, which would normally allow the reset relay to de-energize. However during the time the reset relay was energized, a time delay capacitor connected across the coil of the reset relay was being charged. As the reset relay ground is removed, the capacitor discharges through the coil of the reset relay keeping it energized long enough to enable the alternator output voltage to build up, unimpeded by the undervoltage, overvoltage, and current balance relays. When the output voltage exceeds 108-121 volts, the undervoltage relays are energized, completing part of the overvoltage relay circuit, and breaking the undervoltage circuit to the current balance relay.

When the output voltage builds up to approximately normal output, the field power transformer-rectifier produces the voltage regulator control current which is proportional to the alternator output voltage. This control current varies the resistance of the carbon pile, thereby varying the field current to maintain the output voltage at 212 volts, line-to-line. Releasing the alternator control switch removes holding power from the anti-cycle relay and it deenergizes. The system is now ready for the next operation.

Resetting the Alternator

Resetting is necessary because of the undervoltage protection. Any time the alternator output ceases, the undervoltage system detects it and locks the alternator field power out. To set the alternator back on the line,

the undervoltage protection must be bypassed momentarily, to establish alternator output. This is a function of the reset system.

CAUTION: RESETTING PROCEDURE MUST BE PERFORMED TO BRING AN ALTERNATOR ON THE LINE AT ANY TIME.

Resetting the alternator involves the following procedure for each alternator:

- Place the alternator control switch for that side in the ON position. All loads should be off. The warning light should come on (AC GEN OFF), for that alternator, as it is not on the line.

  NOTE: If voltage appears on the output bus, place switch to OFF and identify defective component/wiring in reset relay circuit.

- Set the AC VOLT SELECTOR switch to the appropriate alternator position so that alternator output can be monitored.
- Engine must be running above 10,000 rpm.
- Push alternator control switch momentarily to RESET. Observe that voltage appears on the voltmeter. AC GEN OFF light for that side should go out. Voltage on meter should read 212 volts.
- Release control switch to ON position.

Anti-Cycle Protection

To prevent the alternator being reset on a fault, an anti-cycle system is incorporated in each protection package. This prevents manually holding the system in operation. Only one reset cycle occurs, even though the button is held down. This is part of the undervoltage bypass cycle. Timing devices allow this bypass long enough to establish the alternator above 108-131 volts. If it is not above 108-131 volts by the time the reset cycle is completed, the undervoltage system removes the alternator from the line. The time involved is a matter of a fraction of a second.

Phase unbalance protection (7.8 amp unbalance at full load)

The current balance relay compares the current flowing through all three phases to each other. If a current unbalance between phases is detected, the current balance relay automatically energizes itself and trips the alternator.

Undervoltage protection (108-131 volts)

If the alternator voltage drops to 108-131 volts, the appropriate undervoltage relay is de-energized. This in turn energizes the current balance relay which trips the alternator.

Overvoltage protection (See paragraph 3 for setting.)

If the alternator voltage increases to overvoltage limitations or higher, the overvoltage sensing transformer-rectifier produces a current sufficient to energize the overvoltage relay. The overvoltage relay then energizes the current balance relay to trip the alternator.

NOTE: If any one of the above malfunctions occur, the alternator will be tripped and the appropriate AC GEN OFF warning light should light up. Upon resetting this light should go out (if fault has been cleared).

3. Alternator Voltage Settings

Aircraft 1 - 85 alternator systems were set for an output of 200 volts under no load condition. With this output voltage setting, when loads cycle on and off, the system responds to yield voltages ranging from 194 to 204 volts. Some operators have experienced deicing system performance described as marginal using this voltage setting.

The manufacturer therefore recommends, but does not make mandatory, raising the alternator output voltage, under no load conditions, to 212 volts. This adjustment will result in a operating spread of 206 to 216 volts.

The recommended limitations for system operation with either a 200 or 212 volt output voltage setting are listed below.

Regulated Voltage (no load)	212 Volts	200 Volts
Overvoltage Limits	223 - 240 Volts	220 - 231 Volts
Undervoltage Limits	108 - 131 Volts	108 - 131 Volts
Phase Unbalance Limits	7.8 Ampere Difference at Full Load	7.8 Ampere Difference at Full Load
Voltage Spread Under Load	206 - 216 Volts	194 - 204 Volts

4. **Location of Components**

(See Figure 4 thru Figure 6)

A. Alternators - one on each engine gearbox.

B. Alternator sensing package (alternator relay box) - electronics compartment. Contains the following:

- Overvoltage sensing relays and dropping resistors.
- Undervoltage relays and dropping resistors.
- Overvoltage transformer/rectifiers.
- Time delay capacitors.
- Reset relays.
- Anti-cycle relays.

C. The mid relay box - under FLR-8 contains:

- Field power relays.
- Field power biasing resistors.

D. The inverter relay box contains:

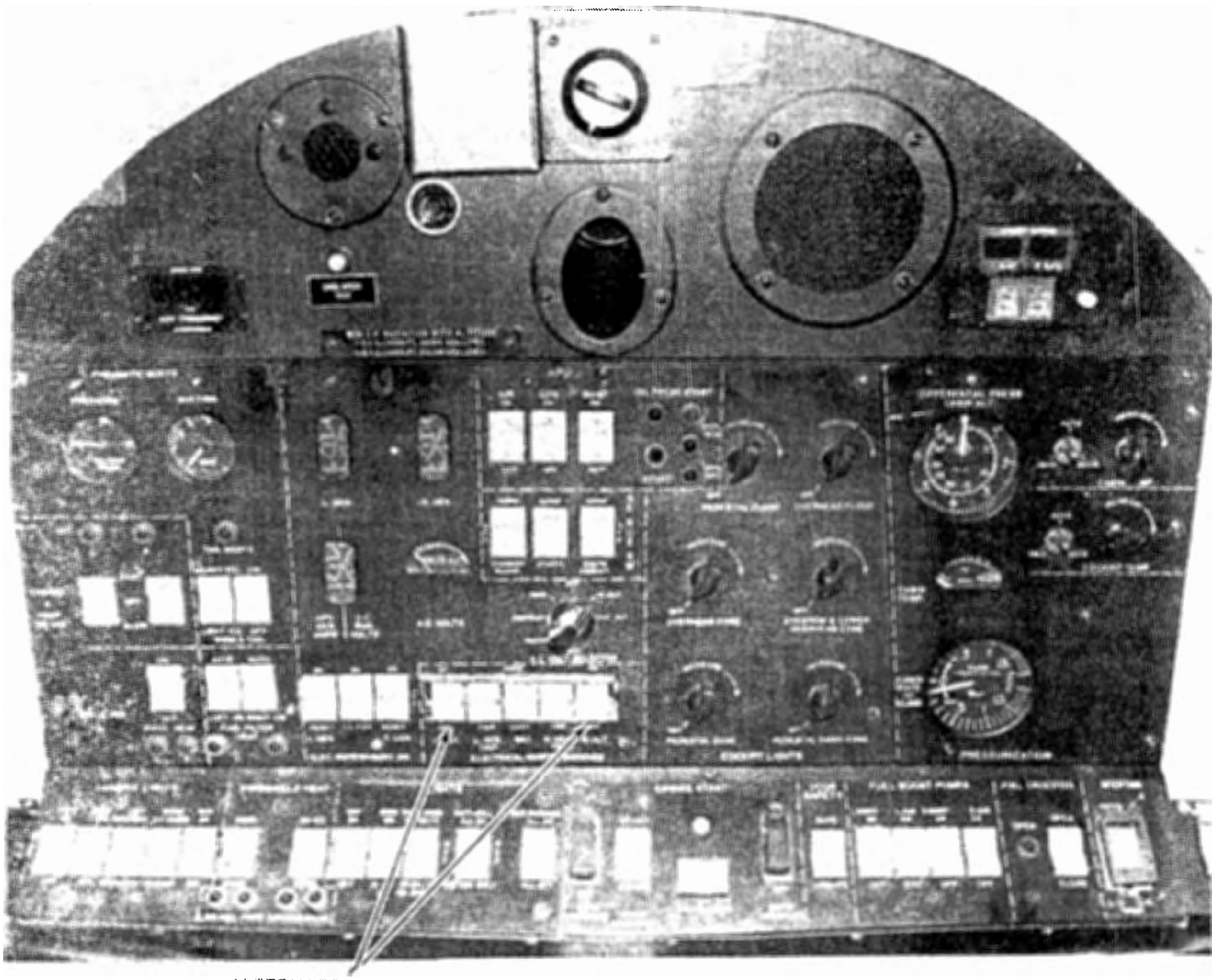
- Remote trimmers.
- Voltage adjustment rheostats.
- Alternator pinjacks.

E. Voltage regulators - below radio rack, Fuselage Station 169-193, right side.(See Figure 4)

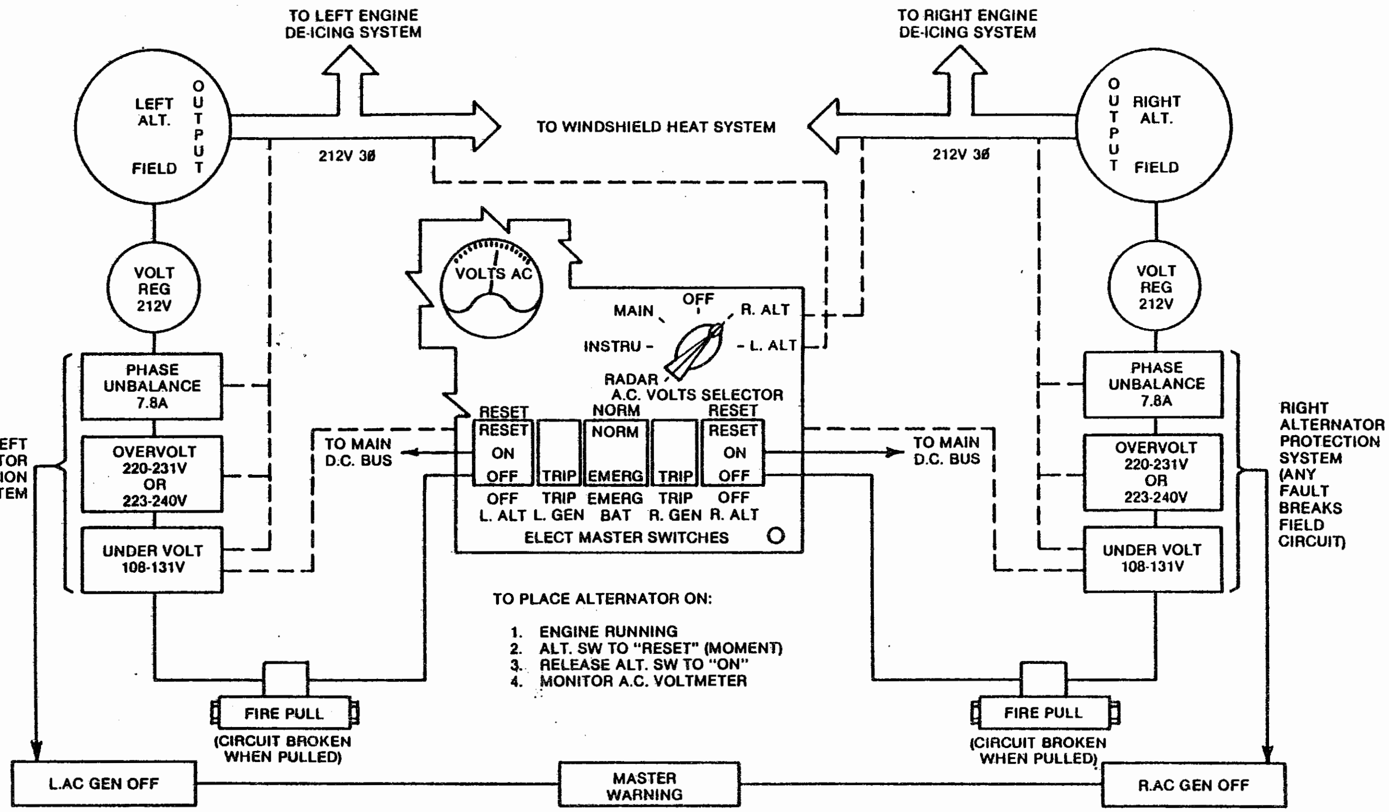
F. Field power transformer/rectifiers - under FLR-12.

G. AC voltmeter circuit breakers - two in each nacelle relay box.(See Figure 5)

H. Current balance relays - one in each nacelle relay box.(See Figure 5)



Alternator Control Switch Location
Figure 1.



Alternator System Block Diagram
Figure 2.

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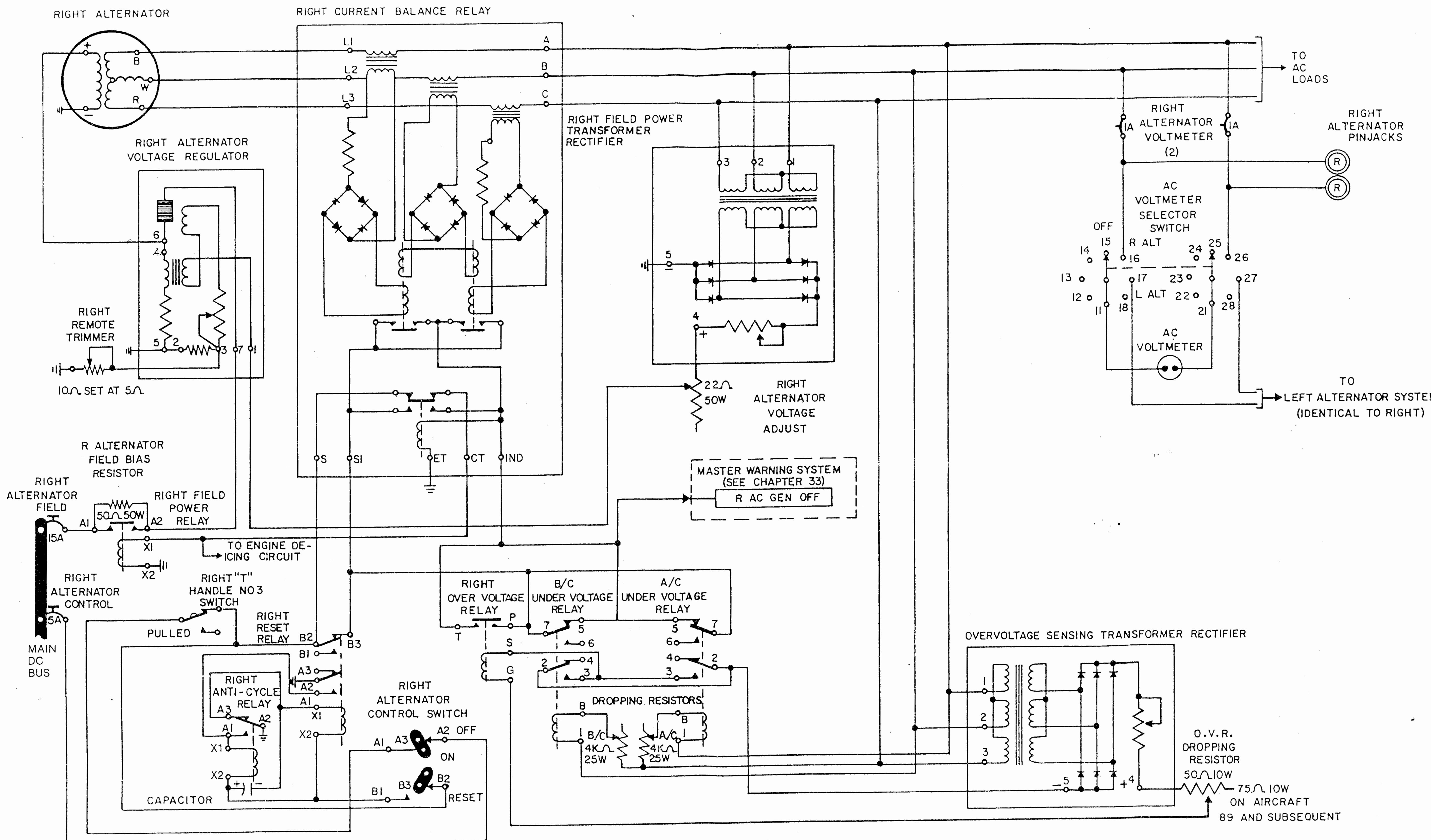
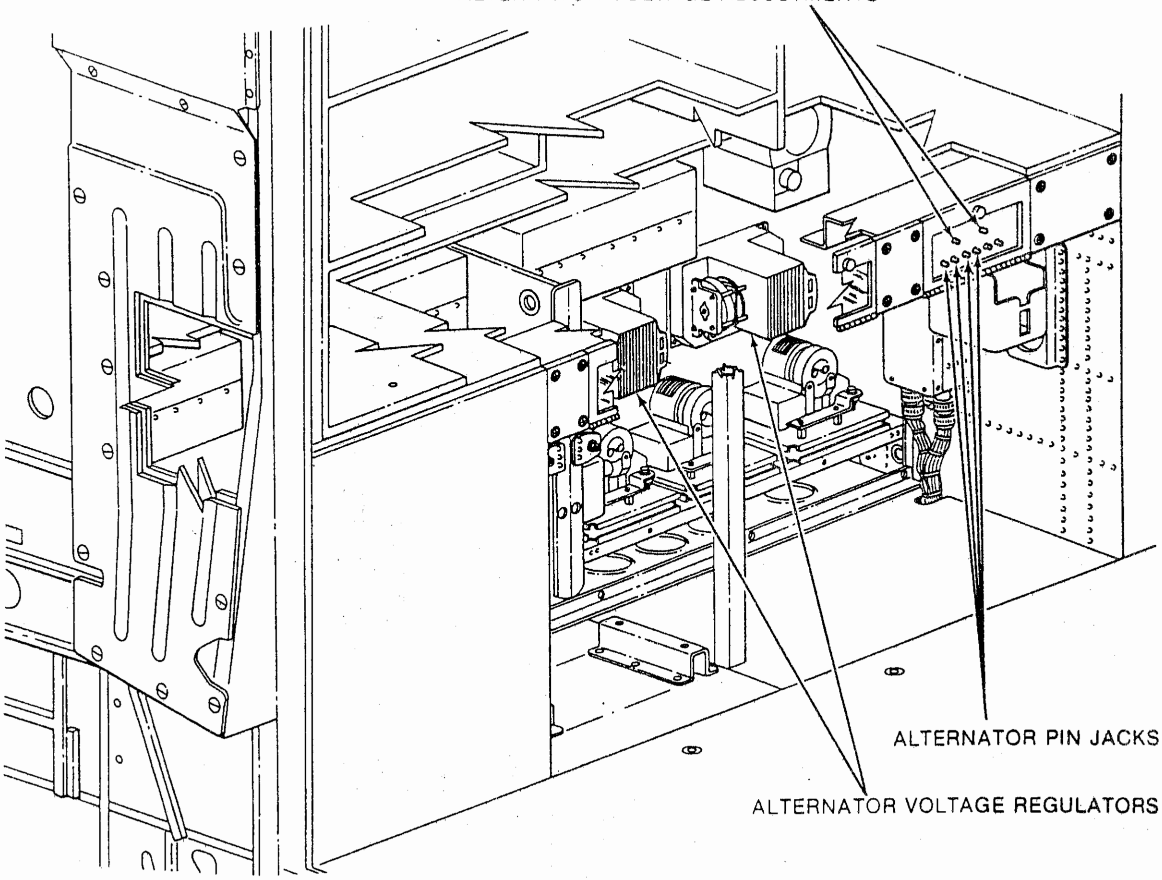
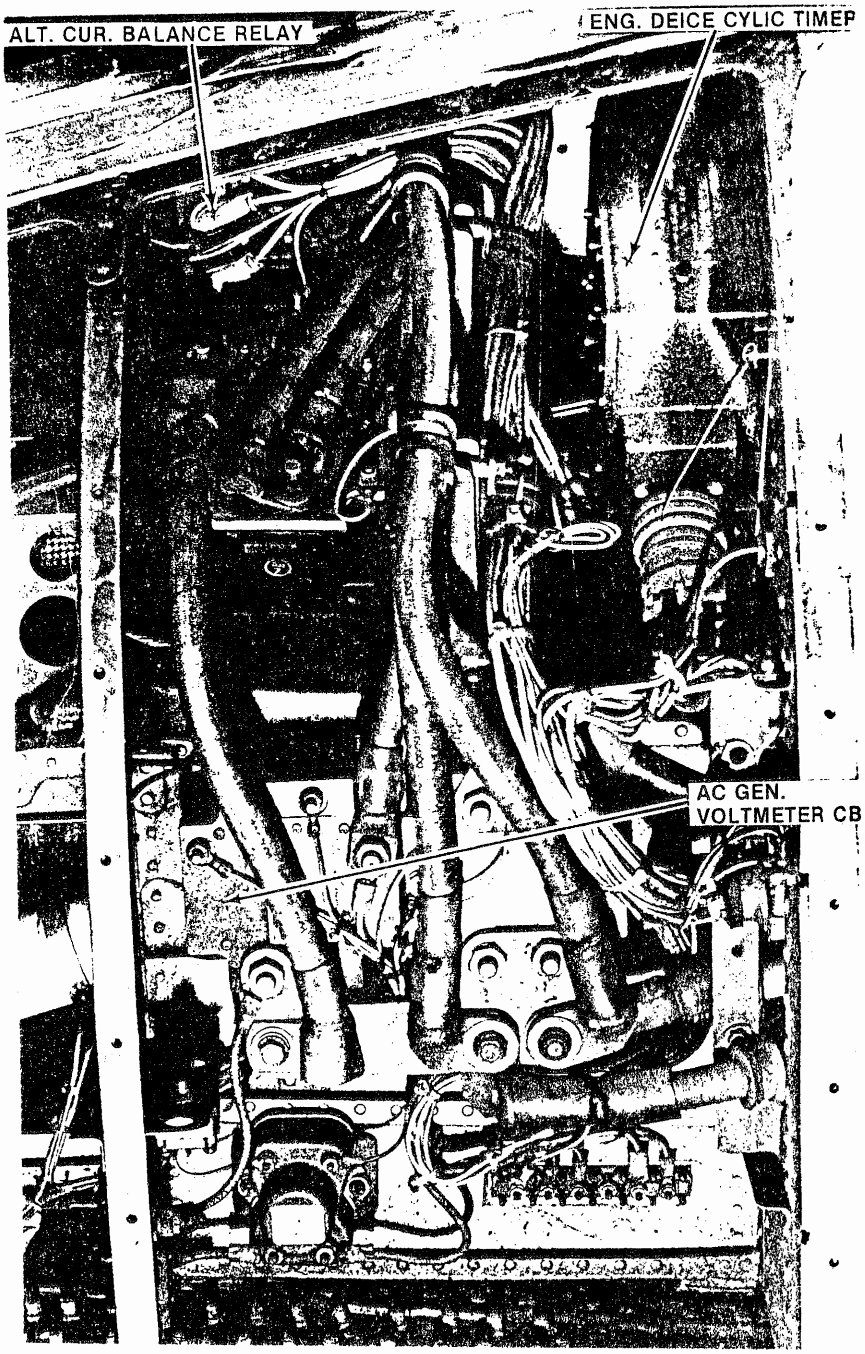


Variable Frequency, Three-Phase, AC Power Supply Circuit Schematic
Figure 3

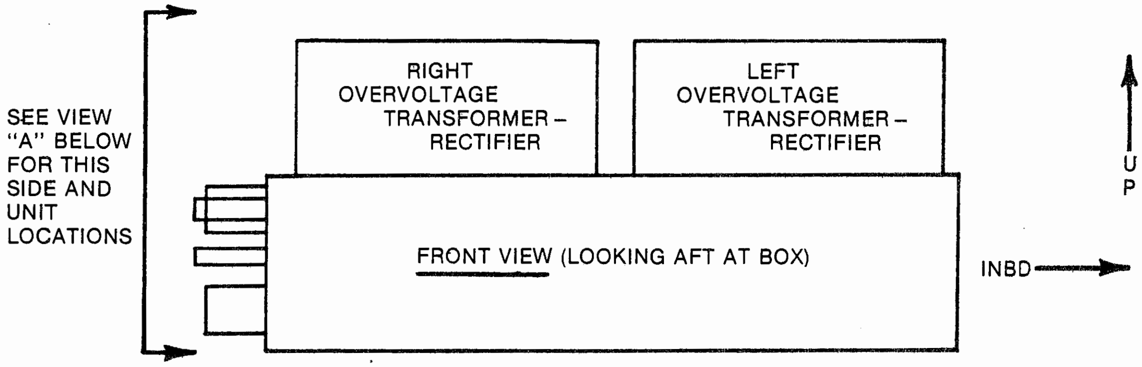


Alternator Voltage Regulator and Voltage Adjustment Locations
Figure 4.



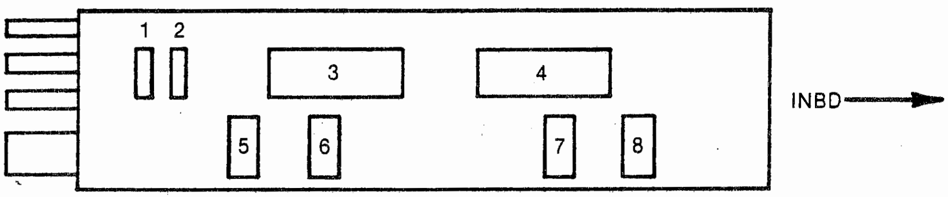
Alternator Equipment Location In Left Nacelle Relay Box
Figure 5.

ALTERNATOR SENSING PACKAGE

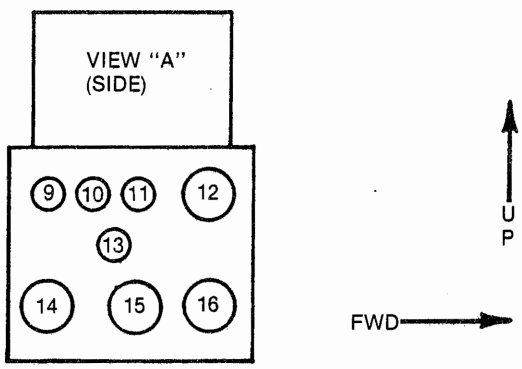


1 – LEFT OVERVOLTAGE DROPPING RESISTOR
2 – RIGHT OVERVOLTAGE DROPPING RESISTOR
3 – LEFT OVERVOLTAGE RELAY
4 – RIGHT OVERVOLTAGE RELAY
5 – LEFT A/C UNDERVOLTAGE RESISTOR
6 – LEFT B/C UNDERVOLTAGE RESISTOR
7 – RIGHT A/C UNDERVOLTAGE RESISTOR
8 – RIGHT B/C UNDERVOLTAGE RESISTOR

VIEW LOOKING INSIDE BOX FROM TOP (WITH TOP REMOVED)



9 – LEFT ANTI-CYCLE RELAY
10 – LEFT RESET RELAY
11 – RIGHT ANTI-CYCLE RELAY
12 – RIGHT UNDERVOLTAGE A/C RELAY
13 – RIGHT RESET RELAY
14 – LEFT UNDERVOLTAGE A/C RELAY
15 – LEFT UNDERVOLTAGE B/C RELAY
16 – RIGHT UNDERVOLTAGE B/C RELAY



NOTE: NORMAL LOCATION OF THE ALTERNATOR SENSING PACKAGE IS IN AFT SECTION OF RADIO RACK ON FORWARD SIDE OF FUSELAGE STATION 193 BULKHEAD, HOWEVER ITS LOCATION MAY VARY WITH FURNISHING AGENCY INSTALLATIONS.

Alternator Sensing Package, Equipment Location
Figure 6.

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ALTERNATORS — DESCRIPTION / OPERATION

1. **General**

An 11.0KvA, three-phase alternator is mounted on and is driven by the accessory gearbox in each nacelle. Each alternator is wye-wound, with neutral not grounded, so that all voltage detecting is done line-to-line. The main dc power system provides the dc voltage for field excitation through the carbon pile of the voltage regulator. The voltage regulator varies the field current in response to the output of the field transformer/rectifier, so that the output voltage of the alternator is maintained at 200 or 212 volts. The frequency of the output voltage is in the range of 206 to 413 cps. When the alternator is turned off, or is automatically tripped, a field bias resistor is inserted into the field circuit by the field power relay. This causes the field current to be reduced to almost nothing. A small amount of current still flows in the field circuit to prevent polarity reversals.

Each alternator is supplied with ram air for cooling. An inlet duct on the upper right side of the nacelle is connected to alternator housing by a blast tube. The exhaust tube for the cooling system vents overboard. A thermal switch is installed in each exhaust tube and functions to light the AC GEN HOT warning light when abnormal temperatures exist at alternator. Each switch is set to close at a temperature of 300°F (149°C) ± 10°F.

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ALTERNATORS — MAINTENANCE PRACTICES

1. Alternator — Removal / Installation

A. Removal

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE. IF THE AIRCRAFTS ELECTRICAL SYSTEM IS TO BE ENERGIZED DURING THE TIME THAT THE ALTERNATOR IS REMOVED FROM THE AIRCRAFT, DISENGAGE THE APPROPRIATE ALT. FLD. PWR. CIRCUIT BREAKER. TAPE AND STOW POSITIVE LEAD.

(1) Gain access to alternator by removing gearbox access covers.

(2) Remove terminal block cover, disconnect, identify, and insulate alternator leads.

(3) Disconnect cooling air inlet and exhaust ports.

(4) Mark alternator mounting plate for alignment on reinstallation.

(5) Depress nut locking device and loosen link assembly bolt nut.

(6) Support alternator and remove quick-release link assembly.

(7) Carefully remove alternator from mounting pad and retain quill shaft.

B. Installation

NOTE: If new or replacement alternator is to be installed remove cooling air inlet and exhaust shroud from old alternator and install on new unit.

(1) Apply engine/gearbox oil to receiving flanges of alternator and gearbox adapter.

(2) Install quill shaft and install alternator to gearbox.

(3) Install link assembly.

(4) Apply engine/gearbox oil to link assembly bolt threads.

(5) Tighten link assembly bolt nut as follows:

(a) Tighten nut initially to 100 inch pounds.

(b) Loosen nut and tighten twice more to ensure proper link assembly seating, final torque to 100 inch pounds.

(c) Lock link nut with spring-loaded locking device.

(6) Connect cooling air inlet and exhaust ducts, ensure connections are tight.

(7) Connect electrical leads to alternator terminal block as previously identified; torque nuts at 12 inch pounds to ensure good electrical connections.

(8) Install terminal block cover and safety.

(9) Perform engine run and check for oil leaks.

CAUTION: DO NOT EXCEED IDLE RPM WITH ACCESS COVERS REMOVED.

(10) Inspect area for foreign objects, security of all attachments.

(11) Install access covers on gearbox.

(12) Perform Alternator Voltage - Adjustment / Check, this Section.

2. Alternator Brush — Inspection

NOTE: If the alternator is Installed on the gearbox, It Is permissible to check the accessible brushes for length and condition and visually check the remainder by comparison (See Figure 201) However, If the visual Inspection of the remainder of the brushes Indicates any variation from the brushes actually removed, the alternator should be removed and each set of brushes given detailed inspection. If the alternator is not installed on the gearbox or is being removed for some other purpose, check all brushes.

A. Gain access to alternator by removing accessory gearbox access covers (if alternator Is installed on gearbox).

B. Remove brush cover band.

C. inspect accessible brushes for length, condition and freedom of movement. (See Note above.) Minimum length is 0.437 inch measured on the long edge. Reinstall brushes.

CAUTION: IF BRUSHES ARE WORN BEYOND LIMITS, THE ALTERNATOR MUST BE REMOVED AND BRUSHES REPLACE BY A CERTIFIED FACILITY OR THE MANUFACTURER.

(1) Reinstall brush cover band.

(2) Inspect for foreign objects then close access.

(3) Perform engine run and check for normal alternator operation.

3. Alternator Field Transformer-Rectifier — Removal / Installation

A. Removal

(1) Gain access to alternator field transformer-rectifier by removing FLR-12 in forward cabin (left alternator T-R is forward, right alternator T-R is aft).

(2) Disconnect, identify and insulate electrical leads from transformer-rectifier terminal block.

(3) Remove hardware and remove transformer-rectifier.

B. Installation

(1) Install transformer-rectifier.

(2) Install electrical leads as previously identified; ensure good electrical connections.

(3) Perform check of alternator involved on engine run to ensure proper operation.

(4) Inspect area for foreign objects, security of all attachments.

(5) Reinstall FLR-12.

4. Alternator Transformer-Rectifier — Removal / Installation

A. Removal

(1) Gain access to alternator overvoltage transformer-rectifier on top of alternator relay box.

NOTE: Normal location of the RELAY BOX is in aft section of radio rack on forward side of Fuselage Station 193, however its location may vary with furnishing agency installations.

(2) Disconnect, identify, and insulate electrical leads from transformer-rectifier terminal block.

(3) Remove mounting hardware and remove transformer-rectifier.

B. Installation

(1) Mount transformer-rectifier.

(2) Connect electrical leads as previously identified, ensure good electrical connections.

(3) Perform Alternator Voltage — Adjustment / Checks, this Section.

5. Alternator Voltage — Adjustment / Check

NOTE: The below checks involve operation of engines, alternators, windshield heat, and engine and prop deicing systems. These checks are intended to be performed in sequence.

A. During accomplishment of the following checks observe that:

(1) The appropriate AC GEN light on the master caution panel comes on when the control switch is selected ON but there is no output voltage on the bus.

(2) The light is out when the control switch is selected OFF.

(3) The light is out when the control switch is selected ON and output voltage is present on the ac bus.

B. Overvoltage Check

(1) Gain access to the alternator voltage regulator voltage adjustment rheostat, located in the face of the inverter relay panel adjacent to the alternator test pinjacks, radio rack section, entrance compartment.

(2) Jack in precision ac voltmeter to appropriate alternator pinjacks (0-300V ac).

(3) Start and run engines using normal procedures. RPM should be 11,000.

(4) Place appropriate alternator control switch to ON and check for output voltage. (If voltage is on the output bus, place switch in the OFF position and identify defective component/wiring in reset relay circuit.) If no voltage appears at the output bus, set switch to RESET then release to ON. Ensure voltage is indicated on the ac voltmeter by selecting ac voltmeter selector switch to appropriate alternator. Also, observe that alternator warning light for that alternator in the master warning panels is not on.

(5) Prop and engine deicing and windshield heat switches must be off at this time.

(6) Allow regulators to stabilize and warm up for about 20 minutes.

(7) Using small insulated screwdriver and observing test meter, turn voltage adjustment rheostat for alternator involved clockwise slowly.

(8) Alternator should drop off the line (meter to 0) between 230 and 240 volts. Alternator light for that unit should come on in the master warning panels.

(9) Move adjustment counterclockwise to decrease setting.

(10) Reset alternator and ensure that it comes back on the line.

(11) Proceed with voltage adjustment.

C. Voltage Adjustment

(1) With no load, adjust voltage rheostat with insulated screwdriver to 212V ac voltmeter.

(2) Turn on PROP AND ENGINE DEICING switch and WINDSHIELD HEAT switch and observe voltmeter. Voltage spread under load should be approximately 206 to 218V ac.

(3) Turn off PROP AND ENGINE DEICING switch and WINDSHIELD HEAT switch.

(4) Proceed with undervoltage check.

D. Undervoltage Check

(1) With no load, while observing voltmeter, retard power lever slowly for alternator system under test. Voltage will decrease proportionately.

(2) With alternator still on, shut down engine.

(3) Voltage will decrease with engine run down until the point where the voltage suddenly will drop to 0 when undervoltage system cuts off alternator. Cutoff will be approximately 108 to 131 volts.

(4) Remove voltmeter.

NOTE: Perform the above procedures for other alternator if both are to be checked / adjusted.

(5) Inspect for presence of foreign objects then close access panels.

6. Alternator Voltage Regulator — Removal / Installation

A. Removal

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

(1) Gain access to alternator voltage regulator (under radio rack, just forward of Fuselage Station 193, entrance compartment, right side), remove aft access cover below radio rack.

(2) Remove and identify carbon piles from dc regulators by holding spring clips open, and remove piles.

(3) Remove alternator terminal block cover.

(4) Remove, identify, and insulate electrical leads.

(5) Remove attaching hardware and remove regulator.

B. Installation

NOTE: Burnish underside of mounting holes to ensure good ground to structure.

(1) Mount regulator with attaching hardware.

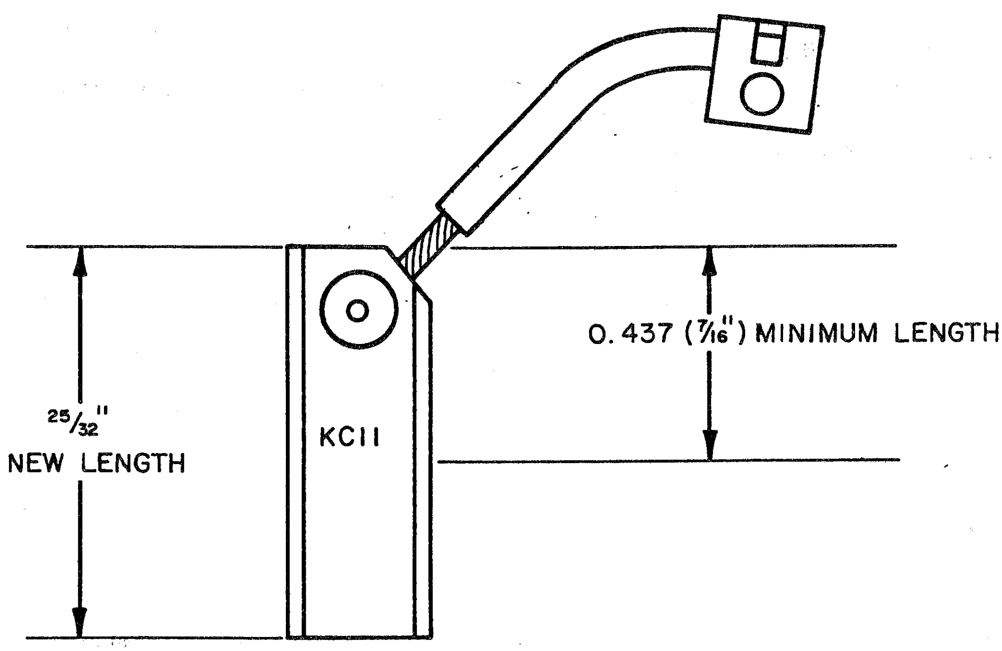
(2) Connect electrical leads as previously identified, ensure good electrical connections.

(3) Install dc carbon piles as previously identified.

(4) Perform engine run and check dc regulators operation.

(5) Operate APU generator and check operation (if pile was removed).

(6) Perform Alternator Voltage — Adjustment / Check, this Section.



Alternator Brush
Figure 201.

AC VOLTAGE REGULATORS — DESCRIPTION / OPERATION

1. **Description and Operation**

Carbon pile voltage regulators, one for each alternator, are mounted below the radio rack above the dc voltage regulators. The left alternator ac voltage regulator is the forward unit (located below A-inverter), the right alternator ac voltage regulator is the aft unit (located below E-inverter).

The alternator field windings excitation current flows through the carbon pile. The field power transformer-rectifier output controls the operation of the carbon pile solenoid, which varies the resistance of the carbon pile. The alternator field voltage is, therefore, determined by the alternator output. A transformer within the voltage regulator has its primary windings in series with the carbon pile solenoid and its secondary windings in the alternator field circuit, and provides a feedback voltage which prevents sudden variations of the alternator field excitation current.

Voltage-adjusting rheostats, one for each regulator, are located on the aft inboard side of the inverter relay box. This rheostat should be set to provide either 200 or 212-volt output from the alternator with no load applied to the alternator. Voltage adjustments should be made at a minimum of 11,000 engine rpm.

A remote trim potentiometer is used in the circuit of each voltage regulator. These units, located on the forward end of the inverter relay box, are set to provides ohms resistance in the control circuit. These trimmers will only give approximately $\pm$ 3 volts variation in system voltage. They should be considered as fixed resistors and treated accordingly unless the voltage adjusting rheostats fail to provide sufficient control.

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FIELD POWER TRANSFORMER-RECTIFIER — DESCRIPTION / OPERATION

1. Description

Field power transformer-rectifiers, one for each alternator system, are installed beneath FLR-12. Each supplies a field control current to the voltage regulator which is in proportion to the alternator three. phase output. Each transformer-rectifier contains a step-down transformer with three primary windings, each primary is connected to a phase of the output voltage. The secondary windings of these transformers are connected to a rectifier network which provides output current proportional to the input. The output current is fed to the voltage regulator through the alternator voltage adjusting rheostat.

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CURRENT BALANCE RELAY — DESCRIPTION / OPERATION

1. Description and Operation

Current balance relays, one for each alternator, are installed in each nacelle relay panel. Each phase of the alternator output is fed through a primary winding of a transformer to the load circuit. Each secondary winding is connected to a bridge rectifier circuit. Two double-coil relays are connected to the bridge circuits. The phase A bridge feeds one coil of one relay, while the phase C bridge feeds one coil of the other relay. The phase B bridge is connected to the two remaining coils. One relay, therefore, is controlled by phase A and B, while the other relay is controlled by phase B and C. Normally, when all phases are balanced, the current flow through each coil of each relay is such that the relays remain de-energized. When an unbalance occurs between any two phases, current flow in the relay coils is such that the related relay is energized. When either of the double coil relays is energized, the balance relay is energized. The current balance relay, when energized, energizes the AC GEN OFF warning light and de-energizes the field power relay, causing the alternator to drop off the line. The de-energized field power relay inserts the field bias resistor into the alternator field, which causes the alternator output to be reduced to almost nothing.

The current balance relay, which has a holding circuit on its coil, may be de-energized after an unbalance has been detected by depressing the RESET position of the alternator control switch. The current balance relay can also be energized by the overvoltage relay, or by one of the undervoltage relays.

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UNDERVOLTAGE RELAYS — DESCRIPTION / OPERATION

1. Description

Two undervoltage relays function to trip their respective alternator when an undervoltage condition exists between phases. One relay coil, in series with a dropping resistor, is connected between phases A and C; the other relay coil, in series with another dropping resistor, is connected between phases B and C. The resistors are adjusted so that the relays will be energized when a line-to-line voltage in the phases between which they are connected exceeds 108 - 131 volts. When the line-to-line voltage between any two phases drops below 108 - 131 volts, the relevant relay is de-energized, its contacts completing a circuit to energize the current balance relay. The current balance relay then de-energizes the field power relay, tripping the alternator.

The dropping resistors and the undervoltage relays for both alternator systems are located in the alternator relay box.

NOTE: Once set, the dropping resistors should not be disturbed unless required by component replacement.

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OVERVOLTAGE SENSING TRANSFORMER-RECTIFIER AND OVERVOLTAGE RELAY — DESCRIPTION / OPERATION

1. Description

Overvoltage sensing transformer-rectifiers are provided for each alternator system. These units function to convert the alternator output voltage into a control current which is proportional to the alternator output. The overvoltage sensing transformer-rectifier is identical to the field transformer-rectifier, having the primary windings of a step down transformer connected to the three-phase output of the alternator, and the secondary windings connected to a rectifier network. The control current output is fed to the overvoltage relay coil through the overvoltage dropping resistor. The resistor is set so that the overvoltage relay is energized when the alternator output voltage exceeds 220-231 volts or 223-240 volts, as applicable. The overvoltage relay, when energized, energizes the current balance relay to trip the alternator.

The overvoltage sensing transformer-rectifiers are mounted atop the alternator relay box. The overvoltage relays and the dropping resistors are located within the alternator relay box.

NOTE: Once set, the dropping resistors should not be disturbed unless required by component replacement.

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ALTERNATOR CONTROL SWITCH, RESET RELAY, AND ANTI-CYCLE RELAY — DESCRIPTION / OPERATION

1. Description

The RESET portion of the alternator control switch is used to de-energize the current balance relay after the alternator has been tripped because of a current unbalance, an overvoltage condition, or an undervoltage condition. Since the 28-volt, dc power to the holding circuit of the current balance relay is routed through a set of contacts of the reset relay, depressing the RESET portion of the alternator control switch energizes the reset relay to open the holding circuit. The current balance relay is then de-energized, energizing the field power relay, which causes the alternator voltage to build up. The reset relay, energized by a portion of the alternator control switch, provides a ground to energize the anti-cycle relay which has a holding circuit on its coil. The anti-cycle relay, when energized, removes the ground from the reset relay, but, because of the time delay capacitor connected across the reset relay coil, the reset relay is not immediately de-energized. The capacitor provides the difference of potential required to keep the reset relay energized long enough to allow the output voltage of the alternator to build up unimpeded by the current balance, overvoltage, and undervoltage circuitry. Once energized, the anti-cycle relay will remain energized as long as the switch is held in the RESET position. The reset relay, however, will remain energized until the time delay is completed. The reset relays and anti-cycle relays are located in the alternator relay box.

LIST OF EFFECTIVE PAGES

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EQUIPMENT/FURNISHINGS

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*** Pages deleted by Revision 45

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CHAPTER 25

EQUIPMENT/FURNISHINGS

PILOTS AND COPILOTS SEATS — DESCRIPTION / OPERATION

1. General

(See Figure 1)

The pilot and copilot seats are adjustable both vertically and horizontally. The vertical adjustment lever is on the inboard side just below the front portion of the seat cushion.

NOTE: The vertical adjustment should be made only when there is a person occupying the seat.

The horizontal adjustment lever is located on the inboard side of the seat, just above the trolley. The seats can be adjusted 4.5 inches vertically and 7.3 inches horizontally.

The seat cushion and seat back are contoured.

Both pilot and copilot seats are the quick removal type. (No tools are required to remove or install the seats.)

Sheet metal and tube construction is used in the manufacture of the seats.

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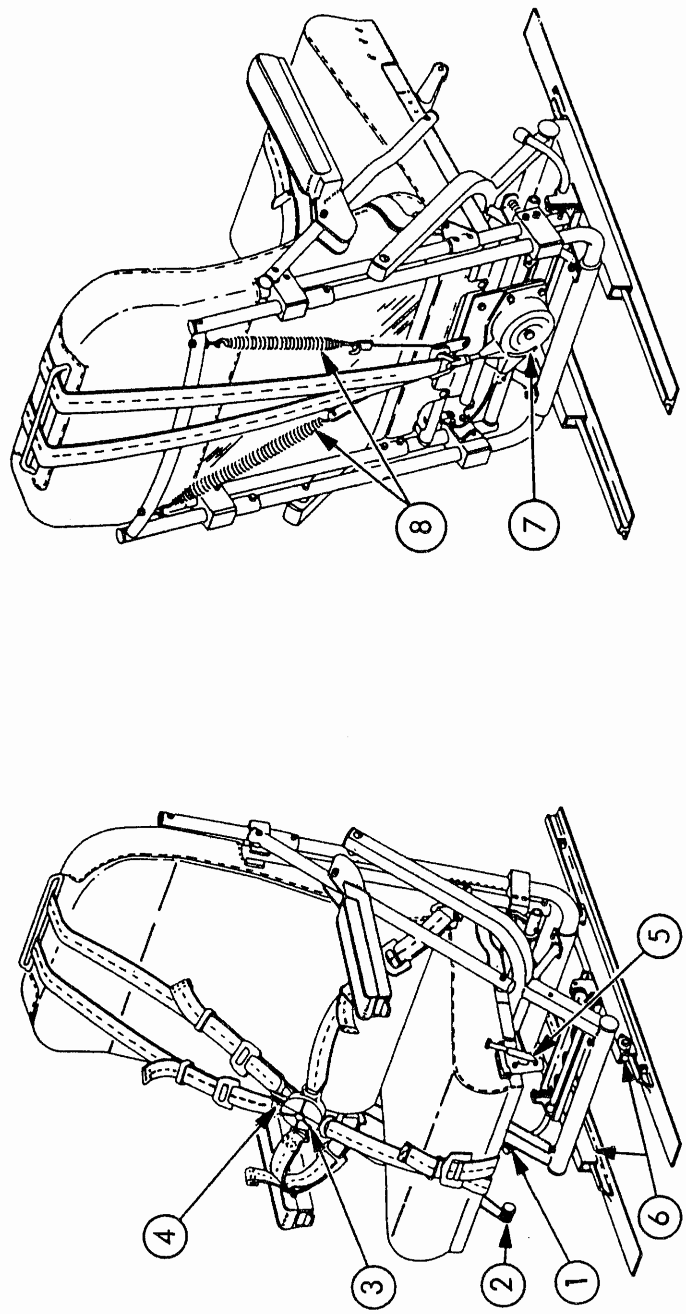

NOTE : PILOT'S SEAT SHOWN;
CO-PILOT'S SEAT OPPOSITE

1. LONGITUDINAL LOCKING LEVER
2. VERTICAL LOCKING LEVER
3. ROTARY LOCKING BUCKLE
4. SHOULDER STRAPS QUICK RELEASE TAB
5. INERTIA REEL LOCKING LEVER
6. TROLLEY
7. INERTIA REEL
8. SEAT RAISING SPRINGS

Pilot and Copilot Seat
Figure 1.

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SAFETY BELTS AND INERTIA REELS — DESCRIPTION / OPERATION

1. General

(See Section 25-0 Figure 1.

The pilots and copilots seats are equipped with safety belts and inertia reels. Their purpose is to maintain the pilot or copilot in his seat when extreme gravitational, centrifugal, and deceleration forces are applied.

The safety belts consist of adjustable straps leading to a rotary locking buckle. The lap straps are secured at the apex of the seat and back rest, and the shoulder harness, consisting of the two shoulder straps, passes through two guides at the top of the back rest, and is attached to a fitting at the inertia reel cable. The rotary locking buckle permits separate insertion and engagement of the two shoulder harness straps, and the inboard seat strap. The buckle is permanently attached to the outboard seat strap.

NOTE: Some aircraft are equipped with a crotch strap which connects to the bottom of the rotary locking buckle.

The pilots and copilots seats are each equipped with an inertia reel, which is attached to the bottom of the seat back. Figure 201. A cable extending from the reel is attached to the shoulder harness. The inertia reel is controlled by a lever located on the outboard side of each seat. With the lever in the UNLOCKED position, the spring-loaded cable is free to extend up to 18 inches from the retracted position, and will retract freely due to the spring loading when released. With the lever in the LOCKED position, ratchet action prevents the cable from extending, but the cable will still retract freely.

If a decelerating force of between two and three g's is encountered with the lever in the UNLOCKED position, an inertia mechanism automatically locks the reel. Normal operation can be restored by moving the lever to the LOCKED position and back to the UNLOCKED position.

2. Operation

A. Safety Belts

- Each safety belt strap is equipped with a buckle to allow adjustment to meet the requirements of the individual. After initial adjustment, no further attention is required. Adjust the seat belts as follows:
- While seated in seat, snap inboard seat strap into rotary locking buckle and adjust strap lengths so that buckle is centered and belt is snug but not restrictive.

B. Inertia Reel

- Locate the inertia reel manual control assembly at the outboard edge of the seat. While holding shoulder straps, place the control in the UNLOCKED position. Allow reel to retract the shoulder straps until the stop is reached.
- Place the inertia reel manual control in the LOCKED position. Snap shoulder harnesses into rotary locking buckle and adjust length of shoulder straps to be snug but not restrictive with shoulder against back of seat.
- Place inertia reel manual control in UNLOCK position. Shoulder straps should allow up to 18 inches forward movement.
- To release all straps simultaneously, rotate the buckle release clockwise. To release only the shoulder straps, move the tab at the top of the buckle forward.

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SAFETY BELTS AND INERTIA REELS — MAINTENANCE PRACTICES

1. Pilots / Copilots Harness — Removal / Installation

(Aircraft 1-178, 322 and 323.)

CAUTION: ENSURE SEATS ARE IN FULL UP POSITION. THE VERTICAL ADJUSTMENT LEVER SHOULD NOT BE MOVED UNLESS THERE IS A PERSON OCCUPYING THE SEAT (WEIGHT IN THE SEAT TO OPPOSE THE VERTICAL SPRING LOAD). KEEP HANDS CLEAR OF VERTICAL SLIDE RAILS TO PREVENT INJURY WHEN SEAT IS MOVED.

A. Removal

(1) Remove seat cushion by unsnapping from seat pan.

(2) Remove crotch strap by removing attaching hardware at forward bottom portion of seat pan.

(3) Unsnap lap belt from fittings provided on back of seat near seat pan.

(4) Unscrew nut and remove screw and washer from inertia reel terminal.

NOTE: Keep rotary buckle locked to prevent loss of a section of harness.

(5) Pull shoulder harness through guide at top of seat back rest.

B. Installation

(1) Slide shoulder harness through guide at top of seat back rest and then down to inertia reel terminal.

NOTE: Adjustment buckles shall face forward.

(2) Install shoulder harness to inertia reel terminal with attaching hardware previously removed.

(3) Snap lap belt to fittings provided on back of seat near seat pan.

(4) Install crotch strap to forward bottom portion of seat pan using hardware previously removed.

(5) Install seat cushion and snap into place.

(Aircraft 179-200 excluding 322 and 323.)

A. Removal

(1) Remove seat cushion by unsnapping from seat pan.

(2) Unsnap lap belt from fittings provided oh back of seat near seat pan.

(3) Unscrew nut and remove screw and washer from inertia reel terminal.

NOTE: Keep rotary buckle locked to prevent loss of a section of harness.

(4) Pull shoulder harness through guide at top of seat back rest.

B. Installation

(1) Slide shoulder harness through guide at top of seat back rest and then down to inertia reel terminal.

NOTE: Adjustment buckles shall face forward.

(2) Install shoulder harness to inertia reel terminal with attaching hardware previously removed.

(3) Snap lap belt to fittings provided on back of seat near seat pan.

(4) Install seat cushion and snap into place.

2. Pilots / Copilots Inertia Reel — Removal / Installation

A. Removal

(1) Remove manual control handle on front outboard side of seat pan.

(2) Remove one cable clamp on outboard rear side of seat pan.

(3) Disconnect harness at inertia reel located behind seat, remove inertia reel.

B. Installation

(1) Install inertia reel with hardware previously removed, connect harness to inertia reel (pilots cable is longer).

(2) Install manual control handle to front outboard side of seat pan.

(3) Install cable clamp at outboard rear side of seat pan.

(4) Perform Inertia Reel — Operational Test, this Section.

3. Inertia Reel — Operational Test

A. At outboard side of seat, place the inertia reel control in the UNLOCK position.

NOTE: Do not remove control or harness cable from inertia reel.

B. Remove inertia reel from back of seat by removing attaching hardware.

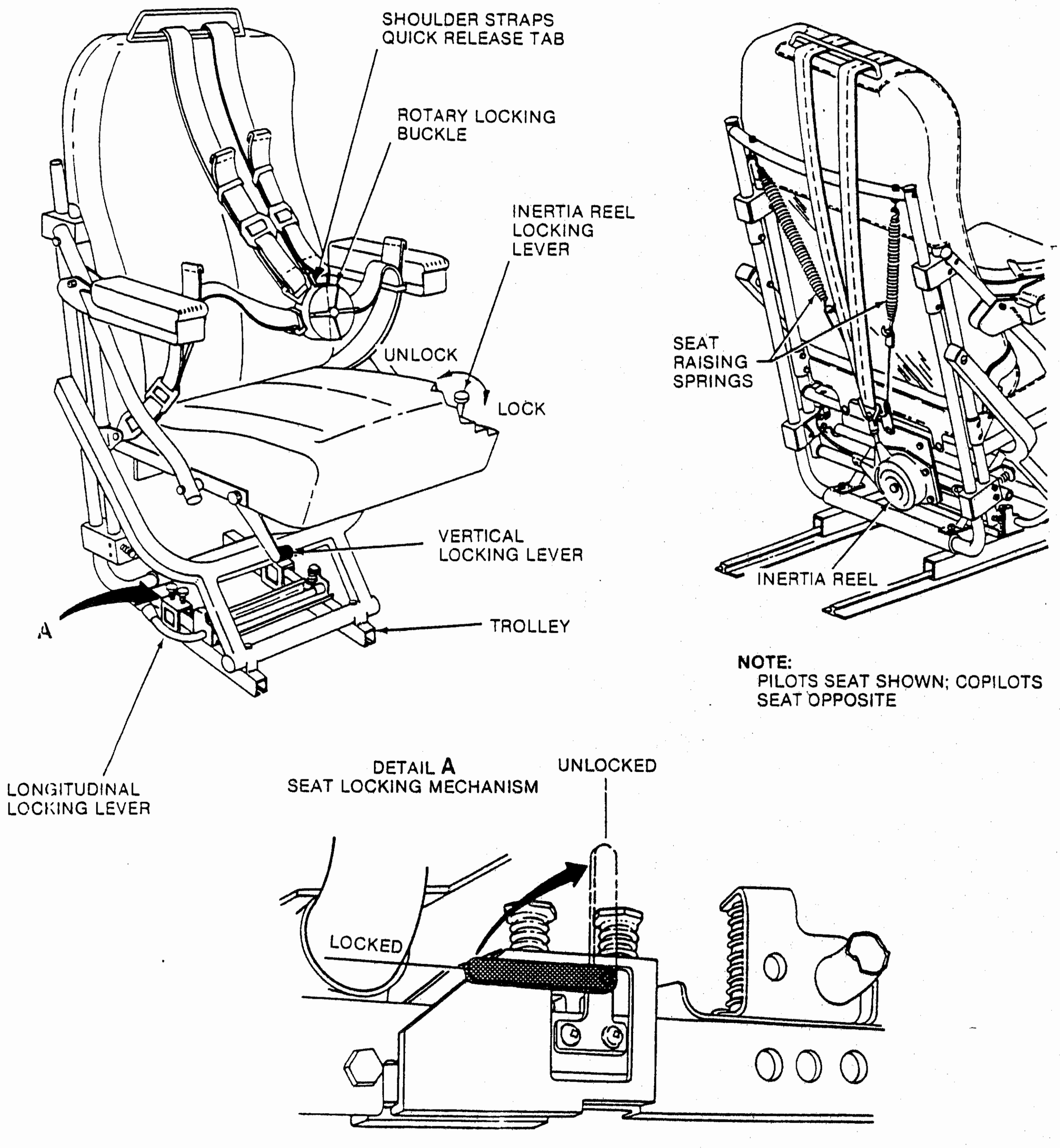
C. While holding reel in one hand with the mounting base down, pull the harness cable slowly from the reel for its full length. Cable should come from reel smoothly and without binding. Cable should be free from kinks and fraying.

D. Allow cable to feed back into reel.

E. Pull the harness cable slowly from the reel. After a few inches are unreeled, continue pulling cable slowly and give the reel a quick downward thrust, stopping the downward motion sharply. This action should prevent any further cable leaving reel. Allow cable to feed back into reel.

F. Install inertia reel assembly on back of seat and secure with attaching hardware.

G. To unlock reel, cycle the control to the forward position, then return to the aft position.



Pilot and Copilot Seat
Figure 201.

PILOT'S AND COPILOT'S SEATS — DESCRIPTION/OPERATION

1. **General** (See figure 1.)

The pilot and copilot seats are adjustable both vertically and horizontally. The vertical adjustment lever is on the inboard side just below the front portion of the seat cushion.

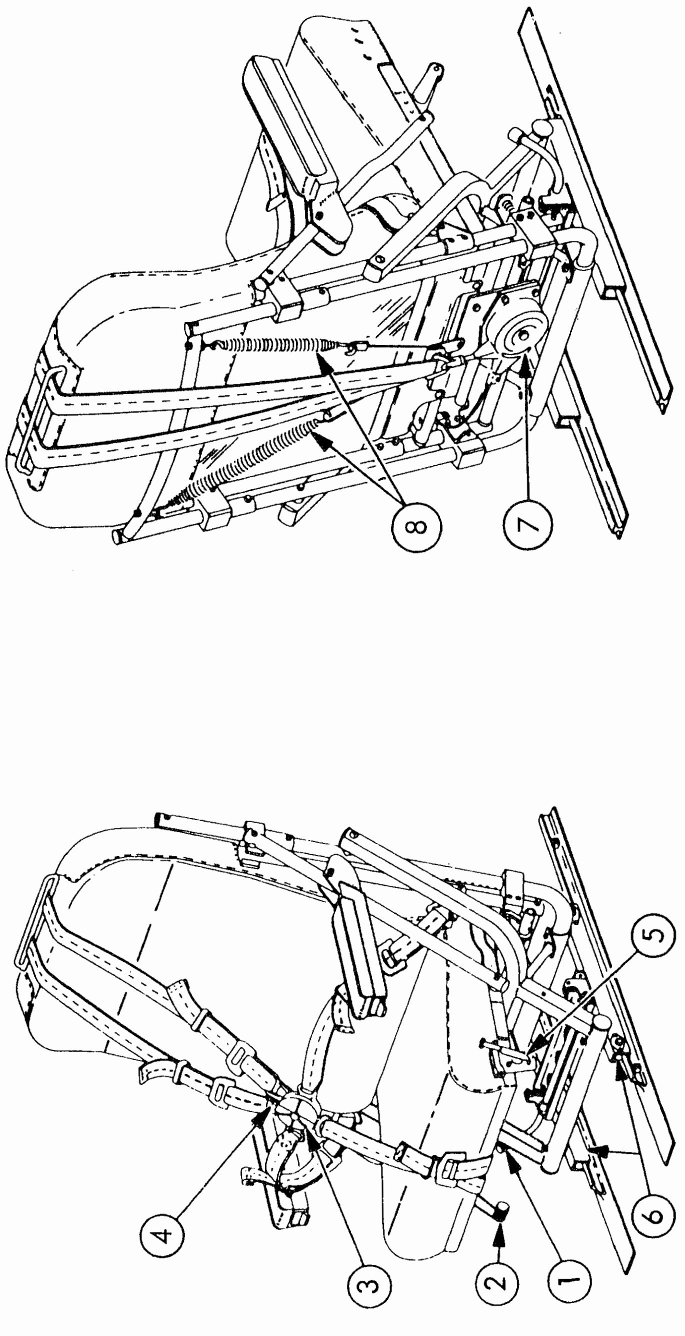
NOTE: The vertical adjustment should be made only when there is a person occupying the seat.

The horizontal adjustment lever is located on the inboard side of the seat, just above the trolley. The seats can be adjusted 4.5 inches vertically and 7.3 inches horizontally.

The seat cushion and seat back are contoured.

Both pilot and copilot seats are the quick removal type. (No tools are required to remove or install the seats.)

Sheet metal and tube construction is used in the manufacture of the seats.



NOTE: PILOT'S SEAT SHOWN;
CO-PILOT'S SEAT OPPOSITE

1. LONGITUDINAL LOCKING LEVER
2. VERTICAL LOCKING LEVER
3. ROTARY LOCKING BUCKLE
4. SHOULDER STRAPS QUICK RELEASE TAB
5. INERTIA REEL LOCKING LEVER
6. TROLLEY
7. INERTIA REEL
8. SEAT RAISING SPRINGS

Pilot's and Co-pilot's Seats
Figure 1

PILOT'S AND COPILOT'S SEAT — REMOVAL/INSTALLATION

1. **Removal of Seats** (Aircraft 1 through 178, including 322 and 323.) (See Figure 401.)

CAUTION: ENSURE SEATS ARE IN FULL UP POSITION. THE VERTICAL ADJUSTMENT LEVER SHOULD NOT BE MOVED UNLESS THERE IS A PERSON OCCUPYING THE SEAT (WEIGHT IN THE SEAT TO OPPOSE THE VERTICAL SPRING LOAD). KEEP HANDS CLEAR OF VERTICAL SLIDE RAILS TO PREVENT INJURY WHEN SEAT IS MOVED.

A. Pull horizontal adjustment handle forward and track seat to middle position.

B. Remove seat from trolley, retaining hardware; remove seat.

NOTE: If trolley is to be left in the aircraft, install washers, spacers, and bolts finger tight in trolley. If trolley is to be removed, push control column to full forward position and accomplish steps C. through F.

C. Remove forward trolley stops from tracks.

D. Pull horizontal seat adjustment handle and remove trolley from track.

E. Install forward trolley stops in tracks.

F. Install trolley to seat.

2. **Installation of Seats** (Aircraft 1 through 178, including 322 and 323.) (See Figure 401.)

NOTE: If trolley was removed from track, accomplish steps A. through I. If trolley was not removed from track, accomplish steps F. through I.

A. Remove trolley from seat, retaining hardware.

B. Remove forward trolley stops from tracks.

C. Position trolley at forward end of tracks with horizontal adjustment handle pulled forward.

NOTE: Control column must be in full forward position.

D. Track trolley to middle position.

E. Reinstall forward trolley stops in tracks using hardware previously removed.

F. Fold Armrests and position seat on trolley; secure seat to trolley using hardware previously removed.

NOTE: Two thick spacers shall be installed on the forward bolts and the two thin spacers on the aft bolts.

G. Move seat forward and aft to check for freedom of movement.

H. Lubricate seat and tracks (if required).

3. **Removal of Seats** (Aircraft 179 through 200, excluding 322 and 323.) (See Figure 401.)

   A. Pull up on horizontal adjustment lever and track seat aft lining up black markings on seat base track locking mechanisms and aft track cutout.

   B. Raise seat latch on seat locking mechanism (located on seat track channel) to unlocked position; do this to both left and right hand sides.

   C. Align red markings on seat base channel both forward and aft, with red markings on track cutouts by tracking seat aft.

   D. Remove seat from tracks.

4. **Installation of Seats** (Aircraft 179 through 200, excluding 322 and 323.) (See Figure 401.)

   A. Raise seat latch on seat locking mechanism (located on seat track channel) to unlocked position; do this to both left and right hand sides.

   B. Align red markings on seat base channel both forward and aft, with red markings on track cutouts.

   C. Install seat onto tracks.

   D. Pull up on horizontal adjustment lever and track seat aft lining up black markings on seat base track locking mechanism and aft track cutout.

   E. Lower seat latch to locked position.

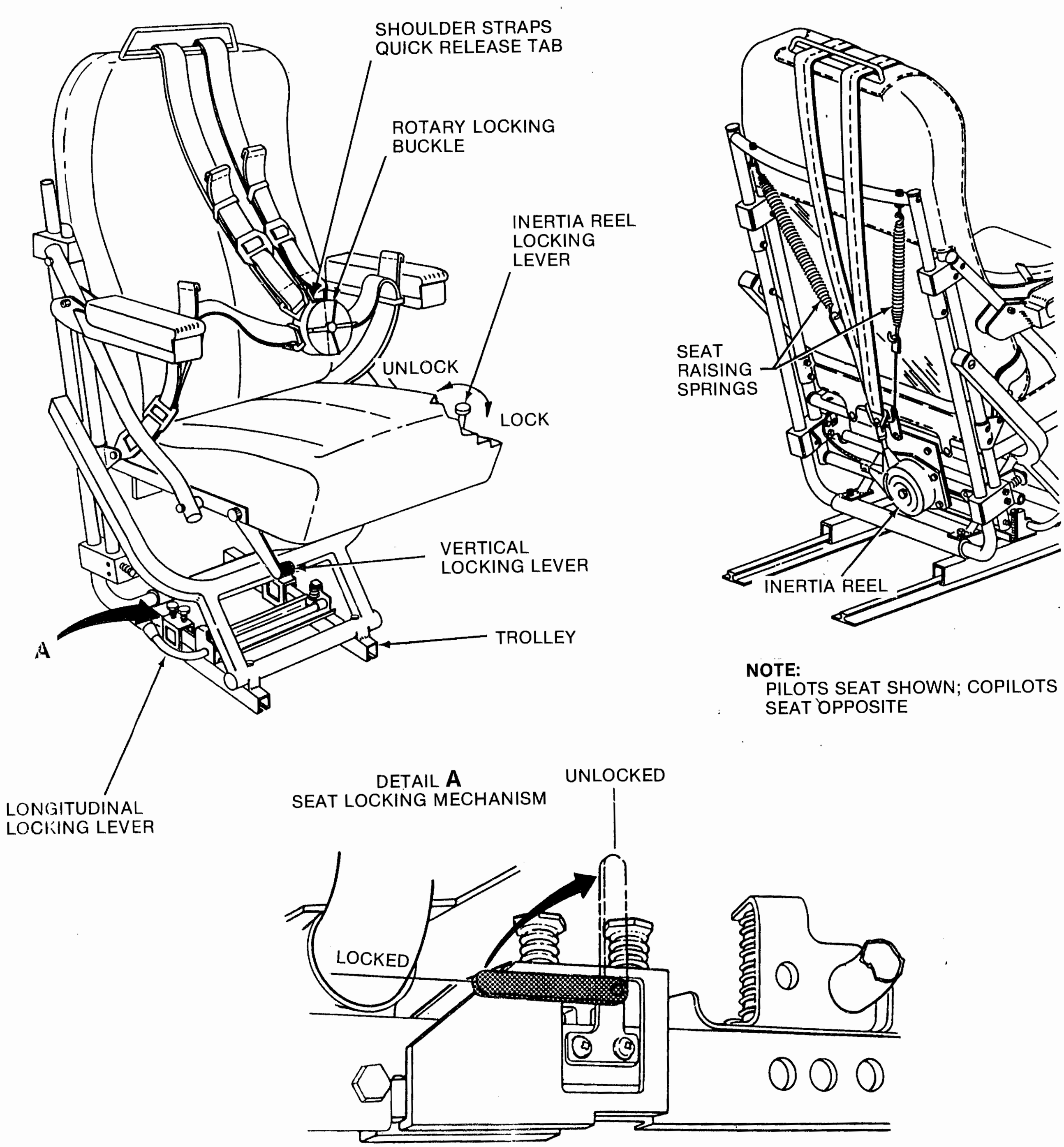
   F. Track seat forward until seat locks in position.

   NOTE: If there is play between the trolley and the track, accomplish step G. If not, accomplish only steps H. and I.

   G. Adjust cam rollers by backing off nut, pulling out knurled cam and rotating until play is taken out of seat between trolley and track; push in knurled cam and secure nut.

   H. Move seat forward and aft to check for freedom of movement.

   I. Lubricate seat and tracks (if required).



Pilot's Seat
Figure 401

SAFETY BELTS AND INERTIA REELS — DESCRIPTION/OPERATION

1. **General** (See Section 25-1-1, Figure 1 and Figure 401.)

The pilot's and copilot's seats are equipped with safety belts and inertia reels. Their purpose is to maintain the pilot or copilot in his seat when extreme gravitational, centrifugal, and deceleration forces are applied.

The safety belts consist of adjustable straps leading to a rotary locking buckle. The lap straps are secured at the apex of the seat and back rest, and the shoulder harness, consisting of the two shoulder straps, passes through two guides at the top of the back rest, and is attached to a fitting at the inertia reel cable. The rotary locking buckle permits separate insertion and engagement of the two shoulder harness straps, and the inboard seat strap. The buckle is permanently attached to the outboard seat strap.

NOTE: Some aircraft are equipped with a crotch strap which connects to the bottom of the rotary locking buckle.

The pilot's and copilot's seats are each equipped with an inertia reel, which is attached to the bottom of the seat back. (See Section 25-1-1, Figure 1 and Figure 401.) A cable extending from the reel is attached to the shoulder harness. The inertia reel is controlled by a lever located on the outboard side of each seat. With the lever in the UNLOCKED position, the spring-loaded cable is free to extend up to 18 inches from the retracted position, and will retract freely due to the spring loading when released. With the lever in the LOCKED position, ratchet action prevents the cable from extending, but the cable will still retract freely.

If a decelerating force of between two and three g's is encountered with the lever in the UNLOCKED position, an inertia mechanism automatically locks the reel. Normal operation can be restored by moving the lever to the LOCKED position and back to the UNLOCKED position.

2. **Operation**

A. Safety Belts

Each safety belt strap is equipped with a buckle to allow adjustment to meet the requirements of the individual. After initial adjustment, no further attention is required. Adjust the seat belts as follows:

While seated in seat, snap inboard seat strap into rotary locking buckle and adjust strap lengths so that buckle is centered and belt is snug but not restrictive.

B. Inertia Reel

(1) Locate the inertia reel manual control assembly at the outboard edge of the seat. While holding shoulder straps, place the control in the UNLOCKED position. Allow reel to retract the shoulder straps until the stop is reached.

(2) Place the inertia reel manual control in the LOCKED position. Snap shoulder harnesses into rotary locking buckle and adjust length of shoulder straps to be snug but not restrictive with shoulder against back of seat.

(3) Place inertia reel manual control in UNLOCK position. Shoulder straps should allow up to 18 inches forward movement.

(4) To release all straps simultaneously, rotate the buckle release clockwise. To release only the shoulder straps, move the tab at the top of the buckle forward.

PILOTS AND COPILOTS HARNESS — MAINTENANCE PRACTICES

1. Pilots and Copilots Harness — Removal / Installation

A. Removal

CAUTION: ENSURE SEATS ARE IN FULL UP POSITION. THE VERTICAL ADJUSTMENT LEVER SHOULD NOT BE MOVED UNLESS THERE IS A PERSON OCCUPYING THE SEAT (WEIGHT IN THE SEAT TO OPPOSE THE VERTICAL SPRING LOAD). KEEP HANDS CLEAR OF VERTICAL SLIDE RAILS TO PREVENT INJURY WHEN SEAT IS MOVED.

(1) Remove seat cushion by unsnapping from seat pan.

(2) Remove crotch strap by removing attaching hardware at forward bottom portion of seat pan. (Aircraft 1-178, including 322 and 323)

(3) Unsnap lap belt from fittings provided on back of seat near seat pan.

(4) Unscrew nut and remove screw and washer from inertia reel terminal.

NOTE: Keep rotary buckle locked to prevent loss of a section of harness.

(5) Pull shoulder harness through guide at top of seat back rest.

B. Installation

(1) Slide shoulder harness through guide at top of seat back rest and then down to inertia reel terminal.

NOTE: Adjustment buckles shall face forward.

(2) Install shoulder harness to inertia reel terminal with attaching hardware previously removed.

(3) Snap lap belt to fittings provided on back of seat near seat pan.

(4) Install crotch strap to forward bottom portion of seat pan using hardware previously removed. (Aircraft 1-178, including 322 and 323)

(5) Install seat cushion and snap into place.

2. Pilots and Copilots Inertia Reel — Removal / Installation

A. Removal

(1) Remove manual control handle on front outboard side of seat pan.

(2) Remove one cable clamp on outboard rear side of seat pan.

(3) Disconnect harness at inertia reel located behind seat; remove inertia reel.

B. Installation

(1) Install inertia reel with hardware previously removed; connect harness to inertia reel (pilots cable is longer).

(2) Install manual control handle to front outboard side of seat pan.

(3) Install cable clamp at outboard rear side of seat pan.

(4) Check inertia reel for proper operation and locking.

3. **Inertia Reel — Operational Test**

A. At outboard side of seat, place the inertia reel control in the UNLOCK position.

NOTE: Do not remove control or harness cable from inertia reel.

B. Remove inertia reel from back of seat by removing attaching hardware.

C. While holding reel in one hand with the mounting base down, pull the harness cable slowly from the reel for its full length. Cable should come from reel smoothly and without binding. Cable should be free from kinks and fraying.

D. Allow cable to feed back into reel.

E. Pull the harness cable slowly from the reel. After a few inches are unreeled, continue pulling cable slowly and give the reel a quick downward thrust, stopping the downward motion sharply. This action should prevent any further cable leaving reel. Allow cable to feed back into reel.

F. Install inertia reel assembly on back of seat and secure with attaching hardware.

G. To unlock reel, cycle the control to the forward position, then return to the aft position.

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FIRE PROTECTION

CHAPTER 26

FIRE PROTECTION

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FIRE PROTECTION — DESCRIPTION / OPERATION

1. General

Fire protection provisions include a detecting and indicating system, and an electrically-controlled fire extinguishing system for each engine nacelle and for the APU, enclosure.

Each engine nacelle area is divided into three zones. The detecting system utilizes sensing elements routed in such a way within these three zones that the paths of air flow are fully covered and high temperatures, resulting from fire in the nacelle area, are detected. The sensing elements of the detecting system are located in the engine compressor-accessory section (zone I), the engine combustion-turbine section (zone II), and the tailpipe section (aft nacelle) (zone III) in each nacelle. Two T-handles, which incorporate red fire warning lights and are placarded FIRE PULL, and three red fire warning lights, of which one is placarded APU FIRE and two AFT NACELLE FIRE, comprise the indicating system. The AFT NACELLE FIRE warning lights are included to provide separate fire warning for zone III of each nacelle.

Spot type detectors, such as thermal switches, are used to detect fire or overheat in a limited space. These spot detectors are tied in directly to the master warning system. They indicate overheated conditions on the engine-driven alternators and generators as well as overheating in the radio rack and air conditioning system blower.

A separate APU FIRE warning light, operating in conjunction with three fire detection thermal switches which are mounted at strategic points in the APU enclosure, provides detection and indication of fire or excessive heat in the APU area.

The baggage compartment fire and smoke detection system is installed by the furnishing agency. The type utilized is at the discretion of the customer subject to FAA approval. Certain provisions for installation of electrical type baggage compartment smoke and fire detection devices are provided in the basic aircraft.

The fire extinguisher system is provided to control hazardous fire within the engine nacelle area and the APU enclosure. The aircraft is equipped with two similar but independent, electrically controlled fire extinguishing systems for the nacelle areas, and a separate fire extinguishing system for the APU enclosure. One system is installed in the left nacelle and another in the right nacelle. Each nacelle includes a fire extinguishing agent container (bottle) which is charged with bromotrifluoromethane (CF_3Br) and nitrogen and equipped with two electrically fired discharge valves. Switches, one located behind each FIRE PULL T-handle and operating in conjunction with the discharge valves, provide a two-shot system which is so arranged that the extinguishing agent in either bottle may be discharged into the compartments or either nacelle. Each bottle discharges in approximately two seconds at ambient temperatures as low as — 65°F.

The APU fire extinguisher system includes a fire extinguishing agent container (bottle) which is charged with bromotrifluoromethane and nitrogen and is equipped with a single electrically actuated discharge valve. Fire extinguishing action in the APU enclosure is effected when the APU EXTINGUISHER SWITCH is actuated.

WARNING: BROMOTRIFLUOROMETHANE (CF_3BR) IS NON-TOXIC AND CLASSED AS NON-POISONOUS. HOWEVER, IT CAN BE HARMFUL AND SHOULD BE HANDLED WITH CARE. DO NOT BREATHE THE VAPOR OR PERMIT LIQUID TO COME IN CONTACT WITH SKIN OR CLOTHING. IF LEAKAGE IN A CLOSED SPACE IS SUSPECTED, THIS SPACE SHOULD NOT BE ENTERED UNTIL ALL VAPOR HAS BEEN DISSIPATED AND THE AREA IS WELL VENTILATED. USE A HALIDE DETECTOR (PRESS-O-LITE, OR EQUIVALENT) TO ENSURE THAT AREA IS SAFE.

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FIRE DETECTION SYSTEM — DESCRIPTION/OPERATION

1. **Description/Operation**

A. Nacelle Fire Detector System (See figures 1, 2 and 3.)

The following major units are included in the nacelle fire detector system.

Component	No. Per Aircraft	Location
Fire Detector Box - FWD (zones I and II)	2	Left nacelle fuselage station 264.781. Right nacelle fuselage station 213.125.
FIRE PULL T-Handle	2	Cockpit Eyebrow Panel.
Fire Detector Box - AFT (zone III)	2	Left and Right nacelle station 331-1/2.
AFT NACELLE FIRE Detector Warning Light	2	Cockpit eyebrow panel adjacent to each FIRE PULL T-handle.
FIRE DET TEST Switch	1	Cockpit eyebrow panel, pilot's side.
Test Relay	1	Mid relay junction box.
Test Relay	2	One adjacent to each aft fire detector box.
R NAC FD #1 circuit breaker	1	Copilot's circuit breaker panel
L NAC FD #1 circuit breaker	1	Copilot's circuit breaker panel
R NAC FD #2 circuit breaker	1	Copilot's circuit breaker panel
L NAC FD #2 circuit breaker	1	Copilot's circuit breaker panel
R NAC AFT FD circuit breaker	1	Copilot's circuit breaker panel
L NAC AFT FD circuit breaker	1	Copilot's circuit breaker panel
NAC FD TEST circuit breaker	1	Copilot's circuit breaker panel
AFT FIRE DET circuit breaker	1	Pilot's circuit breaker panel

Each engine nacelle area is divided into three zones for purposes of fire detection. Sensing elements are installed in each nacelle zone. In the compressor-accessory section (zone I), the combustion-turbine section (zone II), and the tailpipe section (zone III).

(1) Zones I and II. (Engine)

The fire detectors in zones I and II are of the continuous, resetting type and operate on 115 volts 400 cycle ac from the instrument ac bus through the R. NAC. FD #2 circuit breaker and the L. NAC. FD #2 circuit breaker, respectively. The dual warning lights in each FIRE PULL T-handle operate on 28 volts dc from the essential dc bus through the R. NAC. FD #1 circuit breaker and L. NAC. FD #1 circuit breaker, respectively.

The sensing elements are stainless steel capillaries containing central electrodes separated from the capillary walls by temperature-sensitive material which completely fills the capillaries. The electrical resistance of the filling material decreases with increases in temperature and increases as the temperature falls. When the temperature (resistance) of the element reaches a critical value, the current flow in the circuit becomes sufficient to operate the relay in the fire warning circuit. When this occurs, the appropriate T-handle in the cockpit lights up.

The FIRE PULL T-handles, in addition to containing dual fire warning lights, also provide means for the deactivation of certain systems which might contribute fuel or other flammable fluids to a nacelle fire. The deactivation is accomplished by means of switches mounted on the back of the T-handles. When the right T-handle is pulled, the switches are actuated, causing the follow to occur:

The cutout of the right dc generator is opened.

The field of the right ac alternator is opened.

The right hydraulic shutoff valve is energized and closes to stop the flow of hydraulic fluid from the fuselage to the nacelle.

The right main fuel shutoff valve is energized and closes to stop the flow of fuel at the wing rear beam.

The right water methanol shutoff valve is energized and closes to stop the flow of water methanol forward of the nacelle fire wall.

Similarly, when the left T-handle is pulled, limit switches are actuated causing the left side units to be de-activated.

Continuity between the sensing elements and fire detectors of the nacelle fire detector circuits may be checked and tested by depressing the FIRE DET. TEST switch. Pressing the test switch causes test relays in the right and left forward fire detector boxes to energize. This action simulates a temperature rise in zones I and II of the nacelles, and the warning circuits operate, causing the lights in the right and left T-handles to light up.

(2) Zone III. (Aft Nacelle)

The fire detectors in zone III are of the continuous, resetting type and operate on 17-31 volts dc from the essential dc bus through the L. NAC. FD circuit breaker and the R. NAC. AFT. FD circuit breaker, respectively.

The sensing elements consist of a tube surrounding two wires which are separately imbedded in a ceramic core. When the temperature (resistance) of the element reaches a critical value, the indicator relay circuit is triggered, completing a circuit to the warning light mounted on the eyebrow panel.

Pressing the FIRE DET. TEST switch also causes both aft nacelle fire detector test relays to energize. When energized, these relays ground the sensing element circuits to zone III of each nacelle, causing a triggering circuit in the right and left aft fire detector control units to actuate and the right and left AFT NACELLE FIRE detector warning lights come on.

B. APU Fire Detector System (See Figure 3).

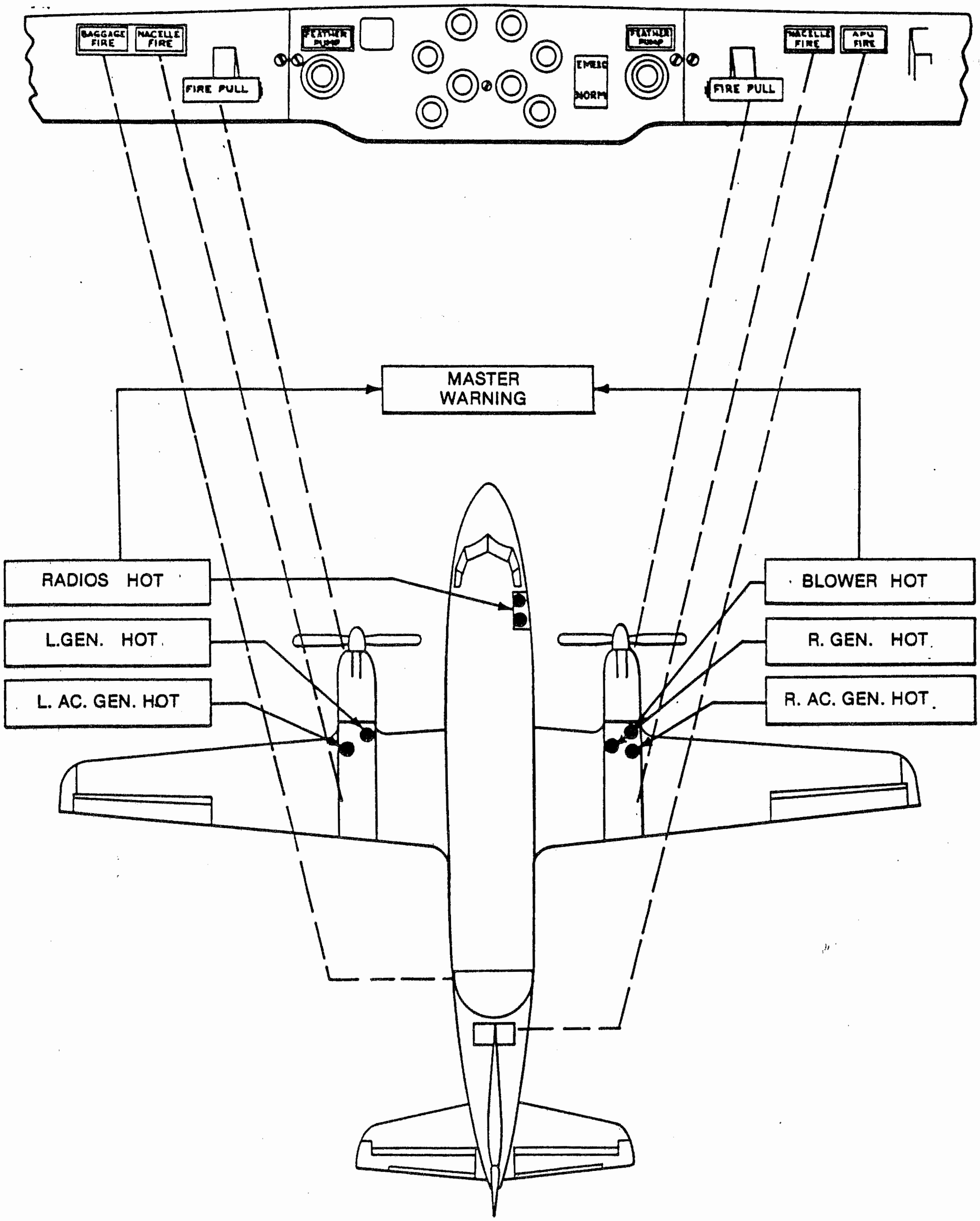
Three fire detector switches are mounted at strategic points in the APU area. Two of these switches, set to close at 450°F, are mounted to monitor temperatures in the vicinity of the fuel control components and the ignition box and generator. The third switch, set to close at 600°F, is mounted adjacent to the combustion chamber to monitor temperatures in that vicinity.

Unit	No. Per A/C	Location
Generator overheat switches	2	One in each exhaust duct
Alternator overheat switches	2	One in each alternator exhaust duct
Radio rack overheat switches	2	One adjacent to each outflow valve
Blower overheat switch	1	Discharge side of blower duct (right engine only).
GEN O'HEAT circuit breaker	1	Pilots circuit breaker panel

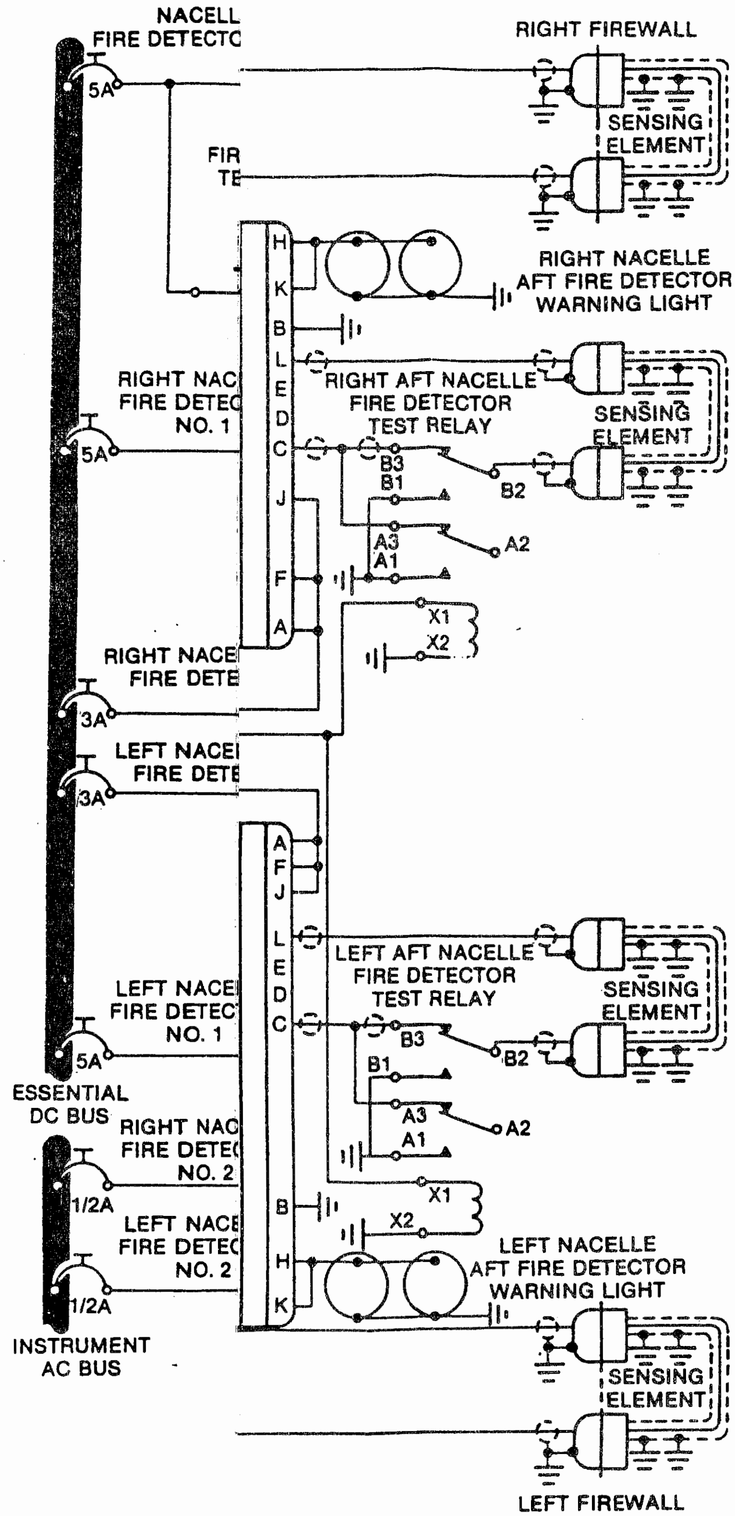
A system consisting of seven thermal switches is used to detect fire or overheating conditions in the engine accessory gearbox driven generators and alternators, the air conditioning system blower and radio rack located at Fuselage Station 133 to 169. The generator and alternator detectors, which are set to close at 300°F, are mounted in the exhaust end of their respective cooling ducts. (See Figure 4). When the air passing through an exhaust duct reaches 300°F, the switch will close and cause an appropriate warning light to light up in the master warning system.

The overheat switches for the radio rack are located at a point just before exhaust air enters the outflow valve. Before the air reaches this point it has passed through the radio rack and inverter area. If these areas have overheated to an abnormal degree (200°F) the switches will close and cause the RADIO HOT warning light to light up in the master warning system (See Figure 5). The circuit receives power from the essential dc bus through the closed GEN O'HEAT circuit breaker.

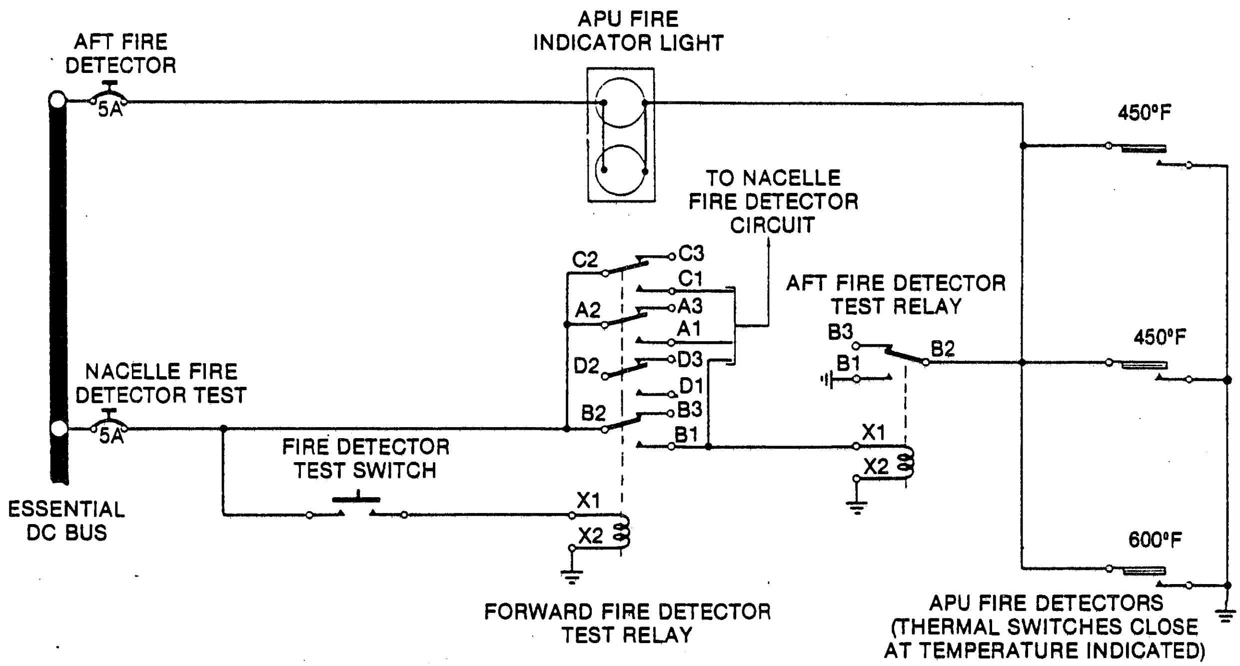
If the blower discharge temperature rises to 410 $\pm$ 10°F, the overheat switch will close causing the BLOWER HOT warning light to light up in the master warning system. Power for this circuit comes from the essential dc bus through the closed GEN O'HEAT circuit breaker. (See Figure 6)



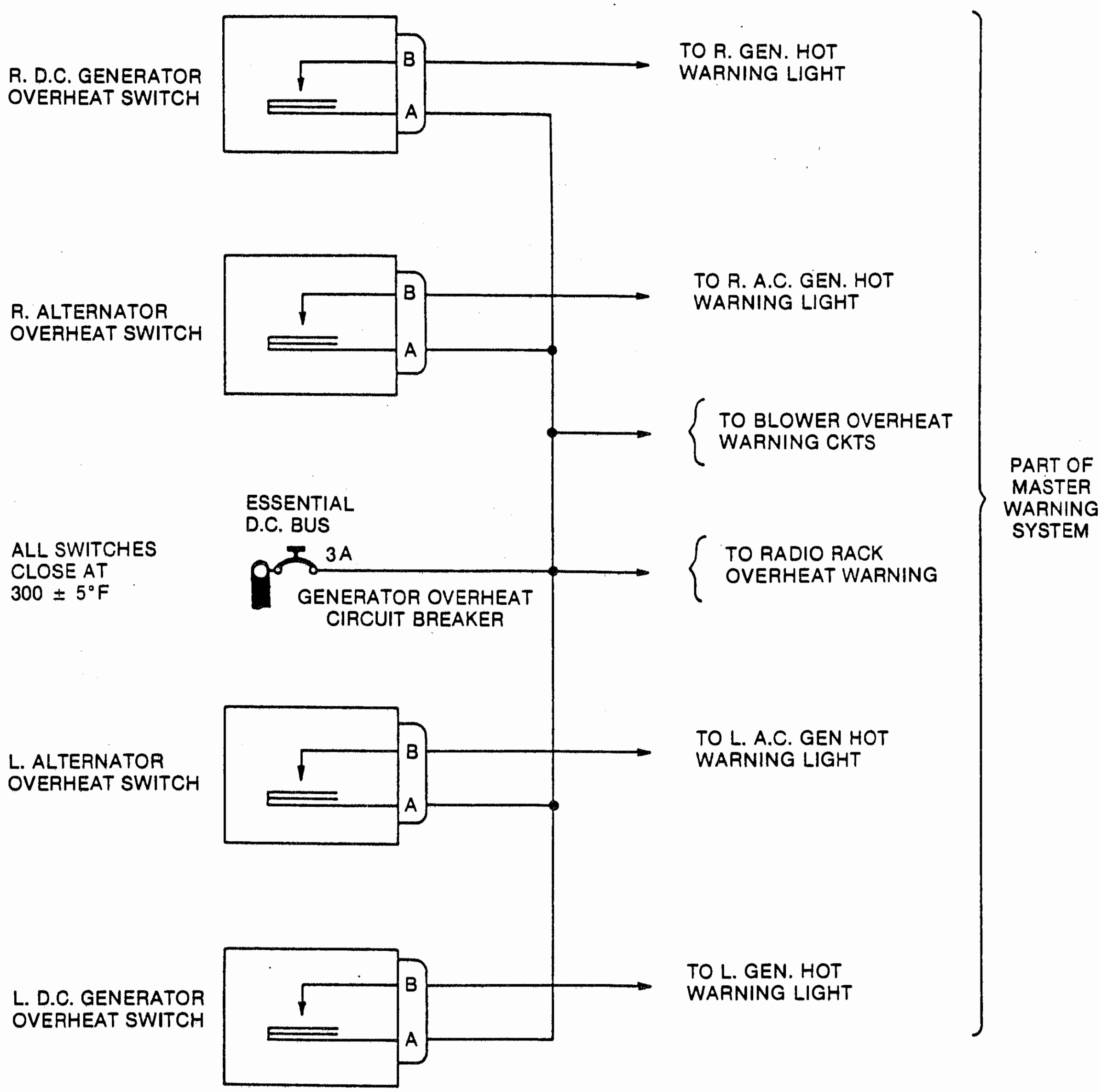
Fire Detection Areas and Indicators
Figure 1.



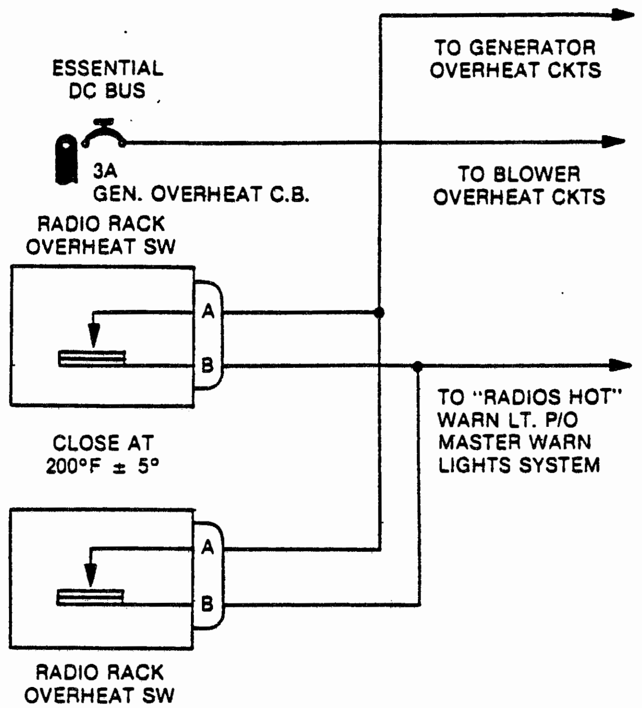
e Detector System — Schematic
Figure 2.



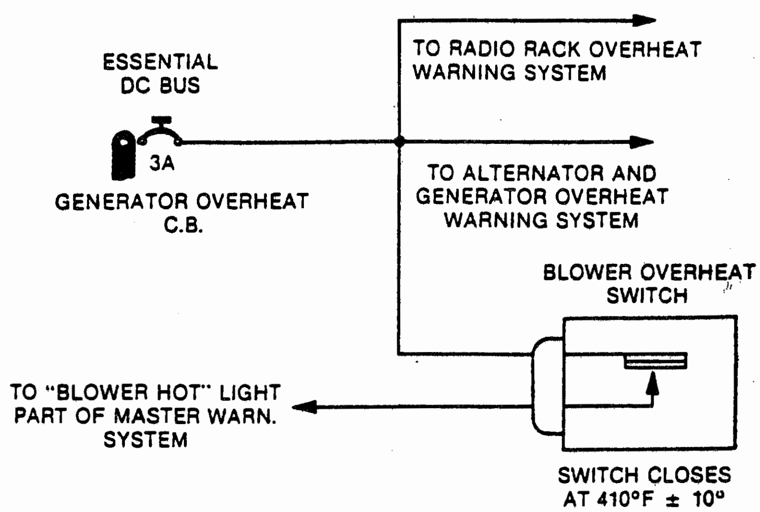
APU Fire Detector System — Schematic
Figure 3.



Generator and Alternator Overheat Circuit — Simplified Schematic
Figure 4.



Radio Rack Overheat Circuit — Simplified Schematic
Figure 5.



Blower Overheat Circuit — Simplified Schematic
Figure 6.

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FIRE DETECTION SYSTEM — FAULT ISOLATION

FAULT	POSSIBLE CAUSE	CORRECTION
FIRE DET. TEST switch pressed, left AFT NACELLE FIRE warning light off.	Defective lamps.	Replace lamps.
	Open supply circuit.	Check power circuit voltage. Reset L NAC AFT FD circuit breaker and NAC FD TEST circuit breaker. Check for short in input (connector plug).
	Malfunctioning test switch or test relays.	Check for 24-volt, dc supply to switch and test relays. Replace switch or relays if defective.
	Malfunctioning left aft fire detector box.	Replace fire detector box.
	Element broken.	Check all connectors. Check element continuity and insulation resistance. Replace if defective.
FIRE DET. TEST switch pressed, right AFT NACELLE FIRE warning light off.	Defective lamps.	Replace lamps.
	Open supply circuit	Check power circuit voltage. reset R NAC AFT FD circuit breaker and NAC FD TEST circuit breaker. Check for short in input (connector plug).
	Malfunctioning test switch or test relays.	Check for 24-volt, dc supply to switch and test relays. Replace switch or relays if defective.
	Malfunctioning right aft detector box.	Replace fire detector box.
	Element broken.	Check all connectors. Check element continuity and insulation resistance. Replace if defective.
FIRE DET. TEST switch pressed, right FIRE PULL T-handle warning light off.	Defective lamps.	Replace lamps.
	Open supply circuit.	Check ac and dc power circuit voltage. Reset R NAC FD No. 2 circuit breaker, R NAC FD No. 1 circuit breaker, and NAC FD TEST circuit break er.
	Malfunctioning test switch or test relays.	Check for 24-volt, dc supply to switch and test relays. Check for 24 volts dc at pin C and ground pin D of forward right fire detector box. Check for 115 volts ac at pin A and ground pin B of forward right fire detector box.
	Malfunctioning forward right fire detector box.	Replace fire detector box.
	Element broken.	Check all connectors. Check element continuity and insulation resistance. Replace if defective.

FAULT	POSSIBLE CAUSE	CORRECTION
FIRE DET. TEST switch pressed, left FIRE PULL T handle warning light off.	Defective lamps.	Replace lamps.
	Open supply circuit.	Check ac and dc power circuit voltage. Reset L NAC FD No. 2 circuit breaker, L NAC FD No. 1 circuit breaker and NAC FD TEST circuit breaker.
	Malfunctioning test switch or test relays.	Check for 24-volt, DC supply to switch and test relays. Check for 24 volts DC at pin C and ground pin D of forward left fire detector box. Check for 115 volts AC at pin A and ground pin B of forward left fire detector box.
	Malfunctioning forward left fire detector box.	Replace fire detector box.
	Element broken.	Check all connectors. Check element continuity and insulation resistance. Replace if defective.
FIRE DET. TEST switch pressed, APU FIRE warning light off.	Defective lamps.	Replace lamps.
	Open supply circuit.	Check power circuit voltage. Reset NAC FD TEST circuit breaker and AFT FIRE DET circuit breaker.
	Malfunctioning test switch or test relays.	Check for 24-volt, dc supply to switch and test relays. Replace switch or relays if defective.
Fire warning light(s) on, test switch not pressed.	Shorted test switch or test relays.	Check for short circuit.
	Damaged elements or thermal swi-tches causing short.	Replace element or thermal switch.
	Malfunctioning fire detector box.	Replace fire detector box.

FIRE DETECTION SYSTEM — MAINTENANCE PRACTICES

1. **Fire Detection System — Operational Test**

   A. Energize essential dc bus, instrument ac bus, and main dc bus. (BATT SW - NORM - ESS AC BUS SEL SW - NORM - EXT POWER SW - ON (if necessary)

   B. To checkout the fire detection system, nacelle (zone III), proceed as follows:

      (1) Close the following circuit breakers: (Copilots panel)

      - L NAC AFT FD circuit breaker
      - R NAC AFT FD circuit breaker
      - NAC FD TEST circuit breaker

      (2) Depress FIRE DET. TEST switch on eyebrow panel. Right and left AFT NACELLE FIRE warning lights should come on, indicating continuity of wiring, fire detector boxes, sensing elements, and relays.

      (3) Release FIRE DET. TEST switch. Right and left AFT NACELLE FIRE warning lights should go out.

   C. To checkout the fire detection system, nacelle (zones I and II), proceed as follows:

      (1) Provide ac power by operating an inverter and close the following circuit breakers: (Copilots panel)

      - L NAC FD No. 1 circuit breaker
      - R NAC FD No. 1 circuit breaker
      - L NAC FD No. 2 circuit breaker
      - R NAC FD No. 2 circuit breaker
      - NAC FD TEST circuit breaker

      (2) Depress FIRE DET TEST switch on the eyebrow panel. Warning lights in left and right FIRE PULL T-handles should come on, indicating continuity of wiring, fire detector boxes, sensing elements, and relays.

      (3) Release FIRE DET TEST switch. Warning lights in right and left T-handles should go out.

   D. To check-out the fire detection system, APU, proceed as follows:

      (1) Close the following circuit breakers

      - AFT FIRE DET. circuit breaker (Pilots panel)
      - NAC FD TEST circuit breaker (Copilots)

      (2) Depress FIRE DET. TEST switch on the eyebrow panel. APU FIRE warning lights should come on.

      (3) Release FIRE DET. TEST switch. APU FIRE warning lights should go out.

2. **Sensing Element Coupling Nut Washers — Replacement**

   NOTE: Replace copper sealing washer if sensing element in either engine or nacelle fire detection system is disturbed. This will ensure the air/moisture tight connection required for proper functioning of the fire detection system.

   A. Removal

      (1) Unscrew coupling nut.

      (2) Remove and discard copper sealing washer.

   B. Installation

      NOTE: Ensure that all parts are free of dirt, oil and moisture before reconnecting. Check for particles of the previous washer in the coupling thread.

      CAUTION: DO NOT USE MORE THAN ONE WASHER AT ANY ONE CONNECTION. THIS CAN CAUSE THE THREADS TO STRIP BECAUSE OF DECREASED ENGAGEMENT.

      ENSURE ELEMENT IS SUPPORTED EVERY 9 TO 12 INCHES TO PREVENT CHAFING.

      (1) Install the following new copper sealing washers:

(a) Graviner part number (on engine system.) - D-2004

(b) Kidde part number (on nacelle system.) - 209592

(2) Tighten coupling nuts as follows:

(a) Torque nuts in engine system 80 to 100 inch-pounds.

(b) Torque nuts in nacelle system 50 to 70 inch-pounds.

3. Thermal Switch — Maintenance

(Generator, Alternator, Blower, Radio Rack and APU Switches)

A. All thermal switches must be checked for trip temperature with appropriate test equipment at intervals specified in Chapter 5 of Maintenance Manual.

NOTE: Those thermal switches which are the screw-in type (all except the three in the APU) when installed in their appropriate boss must not be torqued in excess of 25 inch-pounds. The same practice should be observed when connecting the thermal switch electrical plug. Over torquing of the switch in any form will distort the body of the switch and could result in changing the trip point setting.

B. The APU thermal switches are flange mounted. The screws which hold the switches to the APU container cannot distort the thermal switch body by over torquing, however, these three switches have stud type electrical connections. Precautions should always be exercised that these studs are not torqued in excess of 25 inch-pounds when making electrical connections to the APU thermal switches.

Preferred method for checking trip temperature of overheat switches is by using Techne fluidized bath equipment. Refer to test equipment manufacture instructions to test switch actuation.

C. Any thermal switch which is dented or bent along its sensing body should not be used.

D. Electrical check of aircraft wiring with essential dc bus energized, involves removing the appropriate thermal switch plug and shorting the two pins of the plug together in order to obtain the appropriate indication in the cockpit. In the case of the APU detectors, shorting the two studs together without disconnecting the leads will light the APU FIRE light in the cockpit if essential dc bus power is on the aircraft.

4. Generator and Alternator Overheat Switches — Removal / Installation

CAUTION: BEFORE PERFORMING ANY OF THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to thermal switches. All are located in the accessory gearbox area in each nacelle in the exhaust duct from the generator or alternator cooling shroud.

(2) Disconnect plug.

(3) Unscrew thermal switch and remove from duct.

B. Installation

CAUTION: DO NOT OVER TORQUE AS IT COULD CHANGE TRIP POINT OF THERMAL SWITCH.

(1) Install switch with star washer. Torque to 20 to 25 inch-pounds

(2) Check circuit to cockpit prior to connecting plug to thermal switch by connecting two pins of plug with jumper.

(3) Energize essential dc bus.

(4) Check that associated light in the master warning panels light up.

(5) Remove electrical power.

(6) Connect electrical plug and safety.

5. **Radio Rack Overheat Switches — Removal / Installation**

A. Removal

(1) Gain access to radio rack overheat switches, located under radio rack, adjacent to outflow valves, entrance compartment, right side.

(2) Remove electrical plug.

(3) Remove thermal switch from mounting bracket.

B. Installation

(1) Install switch using washer and nut.

CAUTION: DO NOT OVER TORQUE AS IT COULD CHANGE TRIP POINT OF SWITCH.

(2) Torque 20 to 25 inch-pounds.

(3) Check circuit to cockpit before connecting plug by connecting two pins of plug with jumper at each switch individually.

(4) Energize essential dc bus.

(5) Radios hot light in master warning lights panel should light up when either plug is jumped. Check both plugs.

(6) Remove electrical power.

(7) Remove jumper wire and connect plugs to switches. Safety plugs.

(8) Close access.

6. **APU Fire Detector Switches — Removal / Installation**

A. Removal

(1) Gain access to APU enclosure through tail access door.

(2) Remove, identify and insulate wires from thermal switch terminals.

(3) Remove screws holding each switch to APU enclosure.

(4) Remove switch.

B. Installation

CAUTION: ENSURE 600° FAHRENHEIT SWITCH IS INSTALLED IN RIGHT TOP POSITION.

(1) Install switch into APU enclosure.

(2) Connect electrical leads as previously identified, ensuring good electrical connections.

(3) Energize essential dc bus; APU FIRE light should be off.

(4) Press and hold FIRE DET TEST button on eyebrow panel in cockpit.

(5) APU FIRE light should light up.

(6) Release TEST button.

(7) De-energize essential dc bus.

(8) Close tail access.

7. **Overheat Switch — Trip Point Check**

A. Radio Rack Overheat Switches.

(1) Check trip point of the radio rack overheat switches with suitable thermal switch tester.

(2) Trip point is 200 $\pm$ 10°F.

B. Generator / Alternator Overheat Switches.

(1) Check trip point of blower overheat switch with suitable thermal switch tester.

(2) Trip point is 300 $\pm$ 10°F.

C. Blower Overheat Switch

(1) Check trip point of blower overheat switch with suitable thermal switch tester.

(2) Trip point is 410 ± 10°F.

D. APU Fire Detector Overheat Switches.

(1) Using suitable thermal switch tester, check trip point of each APU detector switch.

(2) Trip points are as follows:

(a) Right bottom - 450 ± 25°F

(b) Right top - 600 ± 25°F

(c) Left top - 450 ± 25°F

8. Nacelle Fire Detector Control Unit (Kidde) — Removal / Installation

A. Removal

(1) Gain access to nacelle fire detector terminal panel, located in main gear wheel wells, aft section. Control unit is attached to this terminal box and it is necessary to remove entire terminal panel to remove control unit.

CAUTION: INSTALL SAFETY STRUTS IN CLAMSHELL DOORS WHILE WORKING IN WHEEL WELL.

(2) Remove mounting hardware attaching terminal box to structure.

(3) Remove terminal box assembly.

(4) Remove plug from fire detector control unit, mounted on outside of box.

(5) Remove control unit mounting hardware.

(6) Remove control unit.

B. Installation

(1) Mount control unit on terminal box.

(2) Inspect plug, receptacle and wire harness for damage and corrosion.

(3) Connect terminal plug to control unit and safety wire.

(4) Mount terminal box assembly to structure.

(5) Energize essential dc bus.

(6) In cockpit, push and hold FIRE DET TEST switch. AFT NAC FIRE lights should light up.

(7) Release TEST button. Lights should go out.

(8) De-energize electrical power.

9. Nacelle Fire Detector Elements (Kidde) — Removal / Installation

A. Removal

(1) Gain access to the element section to be removed by removing applicable gearbox and/or nacelle access panels.

(2) Remove hardware securing clamps and grommets to structure.

(3) Disconnect end couplings of element to be removed.

(4) Discard copper sealing washers from end couplings.

(5) Remove element section.

B. Installation

CAUTION: AVOID SHARP BENDS, EDGES, CRUSHING OR DENTING AND COLD WORKING ELEMENT WHEN ROUTING. ENSURE THAT ELEMENT IS NOT SQUEEZED IN CLAMP TANGS.

(1) Route element picking up clamps and grommets using same routing as element previously removed.

(2) Secure element with clamps, grommets and hardware.

CAUTION: DO NOT INSTALL MORE THAN ONE COPPER SEALING WASHER IN EACH END COUPLING.

(3) Install new copper sealing washers in each end coupling.

(4) Connect end couplings and torque 50 to 70 inch pounds.

(5) Safety couplings.

(6) Energize essential dc bus.

(7) Push and hold FIRE DET TEST Button. L and R NACELLE FIRE lights should light up on eyebrow panel indicating proper operation of control unit and loop continuity.

(8) Release TEST button. Lights should go out.

(9) De-energize electrical power.

(10) Close access previously opened.

10. Kidde Fire Detector Wire — Insulation Resistance Check

NOTE: All insulation resistance checks are to be measured between the center conductor and the outer shell. All readings are in megohms.

	Using 350 - 500V DC Megger			Using 10V DC Megger or less		
Element Temp. °F.	**90 Inch Section**	**130 Inch Section**	**Full Loop 310 Inch**	**90 Inch Section**	**130 Inch Section**	**Full Loop 310 Inch**
30	2.5	1.8	0.74	6.3	4.3	1.8
40	1.9	1.3	0.54	4.6	3.1	1.3
50	1.4	0.94	0.39	3.3	2.3	0.97
60	1.0	0.69	0.28	2.4	1.7	0.71
70	0.72	0.50	0.21	1.8	1.2	0.52
80	0.53	0.37	0.15	1.3	0.90	0.38
90	0.38	0.27	0.11	1.0	0.67	0.28
100	0.28	0.19	0.08	0.70	0.49	0.20

11. Kidde Detector Element Center Conductor — Continuity / Resistance Check

A. Determine length of element to be checked in inches.

NOTE: The system consists of two 90 inch and one 130 inch lengths.

B. With appropriate ohmmeter or resistance bridge, check element center conductor resistance end to end at room temperature.

C. The maximum resistance acceptable is one ohm per foot of length.

12. Fire T-Handle — Removal / Installation

A. Removal

(1) Remove floodlight panel under eyebrow panel in cockpit.

(2) Remove screw securing plastic handle to shank; remove handle.

(3) Remove edge light panel in area of FIRE PULL T-handle involved.

(4) Disconnect and identify leads from bank of switches of FIRE PULL T-handle assembly (leads are soldered to switch terminals); two leads are for warning light, disconnect and identify.

(5) Remove two screws securing handle assembly to eyebrow panel.

(6) Remove T-handle assembly.

B. Installation

(1) Install T-handle assembly with two screws.

(2) Solder leads to switch assemblies and connect warning light leads as previously identified; ensure good electrical connections.

(3) Install edge lighting panel.

(4) Install plastic T-handle over shank using screw.

(5) Pull and push FIRE PULL T-handle and check wire clearances.

(6) Energize main dc bus and check edge lighting panel operation, then de-energize main dc bus.

13. Fire T-Handle — Operational Test

A. Gain access to the following units: (per side)

(1) Main fuel shutoff valve, aft wing beam

(2) Water methanol shutoff valve - Remove box cover in main gear wheel wells

(3) Hydraulic shutoff valve - Remove floorboard 6 for right valve, floorboard 9 for left valve

B. Ensure FIRE EXT SHOT No. 1 and FIRE EXT SHOT No. 2 circuit breakers are pulled. (A/C having ASC 210)

C. Energize main and essential dc buses.

D. PULL left and right WATER METHANOL PUMP circuit breakers.

E. Place WATER METHANOL switch to ARMED.

F. Ensure all three valves for that T-handle are OPEN by flag indication.

G. Pull FIRE PULL T-handle. All three valves should cycle closed.

H. Reset FIRE PULL T-handle to in position. All three valves for that T-handle should cycle open.

I. Place WATER METHANOL switch off. (W/M valve will cycle closed)

J. Reset left and right WATER METHANOL PUMP circuit breakers.

K. Energize instrument ac bus.

L. Push and hold FIRE DET TEST button. Both FIRE PULL T-handles should light up.

M. Release FIRE DET TEST button. Both T-handles should go out.

N. De-energize instrument ac, main and essential dc buses.

O. Close cover on water methanol valve box in wheel wells.

P. Install floorboard 6 and 9.

Q. Operate T-handle several times with flood lighting panel removed from eyebrow panel and check wiring for clearance and general condition.

R. Replace flood lighting panel.

S. Ensure FIRE PULL T-handles are in.

T. Reset FIRE EXT SHOT No. 1 and FIRE EXT SHOT No. 2 circuit breakers.

14. Engine / Accessory Section Fire Extinguisher Spray Pipe — Inspection

A. Fire extinguisher pipe for chafing and security of clips and safety wire.

B. Check discharge holes for obstructions: apply 50 - 90 psi air pressure to discharge tube in main gear wheel well and check that all holes vent freely.

C. Clean blocked holes with No. 60 drill or straight 20 S.W.G. steel wire.

15. Blower Overheat Switch — Removal / Installation

A. Removal

(1) Gain access to blower overheat switch in right nacelle just forward of silencer. Remove access panel from side of nacelle.

(2) Remove electrical plug from thermal switch.

(3) Remove switch and star washer from duct boss.

B. Installation

(1) Install star washer on switch and install switch into duct boss.

CAUTION: DO NOT OVER TORQUE. THIS COULD CHANGE TRIP POINT OF SWITCH.

(2) Torque 20 to 25 inch-pounds.

(3) Check circuit to cockpit before connecting plug, by connecting two pins of plug with jumper.

(4) Energize essential dc bus.

(5) BLOWER HOT light in master warning lights panel should light up.

(6) De-energize essential dc bus.

(7) Remove jumper wire and connect plug to switch; safety wire plug.

(8) Close nacelle access.

16. Graviner Sensing Elements — Supplemental Information

A. Check continuity of the elements as follows:

(1) Determine part number by checking appropriate end fitting (number stamped on one end fitting only for each length).

(2) Use values shown below, to determine total center conductor resistance.

NOTE: The continuity resistance of the individual elements is generally well within these limits.

Part Number	Mark	Continuity (Ohms)
D890	I	15 - 25
D890/N	II	10 - 16.25
D4176/N		10.3 - 17 See NOTE A
D1384	I	7.5 - 12.5
D1384/N	II	5.0 - 8.0
D889	I	5.0 - 8.5
D889/N	II	3.3 - 5.5
D5352N/1		1.0 - 1.3 See NOTE B

NOTE:

A. D4176/N is substitute for D890 or D890/N. It is 2 inches longer.

B. D5352N/1 is the new 250°C trip point breather element which replaces the D1870 and D1870/N.

B. Insulation resistance check of Graviner fire detector wire and interconnecting fittings:

NOTE: All insulation resistance checks are to be measured between the center conductor and outer shell, with elements at room temperature.

(1) Any single piece of element, regardless of length shall indicate a minimum of 20 Megohms with a 250 volt insulation tester.

(2) The entire loop, consisting of all elements connected to each other plus all interconnecting fittings, with the loop disconnected from the control unit must indicate 1 Megohm minimum with 250 volt insulation tester.

(3) Interconnecting, termination and bulkhead fittings, not connected to the loop must indicate 20 Megohms minimum with either a 250 or 500 volt insulation tester.

17. Engine Fire Detector Relay Box (Graviner) — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Gain access located in main gear wheel well.

(2) Disconnect three electrical plugs from box.

(3) Remove mount and box as unit from structure.

(4) Remove hardware securing box to mount bracket.

(5) Remove box.

B. Installation

(1) Mount box to bracket.

(2) Mount box and bracket assembly to structure.

(3) Connect electrical plugs.

(4) Energize essential dc and instrument ac buses.

(5) Push FIRE DET TEST switch; FIRE PULL T-HANDLES should light up, indicating box operation.

(6) Release TEST switch; light should go out.

(7) De-energize essential dc and instrument ac buses.

18. Engine Fire Detection Elements (Graviner) — Removal / Installation

A. Removal

(1) Gain access to engine fire detector elements by opening engine cowling forward of firewall.

(2) Remove hardware securing element clamps to engine. Retain clamps, grommets and hardware.

(3) Disconnect element and couplings.

(4) Remove element section.

(5) Discard copper crush washers from opened coupling.

B. Installation

CAUTION: AVOID SHARP BENDS, EDGES, CRUSHING OR DENTING AND COLD WORKING ELEMENT WHEN ROUTING. ENSURE THAT ELEMENT IS NOT SQUEEZED IN CLAMP TANGS AND IS SUPPORTED EVERY 9 TO 12 INCHES TO PREVENT CHAFING.

(1) Route element, picking up clamps and grommets using the same routing as element previously removed.

(2) Secure element with clamps, grommets and hardware.

CAUTION: DO NOT USE MORE THAN ONE COPPER SEALING WASHER IN EACH END COUPLING.

(3) Install new copper sealing washer in each end coupling.

(4) Connect end couplings and torque 80 to 100 inch pounds.

(5) Safety couplings.

(6) Energize essential dc and instrument ac buses.

(7) In cockpit, depress FIRE DET TEST button. FIRE PULL T-handles will light up, indicating proper operation of control unit and element continuity.

(8) Release TEST button; lights should go out.

(9) Remove electrical power.

(10) Inspect area for foreign objects, security of all attachments.

(11) Close cowling.

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FIRE EXTINGUISHER SYSTEM — DESCRIPTION / OPERATION

1. General

A. The nacelle and APU fire extinguishers must be removed and weighed in accordance with inspection times listed in Chapter 5 and utilizing data listed on the container nameplate.

CAUTION: IF A FIRE EXTINGUISHER IS DISCHARGED INTO A NACELLE OR AN APU ENCLOSURE WITH ENGINE OR APU IN A HOT STATE, MAKE CAREFUL INSPECTION OF ENGINE OR APU BEFORE PLACING IT BACK IN SERVICE AS CF_3BR IS EXTREMELY COLD AND MAY CAUSE FRACTURES ON HOT METALLIC SURFACES. IF SYSTEM IS DISCHARGED, CONTAINER(S) MUST BE REPLACED IMMEDIATELY. NO PURGING IS REQUIRED.

B. The dual squib cartridges must be replaced in accordance with times listed in Chapter 5.

CAUTION: IF A CARTRIDGE IS REMOVED FROM ITS BONNET, DO NOT REINSTALL THAT CARTRIDGE INTO ANOTHER BONNET. HOWEVER, IT MAY BE REINSTALLED INTO THE SAME BONNET FROM WHICH IT WAS REMOVED.

2. Nacelle Fire Extinguisher System

(See Figure 1 and Figure 2)

A two-shot fire extinguisher is provided to combat engine/nacelle fires. This system employs two fire extinguisher containers (bottles), one of which is installed in the right nacelle and one in the left nacelle at approximately Station 320. Each bottle contains 9.5 pounds by weight of CF_3Br (Bromotrifluoromethane) charged with nitrogen to approximately 600 psi at 70°F. See Chapter 12 SERVICING for pressure vs temperature relationship). The system is so arranged, that the extinguishing agent in each bottle can be discharged into the compartment of either nacelle. Two fire extinguisher discharge switches are provided in the cockpit, one for the right engine/nacelle and the other for the left engine/nacelle. They are guarded by the upper part of the FIRE PULL T-handles, and are accessible only if the associated FIRE PULL T-handle is pulled.

Red and yellow discharge indicator disks are installed on the outboard side of the right nacelle and on the inboard side of the left nacelle at Station 332. The yellow disk ejects when the container is discharged through the system to its own side engine/nacelle only. The red disk of the thermal discharge indicator ejects when the container discharges overboard due to excessive internal pressure caused by high nacelle temperature.

Each bottle incorporates a pressure gage which is easily visible through a transparent window in the side of the nacelle above the wing. This gage which is calibrated in pounds per square inch will indicate the pressure inside of the bottle. The bottle, when charged, is pressurized to 600 + 50, − 25 psig at 70°F. The bottle cannot be filled in the field. It is therefore imperative that the crew and maintenance Personnel determine that the bottles are full and ready for use, if required, prior to flight. The pressure gages will be used for this purpose. If a bottle has been dischargered, that gage should read zero.

NOTE: As any compressed gas, such as Nitrogen, reacts to the temperature, it is well to remember that the 600 PS pressure is at 70° If the temperature Surrounding the bottle is higher, the pressure will be Proportionally higher. If the temperature is lower, the pressure will be proportionally lower.

A good reference as to bottle pressure is the opposite bottle gage, providing neither has been discharged. They should agree closely.

When discharging a fire bottle through crossover, there is no disc ejection. The only method of determining if a container is full, is by checking container pressure gage.

Each extinguisher bottle will discharge its entire content when either of its two bonnets are fired. Where the agent goes depends on which bonnet is fired. Of the two bonnets on each bottle, one will direct the agent into its own side engine/nacelle fire zone, and the other will direct it to the opposite engine/nacelle via a crossover system. Therefore, in the event of a fire in either engine/nacelle both bottles can be utilized, if required (its own side bottle, plus the opposite side bottle through the crossover line).

It is virtually impossible for the crew to discharge an extinguisher to the wrong engine. This is accomplished by location and accessibility of the engine/nacelle fire extinguisher switches, and the associated electrical circuitry Pertaining to each bottle system.

The fire extinguisher Switches are located in the glareshield panel in the cockpit. The upper part of the FIRE PULL T-handle forms a guard which prevents actuation of the associated fire extinguisher switch toggle unless the FIRE PULL T-handle is pulled. Once the T-handle is pulled, the associated fire extinguisher switch for that side is then accessible. Each fire extinguisher switch has three positions: 1-OFF-2, with the switch toggle operating laterally, and the center position of the switch toggle being the OFF position.

The placarding of 1 and 2 for each switch signifies which bottle will be fired into that particular engine/nacelle system (its own side bottle, or the opposite bottle through the crossover). Therefore either position of the switch will fire a bottle into the engine/nacelle fire zone which is involved. Obviously, since there are two bottles, if the switch is actuated to both the 1 and 2 positions, there is no further agent available, as both bottles have been fired.

Each bonnet acts not only as a discharge valve but as a plumbing connection (See Figure 1). As an example, if there is a left engine fire, the crew will pull the left FIRE PULL T-handle, this will expose the L FIRE EXT switch. The switch is then moved into the SHOT No. 1 position (See Figure 3). This directs electrical energy from the essential DC bus through the FIRE EXT SHOT No. 1 circuit breaker through the switch to the No. 1 (same side) bonnet of the left fire bottle. The cartridge, fires and the agent is propelled from the bottle by the nitrogen charge into the bonnet. On the side of each bonnet is a tubing connection. The No. 1 bonnet tubing is routed to the double check tee. This is a shuttle valve, which by the pressure of the agent and the nitrogen, will move the shuttle, blocking the crossover line from receiving any agent, and opening the center port. This allows the agent to be discharged into the left engine/nacelle fire zone. Thus, the No. 1 shot fired the left fire bottle into the left engine.

In the event more agent is required, the crew would then actuate the L FIRE EXT switch to the No. 2 position (See Figure 3). This would then route the electrical energy from the essential dc bus, through the FIRE EXT SHOT No. 2 circuit breaker through the switch to the No. 2 (crossover) bonnet of the opposite (right) bottle. The No. 2 bonnet of the right fire bottle would fire, discharge the agent into its crossover line, to the left system double check tee. The pressure would then move the shuttle, blocking the line to the bottle. This allows the agent to be discharged into the left engine/nacelle fire zone. Thus, the No. 2 shot is always opposite bottle, through crossover, to the side involved.

NOTE: Aircraft 1 - 200, 322 and 323 not having ASC 210, circuit breakers are placarded R NAC FIRE EXT and L NAC FIRE EXT.

As can be seen, if the right engine were involved instead of the left engine, the actuation of its fire extinguisher switch would accomplish the same end result for the right engine, that is shot No. 1-right bottle to right engine. Shot No. 2-left bottle through crossover to right engine.

3. APU Fire Extinguisher System — Description / Operation

(See Figure 3 and Figure 4)

The APU fire extinguishing system consists of a single shot, electrically fired extinguisher bottle with associated tubing and electrical components. The bottle is installed in the tail compartment. It is accessible through the tail access door. (See Figure 4)

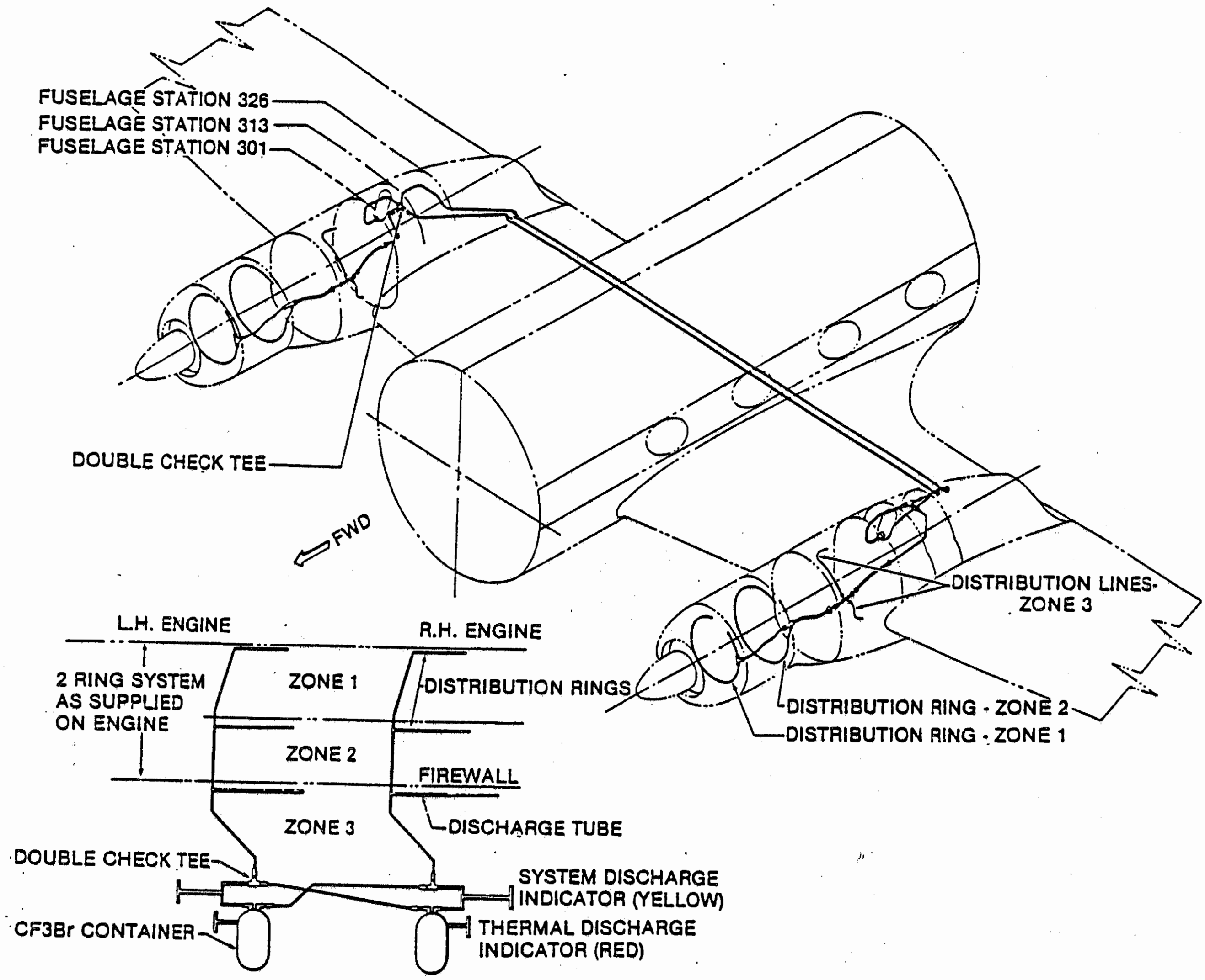
The bottle incorporates a single bonnet with an electrically fired, dual squib cartridge which is identical to those used in the engine bottles.

Red and yellow discharge indicator disks are installed on the right side of the fuselage at approximately Fuselage Station 625 (See Figure 4). These indicators provide a means of checking the condition of the fire extinguisher system. The indicators perform the same function as those in the nacelle fire extinguisher system.

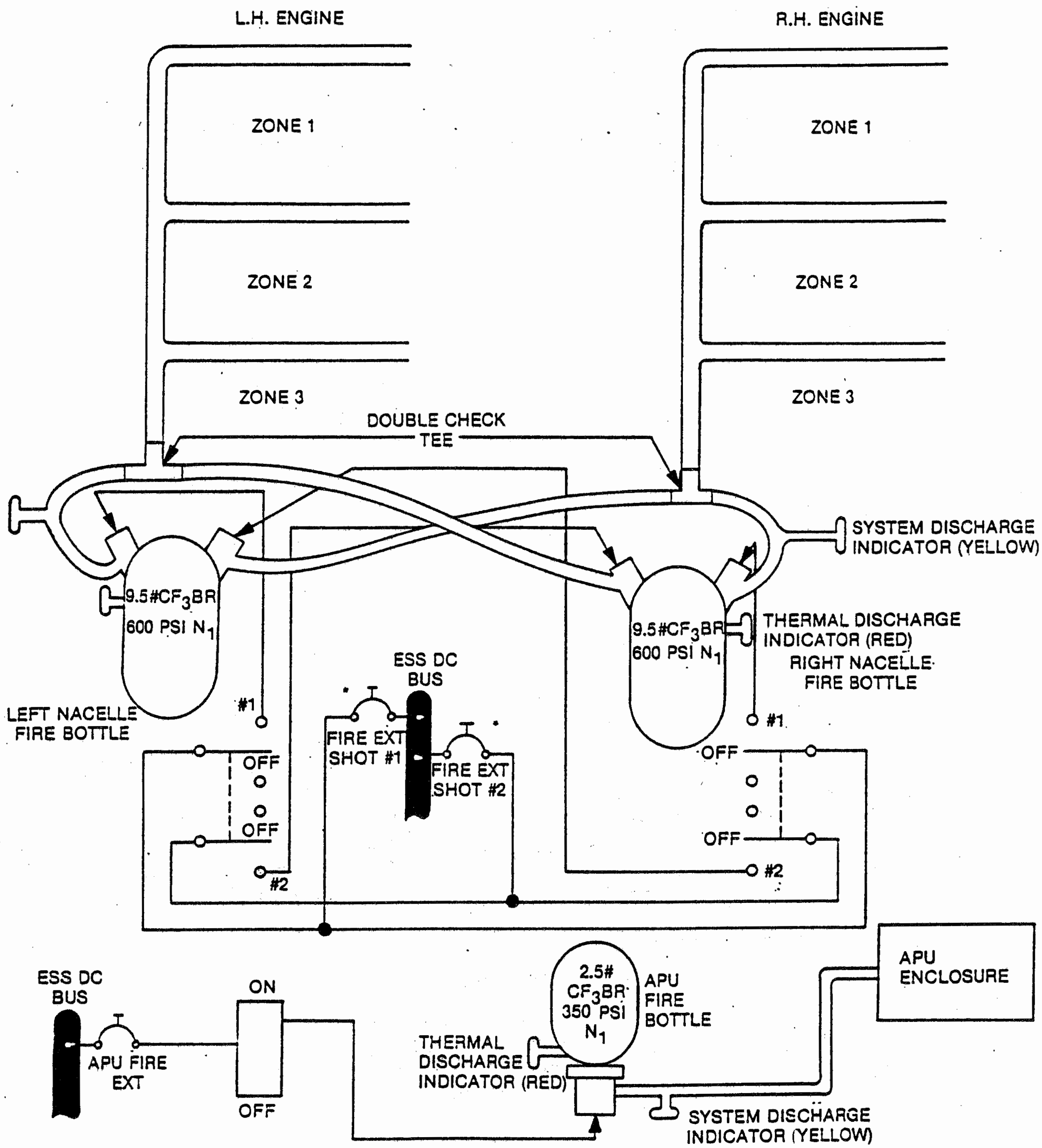
A pressure gage is installed on the container, and is clearly visible from inside of the tail compartment. It cannot, however, be seen from outside of the aircraft, therefore, crew and maintenance personnel should view this gage during preflight inspections to ascertain that a full APU bottle is available for use. The fully charged bottle is 350 + 25 − 0 psi at 70°F. A discharged bottle reads 0 psi.

The APU extinguisher bottle cannot be refilled in the aircraft, as this requires special equipment and can be accomplished only at an authorized facility.

The APU fire extinguisher is controlled by the APU EXT switch, located on the glareshield panel. It is a guarded switch with two positions, ON and OFF. Lifting the guard and moving the switch to the ON position routes electrical energy from the essential dc bus, through the APU EXT circuit breaker, through the switch contacts, directly to the single bonnet on the fire bottle (See Figure 3). The bottle discharges into a single stainless steel line which carries the agent into the APU enclosures, thus surrounding the unit with extinguishing agent.

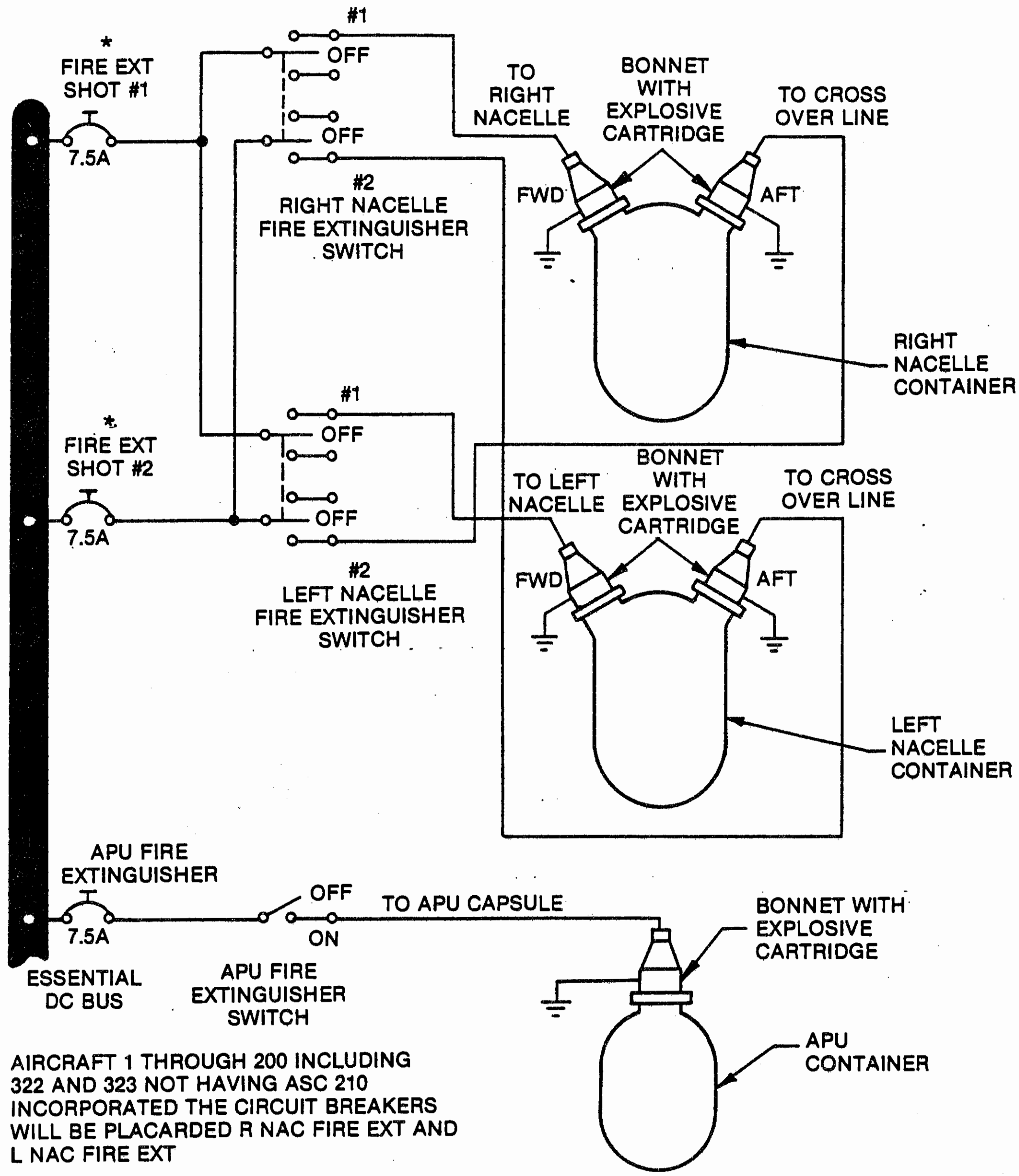


Nacelle Fire Extinguisher System — General Arrangement
Figure 1.



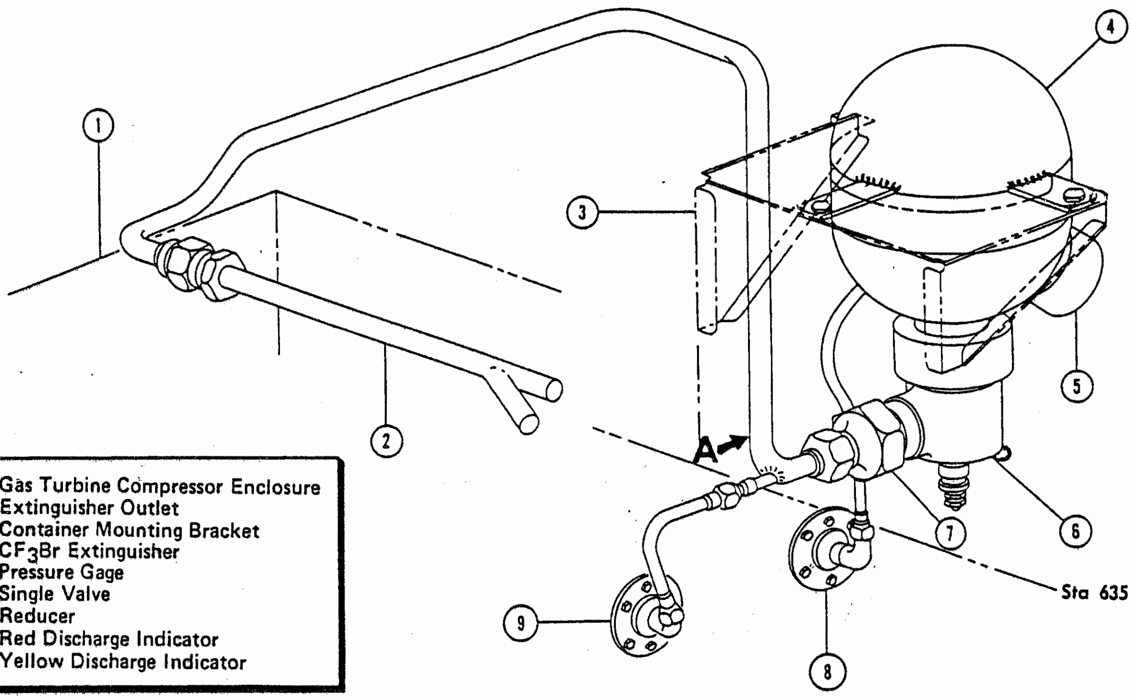
* AIRCRAFT 1 THROUGH 200 INCLUDING 322 and 323
NOT HAVING ASC 210, CIRCUIT BREAKERS ARE PLACARDED
R NAC FIRE EXT AND L NAC FIRE EXT

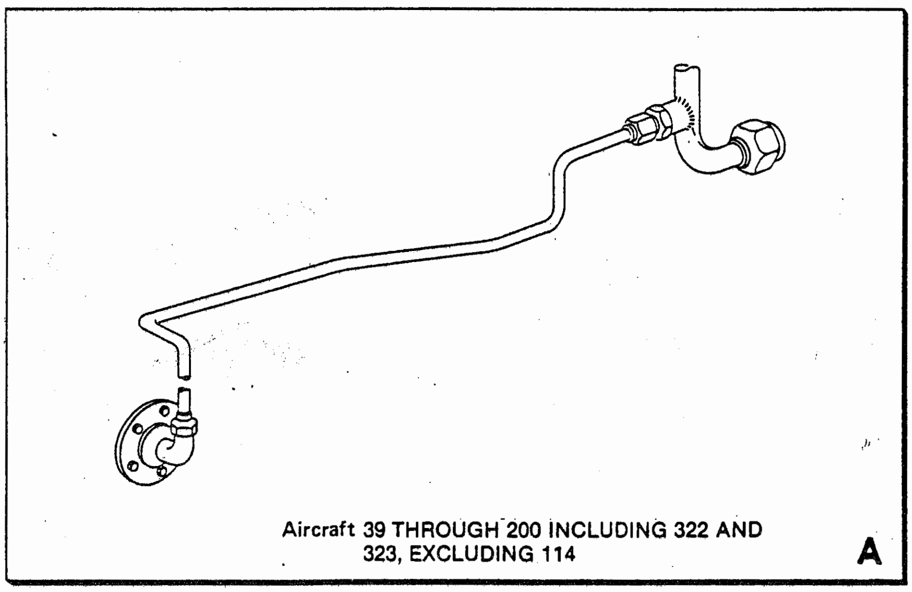
Engine and APU Fire Extinguisher System — Block Diagram
Figure 2.



* AIRCRAFT 1 THROUGH 200 INCLUDING 322 AND 323 NOT HAVING ASC 210 INCORPORATED THE CIRCUIT BREAKERS WILL BE PLACARDED R NAC FIRE EXT AND L NAC FIRE EXT

APU and Nacelle Fire Extinguisher System — Simplified Schematic
Figure 3.





APU Fire Extinguisher System
Figure 4.

FIRE EXTINGUISHER SYSTEM — FAULT ISOLATION

NOTE: Circuit breakers are placarded R NAC FIRE EXT and L NAC FIRE EXT on Aircraft 1 - 200, 322 and 323 not having ASC 210.

FAULT	POSSIBLE CAUSE	CORRECTION
Right nacelle fire extinguisher switch is positioned from off to 1, and no CF_3Br is discharged	Right nacelle container empty.	Recharge container
	Discharge valve defective.	Replace defective discharge valve.
	Tripped FIRE EXT SHOT No. 1 circuit breaker. (See NOTE.)	Check for reason why circuit breaker tripped. Close circuit breaker
	Defective switch.	Replace switch.
Right nacelle FIRE extinguisher switch is positioned from off to 2 and no CF_3Br is discharged.	Left nacelle container empty.	Replace container.
	Discharge valve defective.	Replace defective discharge valve.
	Tripped FIRE EXT SHOT No. 2 circuit breaker. (See NOTE.)	Check for reason why circuit breaker tripped. Close circuit breaker.
	Defective switch.	Replace switch.
Left nacelle FIRE extinguisher switch is positioned from off to 1, and no CF_3Br is discharged.	Left nacelle container empty.	Recharge container.
	Discharge valve defective.	Replace defective discharge valve.
	Tripped FIRE EXT SHOT No. 1 circuit breaker.	Check for reason for circuit breaker opening. Close circuit breaker.
	Defective switch.	Replace switch.
Left nacelle FIRE extinguisher switch is positioned from off to 2, and no CF_3Br is discharged.	Right nacelle container empty.	Recharge container.
	Discharge valve defective.	Replace defective discharge valve.
	Tripped FIRE EXT SHOT No. 2 circuit breaker.	Check reason for tripping. Close circuit breaker.
	Defective switch.	Change switch.
CF_3Br discharge is not in location selected.	CF_3Br lines incorrectly routed.	Reroute lines.
	System incorrectly wired.	Check circuit continuity, and rewire as required.
Loss of CF_3Br, indicated by blown red disc.	Excessive pressure causing discharge from cylinder safety outlet.	Recharge container to correct pressure (weight). Replace safety disc.
	Defective safety disc. (improperly torqued)	Recharge container and replace safety disc.
APU EXT SWITCH is positioned from OFF to ON and no CF_3Br is discharged.	APU container empty.	Recharge container.
	Defective discharge valve.	Replace defective discharge valve.
	Tripped APU FIRE EXT circuit breaker.	Check for reasons why circuit breaker tripped. Close circuit breaker.
	Defective switch.	Replace switch.

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FIRE EXTINGUISHER SYSTEM — MAINTENANCE PRACTICES

NOTE: The fire extinguisher bottles in this aircraft cannot be recharged in the aircraft. Removal and replacement with a serviceable bottle is necessary, as the recharging of the bottles using CF_3Br and nitrogen requires special equipment available only at an authorized agency equipped to perform recharging. The containers conform to ICC-4DA99. The containers are charged with bromotrifluoromethane, military specification MIL- B-12218, and are pressurized with nitrogen, MIL-N-60M.

Maintenance practices which follow are those which are normally performed during bottle replacement if a bottle is fired and during maintenance performed at check periods as specified in the Chapter 5 this manual.

1. **Engine Fire Bottle — Removal / Installation**

(See Figure 201)

WARNING: EACH NACELLE FIRE EXTINGUISHER BOTTLE CONTAINS TWO ELECTRICALLY FIRED CARTRIDGES, ONE IN EACH BONNET (VALVE). THESE CARTRIDGES EACH CONTAIN A SLUG APPROXIMATELY THE SIZE OF A .45 CALIBER PROJECTILE. CARTRIDGES ARE ELECTRICALLY FIRED, NOT PERCUSSION FIRED. INADVERTENT APPLICATION OF ELECTRICAL ENERGY TO A FIRING HEAD WILL CAUSE THE CARTRIDGE TO FIRE. IF THE CARTRIDGE IS INADVERTENTLY FIRED WHEN THE BONNET IS REMOVED FROM THE BOTTLE, FATAL OR SERIOUS INJURY CAN RESULT. IF A BOTTLE IS FIRED, REMEMBER THAT THERE IS ANOTHER UNFIRED CARTRIDGE IN THE OTHER BONNET OF THE DISCHARGED BOTTLE.

TO PREVENT ACCIDENTAL DISCHARGE DURING MAINTENANCE, PULL THE TWO CIRCUIT BREAKERS MARKED FIRE EXT SHOT NO. 1 AND FIRE EXT SHOT NO. 2 (AIRCRAFT 1 - 200, 322 AND 323 NOT HAVING ASC 210. THE CIRCUIT BREAKERS ARE PLACARDED R NAC FIRE EXT AND L NAC FIRE EXT), LOCATED ON THE PILOTS CIRCUIT BREAKER PANEL. AS AN ADDED PRECAUTION DURING MAINTENANCE, IT IS RECOMMENDED THAT AN ELECTRICAL SHUNT BE INSTALLED FROM THE CARTRIDGE ELECTRICAL TERMINAL STUD TO THE CARTRIDGE GROUND OR TO THE FRAME OF THE BONNET IF THE CARTRIDGE IS INSTALLED IN THE BONNET.

A. Removal

(1) Disconnect and identify electrical leads from bonnet.

(2) Disconnect supply lines at B-nuts.

(3) Disconnect flexible thermal discharge line at fitting.

(4) Support bottle and remove bolts holding bottle to support bracket.

(5) Remove the bottle assembly.

(6) Perform weight check and/or hydrostatic test (if due).

NOTE: Individual assembled bottle weights (including bonnet and cartridge) should be obtained from the data plate on the bottle. The nitrogen charge is 600 psi + 25, − 0 at 70°F.

B. Installation

(1) Ensure that circuit breakers marked FIRE EXT SHOT No. 1 and FIRE EXT SHOT No. 2 are pulled. Fire extinguisher switches concealed by the FIRE PULL T-HANDLE must be in the OFF (Center) position.

NOTE: On Aircraft 1 - 200, 322 and 323 not having ASC 210. The circuit breakers are placard R NAC FIRE EXT and L NAC FIRE EXT.

(2) Before installation, check that extinguisher to be installed has following:

(a) Cartridge installed has valid service life and torqued to 175 inch-pounds. Record date stamped on slug and date installed.

(b) Valid weight check. Record date of weight check.

(c) Valid hydrostatic test date. Record date of last hydrostatic test.

(d) Check pressure gage for proper pressure (600 psi at 70° Fahrenheit).

(3) Ensure bonnets are installed with their swivel nuts finger tight.

CAUTION: DO NOT DAMAGE CARTRIDGE TERMINAL STUDS DURING INSTALLATION.

(4) Position bottle on bracket and secure in place using hardware removed in Step A.(4) above.

(5) Connect flexible thermal discharge line to fitting on bottle.

(6) Orient bonnet tubing connection with respect to its supply line. Connect supply line to tubing connection on each bonnet and tighten supply line B-nut.

(7) Tighten swivel nut on bonnets with strap wrench to approximately 75 foot-pounds.

WARNING: DO NOT PERFORM CONTINUITY CHECK UNTIL BONNET IS INSTALLED AND CONTAINER IS CONNECTED TO AIRCRAFT SYSTEM.

(8) Check bonnet installation as follows:

(a) Measure gap, should be 0.023 to 0.285 inches (See Figure 203).

NOTE: If gap is less than 0.023 or greater than 0.285 inches and continuity exists, bonnet/cartridge assembly is correct.

(b) Check for electrical continuity between bonnet and cartridge using a AN31121-1819 lamp, through a switch, to a 24 to 28V dc source.

NOTE: In lieu of using test circuit shown, any circuit may be used if it limits current through bonnet/cartridge to 0.035 ampere.

If continuity does not exist, remove bonnet and inspect installation of cartridge. Correct as required and repeat Step 8 above.

(9) Connect electrical leads as previously identified and ensure good connections.

(10) Place boots over terminal stud on each bonnet.

(11) Close circuit breakers.

2. Engine Fire Bottle — Cartridge Replacement

WARNING: EACH NACELLE FIRE EXTINGUISHER BOTTLE CONTAINS TWO ELECTRICALLY FIRED CARTRIDGES, ONE IN EACH BONNET (VALVE). THESE CARTRIDGES EACH CONTAIN A SLUG APPROXIMATELY THE SIZE OF A .45 CALIBER PROJECTILE. CARTRIDGES ARE ELECTRICALLY FIRED, NOT PERCUSSION FIRED. INADVERTENT APPLICATION OF ELECTRICAL ENERGY TO A FIRING HEAD WILL CAUSE THE CARTRIDGE TO FIRE. IF THE CARTRIDGE IS INADVERTENTLY FIRED WHEN THE BONNET IS REMOVED FROM BOTTLE, FATAL OR SERIOUS INJURY CAN RESULT. IF A BOTTLE IS FIRED, REMEMBER THAT THERE IS ANOTHER UNFIRED CARTRIDGE IN THE OTHER BONNET OF THE DISCHARGED BOTTLE.

TO PREVENT ACCIDENTAL DISCHARGE DURING MAINTENANCE, PULL THE TWO CIRCUIT BREAKERS MARKED FIRE EXTINGUISH SHOT NO. 1 AND FIRE EXT SHOT NO. 2 (AIRCRAFT 1 - 200, 322 AND 323 NOT HAVING ASC 210 THE CIRCUIT BREAKERS ARE PLACARDED R NAC FIRE EXT AND L NAC FIRE EXT), LOCATED ON THE PILOTS CIRCUIT BREAKER PANEL. AS AN ADDED PRECAUTION DURING MAINTENANCE, IT IS RECOMMENDED THAT AN ELECTRICAL SHUNT BE INSTALLED FROM THE CARTRIDGE ELECTRICAL TERMINAL STUD TO THE CARTRIDGE GROUND OR TO THE FRAME OF THE BONNET IF THE CARTRIDGE IS INSTALLED IN THE BONNET.

A. Remove fire bottle. See Engine Fire Bottle — Removal / Installation, this Section.

B. Loosen swivel nut and remove bonnet from container. Use strap wrench to loosen swivel nut.

WARNING: CARTRIDGES ARE CLASS C EXPLOSIVES AND MUST BE DISPOSED OF IN ACCORDANCE WITH REGULATIONS GOVERNING DISPOSITION OF CLASS C EXPLOSIVES.

C. Remove cartridge using 1-inch deep socket wrench. Discard cartridge.

D. Remove and discard O-ring from neck of container.

E. Check the following on replacement cartridge:

(1) Check date stamped on slug.

NOTE: Service life is 24 months providing shelf and service life of cart ridges do not exceed 72 months from date on cartridge.

(2) Check all crimped areas on cartridge to determine that crimped parts are secure and they do not rotate.

(3) Check threads for distortion or damage.

(4) Check soldered contact for damage.

F. Apply grease (MIL-G-7711) to cartridge threads and install into bonnet. Torque to 175 inch-pounds.

(1) Install new O-ring in container neck.

(2) Apply small quantity of grease (MIL-G-7711) to threads of container neck.

(3) Ensure that O-ring is in place and assemble bonnet on container; tighten nut by hand. Nut will be torqued when container is installed.

(4) Repeat procedure for other cartridge if both are to be replaced.

(5) Install fire bottle. (See Engine Fire Bottle — Removal / Installation, this Section.

3. **Fire Bottle — Hydrostatic Test**

A. Remove Fire Bottle. (See Engine / APU Fire Bottle — Removal / Installation, this Section)

B. Preform hydrostatic test

NOTE: Hydrostatic test must be performed by a certified overhaul agency.

C. Replace cartridges if required. (See Engine / APU Bottle Cartridge — Replacement)

D. Install Fire Bottle. (See Engine / APU Fire Bottle — Removal / Installation this section)

4. **Fire Bottle — Weight Check**

A. Remove Fire Bottle. (See Engine / APU Fire Bottle — Removal / Installation this section)

B. Preform weight check.

C. Replace cartridges if required. (See Engine / APU Bottle Cartridge — Replacement)

D. Install Fire Bottle. (See Engine / APU Fire Bottle — Removal / Installation this section)

5. **Fire Extinguisher System — Functional Test**

A. Engine Extinguisher System.

NOTE: This is a service check which does not discharge the fire bottles. The equipment required is four 28V test lights appropriately identified with suitable insulated alligator clips to connect to positive lead of each bonnet, and to ground.

On Aircraft 1 - 200, 322 and 323 not having ASC 210. The circuit breakers are placarded R NAC FIRE EXT and L NAC FIRE EXT

(1) Ensure that all fire extinguisher switches are OFF and FIRE EXT SHOT No. 1 and No. 2 circuit breakers are pulled (pilots circuit breaker panel).

(2) Disconnect the two positive leads from each bonnet cartridge terminal stud, and tag each lead as to the bonnet from which it was disconnected. (Forward bonnet is No. 1, and the rear bonnet is No. 2 in each case.)

CAUTION: ENSURE THAT POSITIVE LEADS ARE NOT TOUCHING BONNET CARTRIDGE TERMINAL STUD OR GROUND. INSULATED ALLIGATOR CLIPS ARE RECOMMENDED FOR THE TEST LIGHTS.

(3) Connect a 28V dc test lamp between each positive lead and ground.

(4) Energize essential dc bus.

(5) Close both FIRE EXT SHOT No. 1 and No. 2 circuit breakers.

(6) Pull both FIRE PULL T-handles to expose extinguisher switches.

NOTE: The following test will require two men, one in the cockpit to operate the switches and one outside to observe both left and right test lights.

(7) Working first with the left extinguisher switch move it to shot No. 1 position, then to shot No. 2 position then to off as reflected in the following table. Circuits are functioning properly if the proper test lamps are energized in the order shown. Then proceed with the right extinguisher switch as specified on the following table:

NOTE: *Denotes appearance of electrical power at light.

SWITCH	POSITION	EXTINGUISHERS			
		LEFT BOTTLE		RIGHT BOTTLE	
		No. 1 BONNET	No. 2 BONNET	No. 1 BONNET	No. 2 BONNET
LEFT EXT. SWITCH	OFF	NONE	NONE	NONE	NONE
	No. 1	*	NONE	NONE	NONE
	No. 2	NONE	NONE	NONE	*
	OFF	NONE	NONE	NONE	NONE
RIGHT EXT. SWITCH	OFF	NONE	NONE	NONE	NONE
	No. 1	NONE	NONE	*	NONE
	No. 2	NONE	*	NONE	NONE
	OFF	NONE	NONE	NONE	NONE

(8) Ensure that both switches are OFF and push in FIRE PULL T-handles.

(9) Remove electrical power from aircraft if no longer required.

(10) Pull FIRE EXT SHOT No. 1 and No. 2 circuit breakers.

(11) Remove test lamps and reconnect positive leads to appropriate bonnet cartridge terminal studs ensuring good electrical connection. Place boots over positive leads.

(12) Close circuit breakers.

B. APU Fire Extinguishing System.

NOTE: The following is a functional test of the APU fire extinguishing system which tests the electrical function without firing the cartridge. The equipment required is one 28V test lamp with suitable insulated alligator clip to connect to the positive lead of the bonnet and to ground.

(1) Ensure APU FIRE EXT switch is off and APU FIRE EXT circuit breaker is pulled.

(2) Disconnect ground and positive leads. (See Figure 202)

(3) Connect a 28V dc test light between positive lead and ground.

(4) Energize essential dc bus.

(5) On pilots circuit breaker panel, depress the APU FIRE EXT. circuit breaker.

(6) Place APU FIRE EXT switch ON. Test light should come on.

(7) Place APU FIRE EXT switch OFF. Test light should go out.

(8) Pull APU FIRE EXT circuit breaker and remove electrical power.

(9) Connect positive lead to bonnet and ensure good connection at terminal stud and ground.

(10) Close circuit breakers.

6. APU Fire Bottle — Cartridge Replacement

A. Remove fire bottle from aircraft. See APU Bottle — Removal / Installation this section.

B. Loosen swivel nut, using a strap wrench, and remove bonnet from container.

WARNING: CARTRIDGES ARE CLASS "C" EXPLOSIVES AND MUST BE DISPOSED OF IN ACCORDANCE WITH REGULATIONS GOVERNING THE DISPOSITION OF CLASS "C" EXPLOSIVES.

C. Remove cartridge, using 1 inch deep socket. Discard cartridge.

D. Remove and discard O-ring from neck of container.

E. Check following on replacement cartridge:

(1) Check date stamped on slug.

NOTE: Service life is 24 months providing shelf and service life of cart ridge does not exceed 72 months from date on cartridge.

(2) Check all crimped areas on cartridge to determine crimped parts are secure and do not rotate.

(3) Check threads for disposition and damage.

(4) Check soldered contact for damage.

F. Apply grease (MIL-G-7711) to cartridge threads and install cartridge into bonnet. Tighten to 175 inch-pounds.

G. Install new O-ring in container neck.

H. Apply light coat of grease (MIL-G-7711) to threads of container neck.

I. Ensure O-ring is in place and assemble bonnet on container; tighten nut hand tight. Nut will be torqued when container is installed.

J. Install fire bottle. See APU Bottle — Removal / Installation, this Section.

7. APU Fire Bottle — Removal / Installation

(See Figure 201)

WARNING: THE APU FIRE BOTTLE IS ELECTRICALLY FIRED. TO PREVENT ACCIDENTAL DISCHARGE DURING MAINTENANCE, IT IS RECOMMENDED THAT THE CIRCUIT BREAKER MARKED APU FIRE EXT. LOCATED ON THE PILOTS CIRCUIT BREAKER PANEL IN THE COCKPIT, BE PULLED. AS AN ADDED PRECAUTION DURING MAINTENANCE, IT IS RECOMMENDED THAT AN ELECTRICAL SHUNT BE INSTALLED FROM THE CARTRIDGE ELECTRICAL TERMINAL STUD TO THE CARTRIDGE GROUND OR TO THE FRAME OF THE BONNET IF THE CARTRIDGE IS INSTALLED IN THE BONNET.

A. Removal

(1) Disconnect electrical leads from bonnet.

(2) Disconnect supply line from the bonnet.

(3) Disconnect flexible thermal discharge line at fitting.

(4) Remove bolts holding bottle to bracket.

(5) Remove container from aircraft.

(6) Perform weight check and/or hydrostatic test (if due).

NOTE: Individual assembled bottle weights (including bonnet and cartridge) should be obtained from the data plate on the bottle. The nitrogen charge is 350 psi + 25, − 0 at 70°F.

B. Installation

(1) Ensure that circuit breaker marked APU FIRE EXT is pulled, and the APU FIRE EXT switch on the glareshield panel in the cockpit is OFF (guard down).

(2) Before installation, check that fire bottle to be installed has following:

(a) Cartridge installed has valid service life. Record date stamped on slug and date installed.

(b) Valid weight check. Record date of weight check.

(c) Valid hydrostatic test date. Record date of last hydrostatic test.

(d) Check pressure gage for proper pressure (350 psi at 70° Fahrenheit).

(3) Install fire bottle using mounting bolts previously removed. Tighten bolts securely.

(4) Connect flexible thermal discharge line to fitting on bottle.

(5) Orient bonnet and mate fitting with supply line B-nut; tighten B-nut.

(6) Tighten swivel nut on bonnet with strap wrench to approximately 75 foot-pounds.

WARNING: DO NOT PERFORM CONTINUITY CHECK UNTIL BONNET IS INSTALLED AND CONTAINER IS CONNECTED TO AIRCRAFT SYSTEM.

(7) Check bonnet installation as follows;

(a) Measure gap, should be 0.023 to 0.285 inches (See Figure 203).

NOTE: If gap is less than 0.023 or greater than 0.285 inches and continuity exists bonnet/cartridge assembly is correct.

(b) Check for electrical continuity between bonnet and cartridge using a AN31121-1819 lamp, through a switch, to a 24 to 28V dc source.

NOTE: In lieu of using test circuit shown, any circuit may be used if it limits current through bonnet/cartridge to 0.035 ampere.

If continuity does not exist, remove bonnet and inspect installation of cartridge. Correct as required and repeat Step 7 above.

(8) Connect electrical leads as previously identified and ensure good connections.

(9) Place boots over terminal stud on each bonnet.

(10) Close circuit breakers.

8. Fire Extinguisher Double Check Valve Tee — Functional Test

A. Ensure FIRE EXT SHOT No. 1 and the FIRE EXT SHOT No. 2 circuit breakers are pulled.

NOTE: On Aircraft 1 - 200, 322 and 323 not having ASC 210. The circuit breakers are placard R NAC FIRE EXT and L NAC FIRE EXT.

B. Disconnect double check tee valve from the bonnet tubing and the crossover and distribution lines by loosening the B-nuts.

C. Remove check valve.

D. Test poppet for freedom of movement by gravity. It should move freely from one seat to the other and back without showing any signs of sticking or interference.

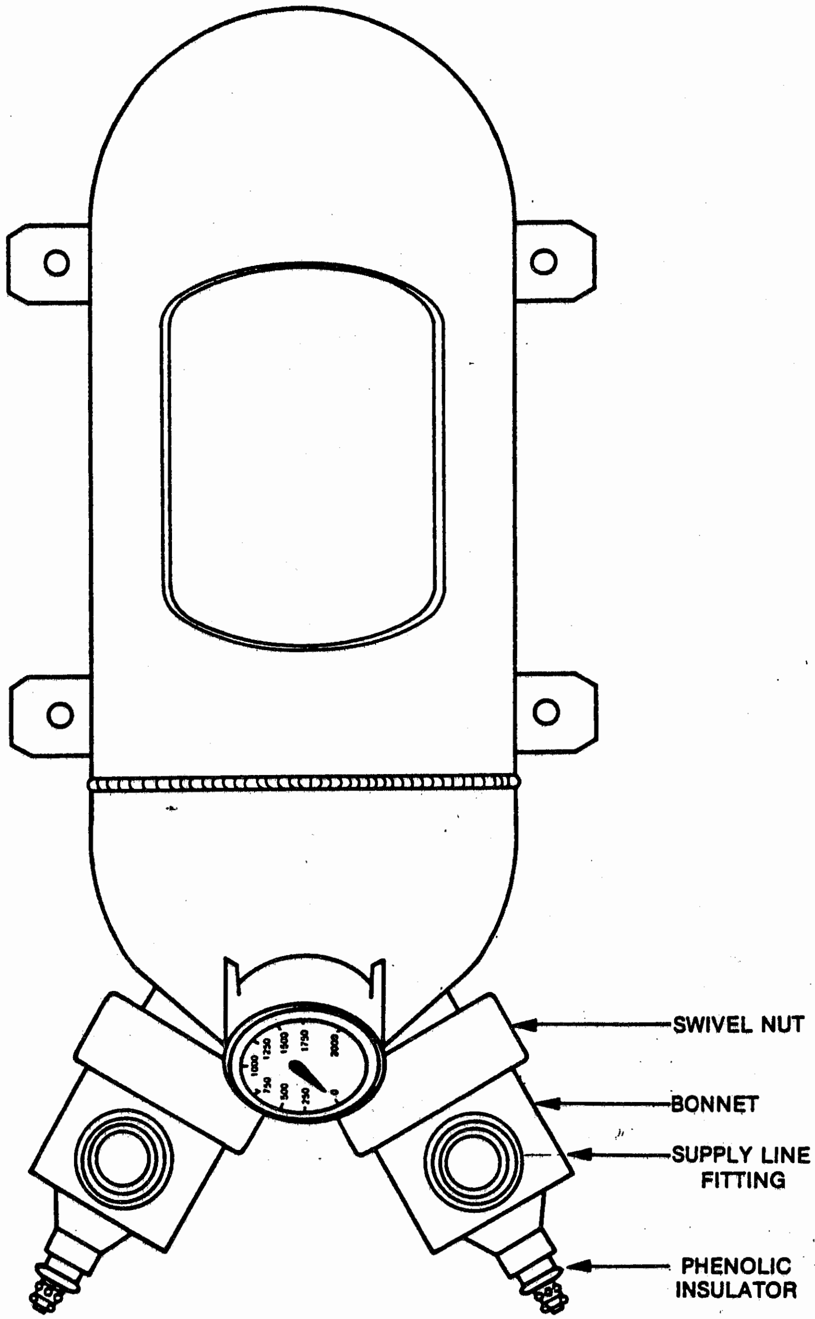
E. Perform functional test as follows:

(1) Cap off the outlet with a piece of short tubing containing a 1/8 inch orifice. Apply air or nitrogen at 40 psi quickly to first one inlet and then the other. Check that the poppet goes from seat to seat. The poppet must shuttle freely when valve is in any position.

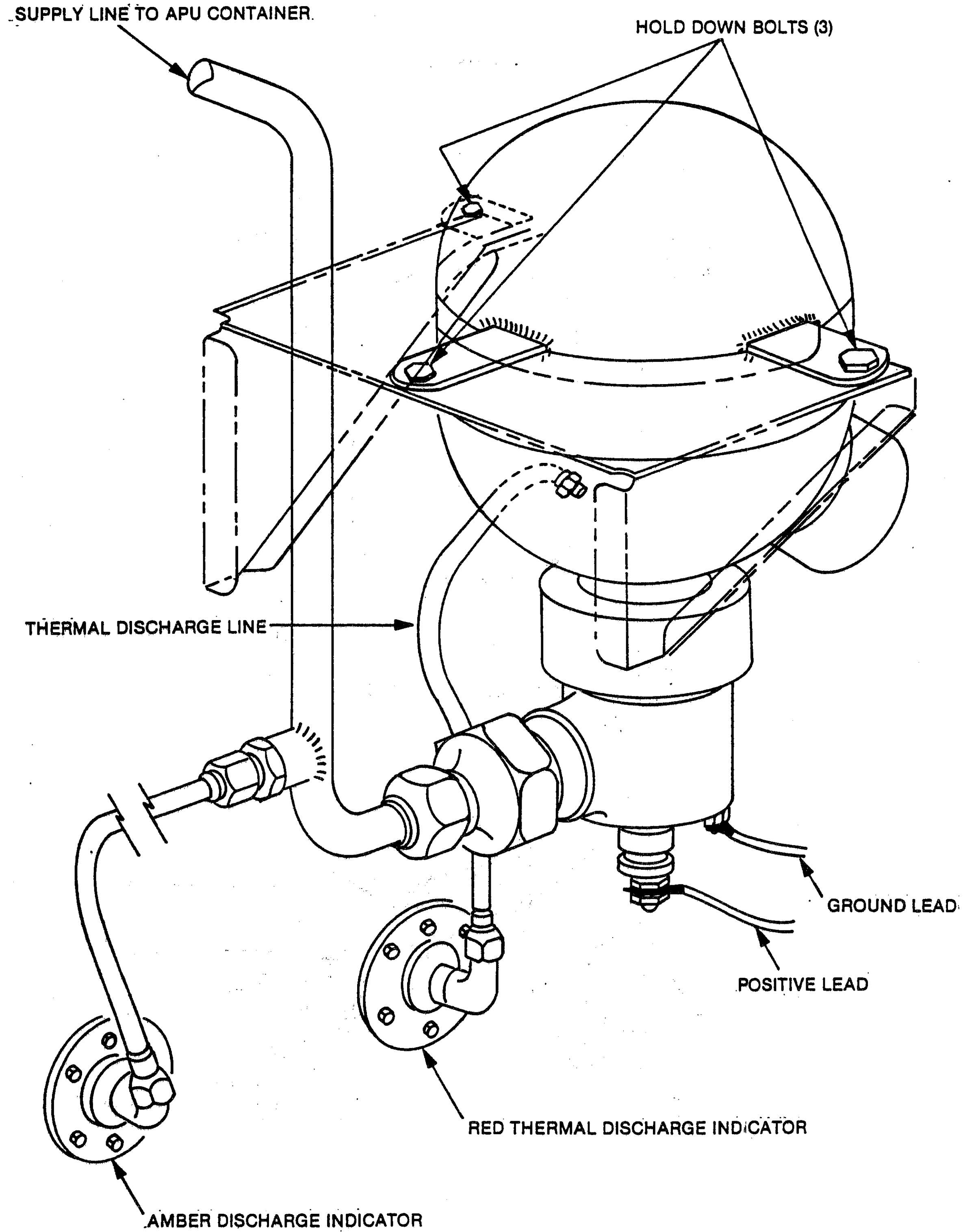
(2) Cap off the outlet and using a high flow regulator, apply air or nitrogen at 250 psi to one inlet. Allowable leakage from the other inlet shall not exceed one cfm.

(3) Reverse connections and repeat test to check second seat.

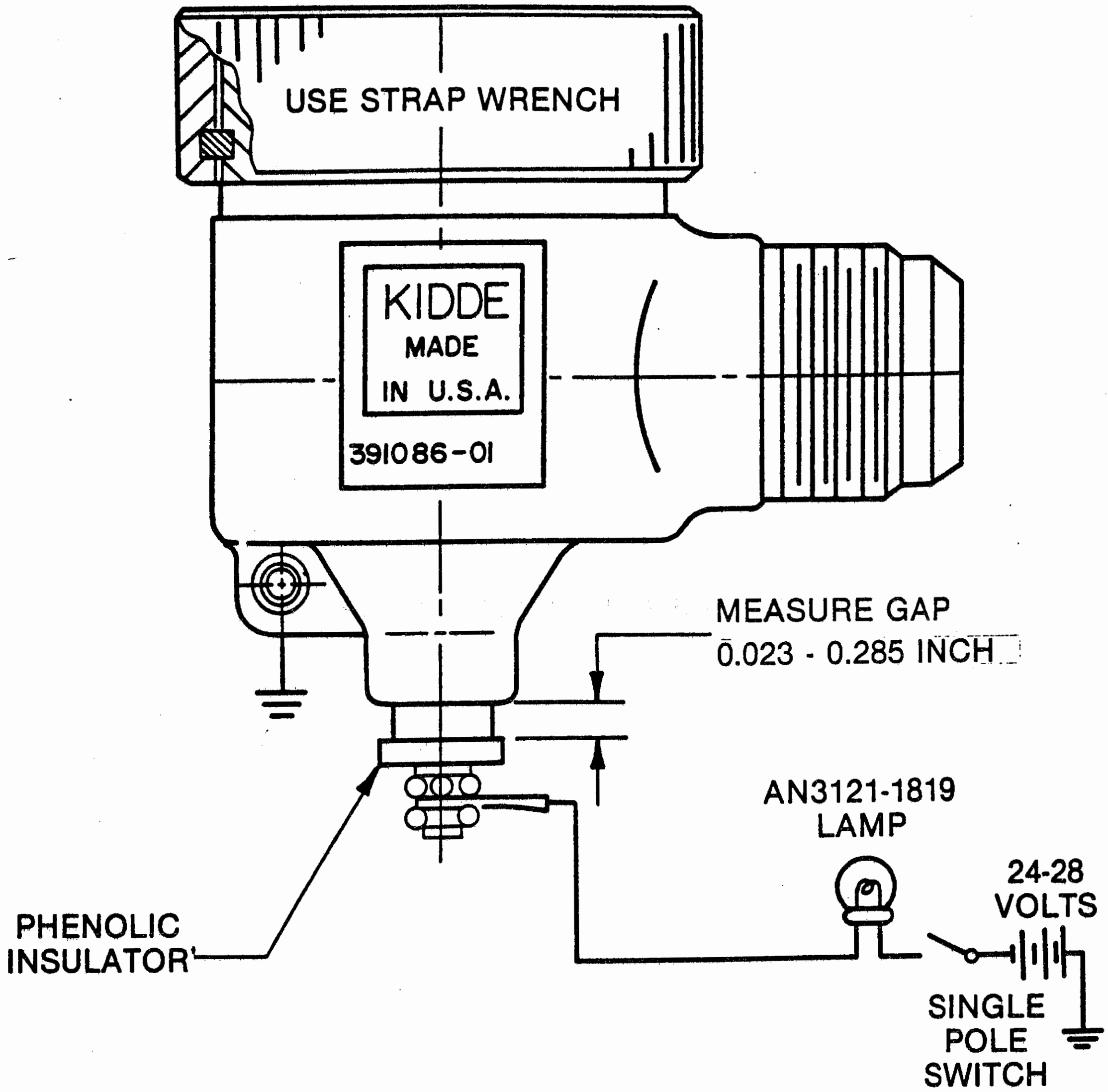
NOTE: To clean, flush the two-way check valve with cleaning solvent, Federal Specification PS-661 or equivalent. Dry with clean, dry compressed air.



Engine Fire Extinguisher Bottle Assembly
Figure 201.



APU Fire Extinguisher Installation
Figure 202.



Engine and APU Fire Extinguisher Bonnet Installation
Figure 203.

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FIRE EXTISNGUISHER — INSPECTION / CHECK

1. **General**

A. The Nacelle and APU fire extinguishers must be removed and weighed in accordance with inspection times listed in the GULFSTREAM I INSPECTION SCHEDULE and utilizing data listed on the container nameplate.

CAUTION: IF A FIRE EXTINGUISHER IS DISCHARGED INTO A NACELLE OR AN APU ENCLOSURE WITH THE ENGINE OR APU IN A HOT STATE, MAKE CAREFUL INSPECTION OF THE ENGINE OR APU BEFORE PLACING IT BACK IN SERVICE AS CF_3Br IS EXTREMELY COLD AND MAY CAUSE FRACTURES ON HOT METALLIC SURFACES. IF THE SYSTEM IS DISCHARGED, THE CONTAINER(S) MUST BE REPLACED IMMEDIATELY. NO PURGING IS REQUIRED.

B. The dual squib cartridges must be replaced in accordance with times listed in the GULFSTREAM I INSPECTION SCHEDULE.

CAUTION: IF A CARTRIDGE IS REMOVED FROM ITS BONNET, DO NOT REINSTALL THAT CARTRIDGE INTO ANOTHER BONNET. HOWEVER, IT MAY BE REINSTALLED INTO THE SAME BONNET FROM WHICH IT WAS REMOVED.

PORTABLE FIRE EXTINGUISHERS — DESCRIPTION / OPERATION

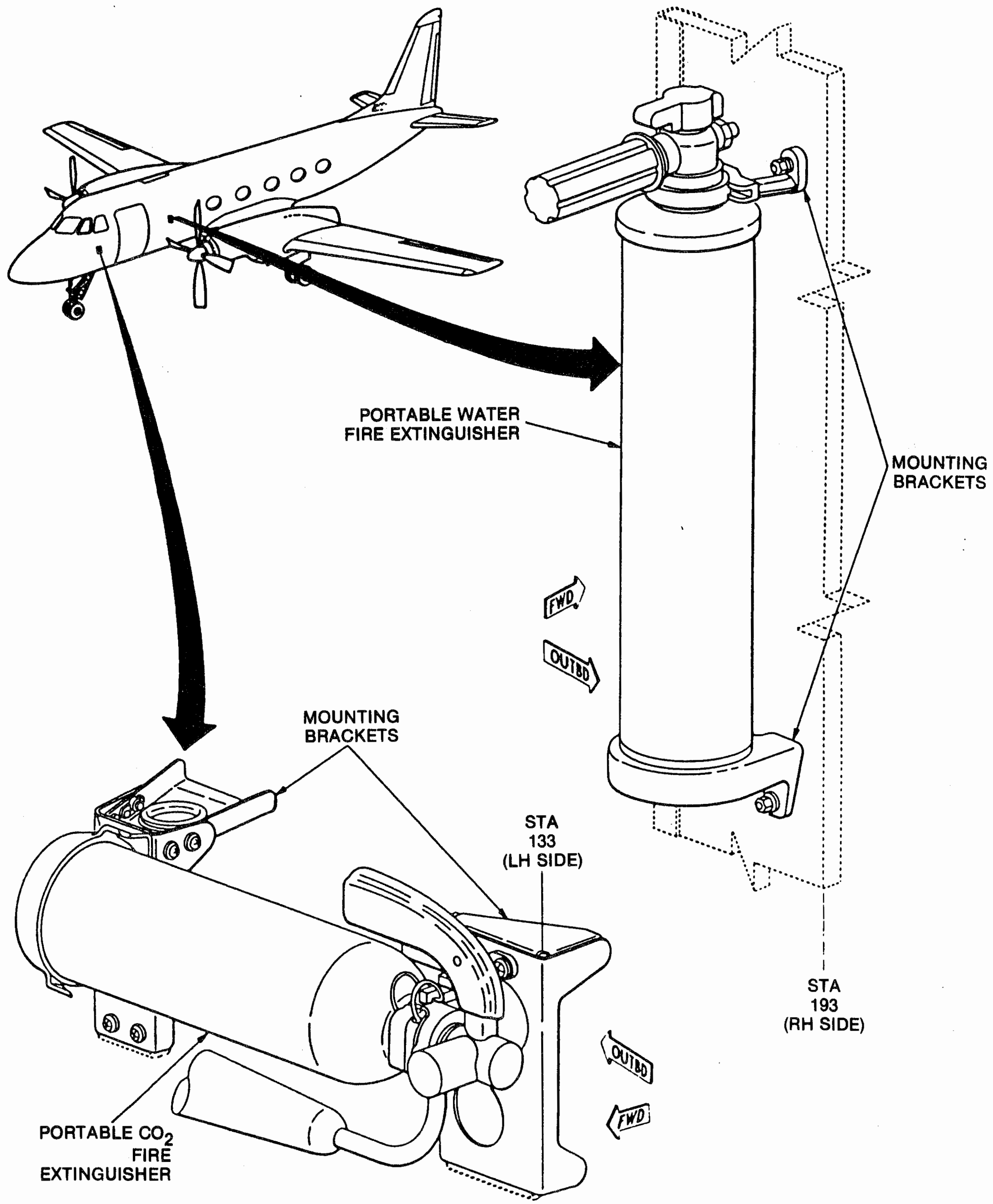
1. **Portable CO_2 Extinguisher**

As delivered, the Gulfstream I is equipped with a portable CO_2 fire extinguisher which is mounted on appropriate quick release brackets on the forward left side of Fuselage Station 133, in the cockpit, just behind the pilot's seat. This is a trigger actuated, swinging horn type extinguisher. The fully charged bottle weighs 8.2 pounds, and when actuated will completely discharge from 8 to 12 seconds. It is mounted in a horizontal position. The ambient temperature range for this extinguisher is from -40° C(-40°F) to +54.4°C (+130°F). The bottle contains a safety disc to permit release of the content in the event of excessive pressures. Content of the extinguisher is discharged through the horn upon the rupture of the safety disc.

2. **Portable Water Extinguisher**

As delivered, the aircraft is equipped with a portable water fire extinguisher, which is mounted on appropriate quick release brackets on the aft side of fuselage station 193 bulkhead, right side. Twisting of the carrying handle punctures a carbon dioxide cartridge, whose pressure expells the water when the thumb-operated valve lever is depressed. The extinguisher is mounted in a vertical position. Fully charged with 1-3/8 quarts of anti-freeze water solution, and ready for use, the extinguisher weights about 7 pounds. It is especially designed to combat Class "A" aircraft cabin fires. The time of discharge is between 30 to 45 seconds at 70°F, with a minimum range of 20 feet.

CAUTION: DO NOT TURN THE HANDLE UNLESS THE EXTINGUISHER IS REQUIRED TO COMBAT A FIRE, AS TURNING THE HANDLE PRESSURIZES THE CONTAINER WITH THE CO_2. IF THE HANDLE IS ACCIDENTALLY TURNED, DISCHARGE THE CONTAINER AND PERFORM SERVICING AND REFILLING PROCEDURES AS LISTED IN SECTION 26-2-1 — MAINTENANCE PRACTICES. DO NOT LEAVE A PRESSURIZED CONTAINER (CO_2 CARTRIDGE DISCHARGED, BUT THE WATER NOT DISCHARGED) INSTALLED IN THE AIRCRAFT.



Portable Fire Extinguishers — Installation
Figure 1

PORTABLE FIRE EXTINGUISHER — MAINTENANCE PRACTICES

1. **Portable CO_2 Extinguisher**

   A. If the portable CO_2 extinguisher is discharged, either by trigger action or safety seal rupture, it must be removed and replaced by a serviceable unit. Recharging of this extinguisher cannot be done in the field unless specifically equipped to do so and by agencies authorized to perform recharging and testing.

   B. Normal maintenance involves checking seal wire, security of mounting and validity of inspection date.

   C. At intervals indicated in the Gulfstream I Inspection Schedule, the extinguisher must be weighed and validated in accordance with the data inscribed on the extinguisher nameplate.

2. **Portable Water Fire Extinguisher** (See Figure 201)

   A. Disassembly of Cylinder Assembly from the Valve Assembly

      (1) Remove from aircraft.

      (2) Remove the seal, and the seal wire (if intact).

      (3) Completely discharge extinguisher in a suitable area by twisting the holder in the direction indicated by the arrow. This causes the CO_2 cartridge to be punctured, releasing the charge and pressurizing the cylinder. Depress the lever, allowing the pressure to force the water through the nozzle until all water and pressure are dissipated.

      CAUTION: NEVER DISASSEMBLE THE EXTINGUISHER UNTIL IT IS COMPLETELY DISCHARGED ONCE THE CARTRIDGE IS PUNCTURED.

      (4) Remove the three set screws and the lock ring. Unscrew the holder from the valve assembly.

      (5) Remove the spring from inside the holder, and the used cartridge from the valve. Discard used cartridge.

      (6) Unscrew the cylinder assembly from the valve assembly. The instruction decal and the nameplate decal need not be removed unless either is damaged.

      (7) Remove the syphon tube and strainer assembly from the valve assembly.

   B. Disassembly of Valve Assembly

      (1) Remove the upper syphon tube preformed packing from the valve.

      (2) Hold the lever stationary, and using a screwdriver, unscrew the check assembly.

      (3) Drive out the forward groove pin and remove the lever and the stem.

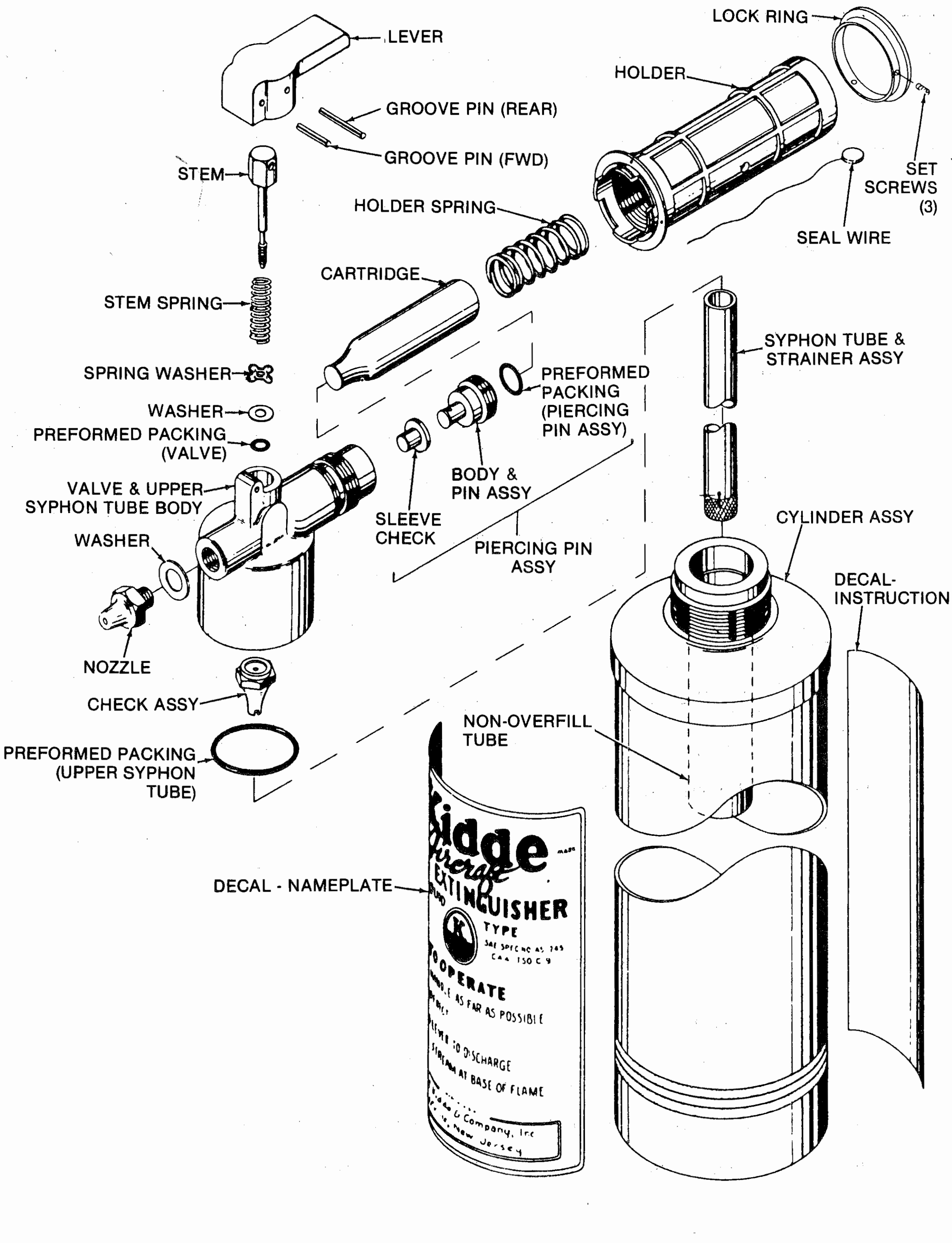
      (4) Drive out the rear groove pin and separate the lever and the stem.

      (5) Carefully invert the valve. The stem spring, spring washer, washer and the valve preformed packing will drop out.

      NOTE: A limited number of valve assemblies have a Tru-Arc ring in lieu of the spring washer. The spring washer and the Tru-Arc ring are interchangeable.

      (6) Carefully unscrew the nozzle and remove the washer from the nozzle.

      (7) Unscrew the piercing pin assembly from the valve body.



Portable Water Fire Extinguisher - Exploded View
Figure 201

C. Disassembly of the Piercing Pin Assembly

(1) Remove the piercing assembly preformed packing from the internal groove in the body and pin assembly.

(2) Slip the sleeve check from the body and pin assembly.

D. Cleaning Components of the Water Extinguisher

(1) Wash all metallic parts in dry cleaning solvent, Federal Specification P-S-661 or equivalent. Dry with clean, dry compressed air.

(2) Use clean, dry compressed air to dislodge any particles of foreign matter from the strainer portion of the syphon tube and strainer assembly.

E. Inspection of Water Extinguisher Components

(1) Examine seat of the check assembly, and the corresponding seating surface within the valve body and upper syphon tube for scratches or foreign matter which might affect proper operation of the extinguisher.

(2) Check all springs for deformation.

(3) Inspect the pin of the body and pin assembly for damage and tightness.

(4) Check the cylinder assembly for a non-overfill tube. Reject the assembly if the tube is not present.

(5) Inspect that the orifice of the nozzle is clear of obstructions.

(6) Examine interior of the cylinder assembly for foreign material, rust, corrosion, or excessively worn cylinder walls.

NOTE: Repair of the detailed components of the extinguishesr is not practical and is not recommended.

(7) Replace the following:

(a) All damaged parts.

(b) The three preformed packings.

(c) The sleeve check.

F. Assembling the Piercing Pin Assembly

(1) Lubricate new performed packing for piercing pin assembly with grease MIL-G-7711 and install packing in the internal groove in the body and pin assembly.

(2) Lubricate new preformed packing for the valve with grease MIL-G-7711 and seat the preformed packing in the valve body. Drop the washer on to the top of the preformed packing.

(3) Using an appropriate size rod, press the spring washer over the washer and preformed packing. Make certain that the feet of the spring washer face out of the valve body, and that the washer is held securely in place.

(4) Assemble the lever to the stem, using the groove pin (rear).

(5) Slip the stem spring over the stem of the lever and stem assembly and insert into the valve body.

(6) Engage the check assembly with the threads of the stem. Assemble the lever to the valve body using the groove pin (fwd.).

(7) Using a screwdriver, tighten the check assembly until the bottom of the lever is parallel with the horizontal centerline of the valve body.

(8) Lubricate new preformed packing for the upper syphon tube with grease MIL-G-7711, and install packing in the valve body and upper syphon tube.

(9) Place washer over the nozzle and screw into the valve body. Apply a torque of 180 inch pounds to the nozzle.

G. Assembly of Cylinder Assembly to the Valve Assembly

(1) Carefully press the syphone tube and strainer assembly into the valve assembly.

(2) If the instruction decal or the nameplate decal have been removed from the cylinder, apply new decals to the cylinder.

CAUTION: APPLICATION OF GREASE MIL-G-4343 IS ESSENTIAL TO PREVENT SEIZING OF THE COMPONENT THREADS.

(3) Lubricate the neck of a new cartridge with grease MIL-G-7711. Insert the holder spring into the holder, and the new cartridge into the valve body. Apply grease MIL-G-4343 to external threads of the valve body, and to the internal threads of the holder. Place the holder over the cartridge and turn the holder on to the valve body until red lines on the holder and the valve body match.

(4) Test the cylinder and body assembly as follows:

(a) Fill the cylinder assembly with 1-3/8 quarts of clean, distilled water.

(b) Assemble the valve assembly to the cylinder assembly hand tight.

(c) Operate the extinguisher.

(d) The discharge should continue for 30 to 45 seconds at 70°F, and should cover a minimum range of 20 feet.

(e) No leakage is permissible at the valve and cylinder connection.

(f) If the discharge stream sprays excessively, install a new stem and retest.

(5) Charge and completely assemble the extinguisher as follows:

(a) Depress the lever and flush the valve and syphon tube with water.

CAUTION: DO NOT ALLOW ANY UNDISSOLVED AGENT OR OTHER FOREIGN MATTER TO BE POURED INTO THE EXTINGUISHER.

DO NOT USE SOLUTION (800341) IN EXTINGUISHER P/N 892480 DUE TO ITS HIGHLY CORROSIVE PROPERTIES. USE SOLUTION (213923).

(b) Thoroughly dissolve the contents of anti-freeze package (Kidde 800341) in 1 quart of clean water. Add enough additional water to bring the solution to 1-3/8 quarts. Stir the solution thoroughly. Allow the solution to attain room temperature and then pour the solution through a screening cloth into the extinguisher.

(c) Assemble the valve assembly to the cylinder assembly hand tight.

(d) Install a new cartridge as outlined in step 2.G. (3) of this seciton.

(e) Slip the lockring over the holder. Position the lockring with its outside surface flush with the outside surface of the lever. Lock the lockring in place using the three set screws.

(f) Attach a new seal wire and a new seal.

(6) Reinstall extinguish in aircraft, checking security and condition of mounting brackets.

NOTE: Three preformed packings and the sleeve check which must be replaced in accordance with manufacturer's instructions reflected in this section are as follows:

NAME	PART NUMBER (KIDDE)
Preformed Packing (Piercing Pin Assembly)	202331
Preformed Packing (Valve)	1820-1
Preformed Packing (Upper Syphon Tube)	1820-19
Sleeve Check	202381

Other expendable items required for servicing the extinguisher are:

NAME	PART NUMBER (KIDDE)
Cartridge, CO_2	201386
Seal, Wire	204960
Agent, Anti-Freeze	800341

Information concerning all of the above, and any component in the portable water fire extinguisher can be obtained from the manufacturer:

Walter Kidde & Company, Inc.
675 Main Street
Belleville, New Jersey 07109

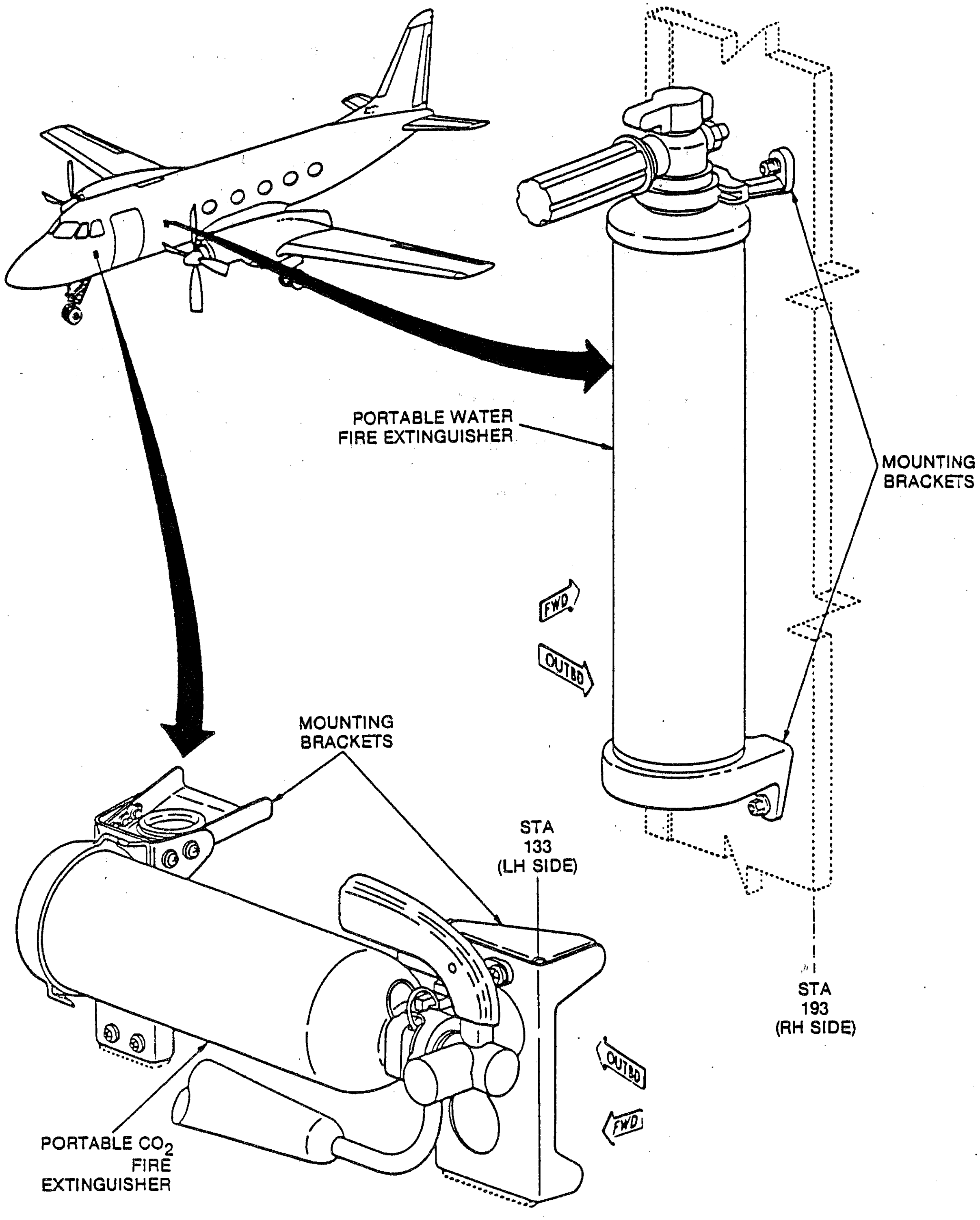
PORTABLE FIRE EXTINGUISHER — DESCRIPTION / OPERATION

1. **Portable CO2 Fire Extinguisher**

   As delivered, the aircraft (See Figure 1) is equipped with a portable CO_2 fire extinguisher which is mounted on appropriate quick release brackets on the forward left side of Fuselage Station 133, in the cockpit, just behind the pilots seat. This is trigger actuated, swinging horn type extinguisher. The fully charged bottle weighs 8.2 pounds, and when actuated will completely discharge in 8 to 12 seconds. It is mounted in a horizontal position. The ambient temperature range for this extinguisher is from − 40°C (− 40°F) to + 54.4°C (+ 130°F). The bottle contains a safety disc to permit release of the content in the event of excessive pressures. Content of the extinguisher is discharged through the horn upon the rupture of the safety disc.

2. **Portable H2O Fire Extinguisher**

   As delivered, the aircraft (See Figure 1) is equipped with a portable water (H_2O) fire extinguisher, which is mounted on appropriate quick release brackets on the aft side of Fuselage Station 193 bulkhead, right side. Twisting of the carrying handle punctures a carbon dioxide cartridge, whose pressure expels the water when the thumb-operated valve lever is depressed. The extinguisher is mounted in a vertical position. Fully charged with 1 3/8 quarts of anti-freeze water solution, and ready for use, the extinguisher weighs about seven pounds. It is especially designed to combat Class "A" aircraft cabin fires. The time of discharge is between 30 to 45 seconds at 70°F, with a minimum range of 20 feet.



Portable Fire Extinguishers
Figure 1.

PORTABLE FIRE EXTINGUISHER — MAINTENANCE PRACTICES

1. **Portable CO2 Fire Extinguisher**

If the portable CO_2 fire extinguisher is discharged, either by trigger action or safety seal rupture, it must be removed and replaced by a serviceable unit. Recharging of this fire extinguisher cannot be done in the field unless specifically equipped to do so and by agencies authorized to perform recharging and testing.

Normal maintenance involves checking seal wire, security of mounting and validity of inspection date.

At intervals indicated in Chapter 5, Time Limits / Maintenance Checks, the fire extinguisher must be weighed and validated in accordance with the data inscribed on the extinguisher nameplate.

2. **Portable H2O Fire Extinguisher — Functional Test / Service**

NOTE: Information concerning any component in the portable water fire extinguisher can be obtained from the manufacturer: Walter Kidde Aerospace, Inc..

A. Disassembly of Cylinder Assembly from the Valve Assembly (See Figure 201).

(1) Remove fire extinguisher from aircraft.

(2) Remove seal and seal wire (if intact).

(3) Completely discharge extinguisher in a suitable area by twisting the holder in the direction indicated by arrow. This causes the CO2 cartridge to be punctured, releasing the charge and pressurizing the cylinder. Depress the lever, allowing pressure to force the water through the nozzle until all water and pressure are dissipated.

CAUTION: NEVER DISASSEMBLE THE EXTINGUISHER UNTIL IT IS COMPLETELY DISCHARGED ONCE THE CARTRIDGE IS PUNCTURED.

(4) Remove set screws and lock ring. Unscrew the holder from valve assembly.

(5) Remove the spring from inside the holder and the used cartridge from the valve. Discard used cartridge.

(6) Unscrew the cylinder assembly from the valve assembly. The instruction decal and the nameplate decal need not be removed unless either is damaged.

(7) Remove syphon tube and strainer assembly from the valve assembly.

B. Disassembly of Valve Assembly

(1) Remove the upper syphon tube preformed packing from the valve.

(2) Hold the lever stationary and using a screwdriver, unscrew the check assembly.

(3) Drive out the forward groove pin and remove the lever and stem.

(4) Drive out the rear groove pin and separate the lever and stem.

(5) Carefully invert the valve. The stem spring, spring washer, washer and valve preformed packing will drop out.

NOTE: A limited number of valve assemblies have a Tru-Arc ring in lieu of the spring washer. The spring washer and the Tru-Arc ring are interchangeable.

(6) Carefully unscrew the nozzle and remove the washer from the nozzle.

(7) Unscrew the piercing pin assembly from the valve body.

C. Disassembly of the Piercing Pin Assembly

(1) Remove the piercing pin assembly preformed packing from the internal groove in the body and pin assembly.

(2) Slip the sleeve check from the body and pin assembly.

D. Cleaning Components of the Water Extinguisher

(1) Wash all metallic parts in dry cleaning solvent, Federal Specification P-S-661 or equivalent. Dry with clean, dry compressed air.

(2) Use clean, dry compressed air to dislodge any particles of foreign matter from the strainer portion of the syphon tube and strainer assembly.

E. Inspection of Water Extinguisher Components

(1) Examine seat of check assembly and the corresponding seating surface within the valve body and upper syphon tube for scratches or foreign matter which might affect proper operation of extinguisher.

(2) Check all springs for deformation.

(3) Inspect pin of body and pin assembly for damage and tightness.

(4) Check cylinder assembly for a non-overfill tube. **Reject the assembly if the tube is not present.**

(5) Ensure that the orifice of the nozzle is clear of obstructions.

(6) Examine interior of cylinder assembly for foreign material, rust, corrosion, or excessively worn cylinder walls.

NOTE: Repair of detailed components of the extinguisher is not practical and is not recommended

(7) Replace all damaged parts, the three preformed packings and the sleeve check.

F. Assembling the Piercing Pin Assembly

(1) Lubricate new preformed packing for piercing pin assembly with grease MIL-L-7711 and install packing in the internal groove in the body and pin assembly.

(2) Lubricate new preformed packing for valve with grease MIL-L-7711 and seat preformed packing in the valve body. Drop the washer on the top of the preformed packing.

(3) Using an appropriate size rod, press spring washer over the washer and preformed packing. Ensure that the feet of spring washer face out of the valve body and that washer is held securely in place.

(4) Assemble lever to stem, using the groove pin (rear).

(5) Slip stem spring over the stem of the lever and stem assembly and insert into valve body.

(6) Engage the check assembly with threads of the stem. Assemble lever to valve body using the groove pin (forward).

(7) Using a screwdriver, tighten check assembly until bottom of lever is parallel with horizontal centerline of valve body.

(8) Lubricate new preformed packing for upper syphon tube with grease MIL-L-7711 and install packing in valve body and upper syphon tube.

(9) Place washer over nozzle and screw into valve body. Apply a torque of 180 inch-pounds to the nozzle.

G. Assembly of Cylinder Assembly to the Valve Assembly

(1) Carefully press syphon tube and strainer assembly into valve.

(2) If the instruction decal or the nameplate decal have been removed from cylinder, apply new decals to the cylinder.

(3) Lubricate neck of a new cartridge with grease MIL-L-7711. Insert holder spring into holder and new cartridge into valve body. Apply grease MIL-L-4343 to external threads of valve body and to internal threads of the holder. Place the holder over cartridge and turn holder on to valve body until the **red line** on the **holder** and the **valve body match**.

CAUTION: APPLICATION OF GREASE MIL-L-4343 IS ESSENTIAL TO PREVENT SEIZING OF THE COMPONENT THREADS.

(4) Test cylinder and body assembly as follows:

(a) Fill the cylinder assembly with 1 3/8 quarts of clean, distilled water.

(b) Assemble valve assembly to cylinder assembly hand tight.

(c) Operate the extinguisher

(d) The discharge should continue for 30 to 45 seconds at 70°F and should cover a minimum range of 20 feet.

(e) No leakage is permissible at the valve and cylinder connection.

(f) If the discharge stream sprays excessively, install a new stem and retest.

(5) Charge and completely assemble the extinguisher as follows:

(a) Depress the lever and flush the valve and syphon tube with water.

(b) Thoroughly dissolve contents of anti-freeze package (Kidde 800341) in one quart of clean water. Add enough additional water to bring the solution to 1 3/8 quarts. Stir the solution thoroughly. Allow solution to attain room temperature and then pour solution through a screening cloth into extinguisher.

CAUTION: DO NOT ALLOW ANY UNDISSOLVED AGENT OR OTHER FOREIGN MATTER TO BE POURED INTO THE EXTINGUISHER.

(c) Assemble valve assembly to cylinder assembly hand tight.

(d) Install a new cartridge as outlined in Step G.(3) above.

(e) Slip lock ring over the holder. Position lock ring with its outside surface flush with outside surface of the lever. Lock the lock ring in place using three set screws.

(f) Attach a new seal wire and a new seal.

(6) Install extinguisher in aircraft, check security and condition of mounting brackets.

3. Portable (CO2 / Halon) Fire Extinguisher — Weight Check

NOTE: Once used, even if only a puff discharge, extinguisher must be recharged promptly. If there is evidence of physical damage, shell must be subjected to hydrostatic test to determine reliability.

A. Remove fire bottle and perform weight check.

B. Check wire seal for security then install fire bottle.

4. Portable (CO2 / H2O / Halon) Fire Extinguisher — Hydrostatic Test

NOTE: Hydrostatic test must be performed by a certified overhaul agency.

A. Remove portable fire bottle and perform hydrostatic test.

B. Check wire seal for security then install portable fire bottle.

5. Dry Chemical Extinguisher — Inspection

NOTE: Once used, even if only a puff discharge, extinguisher must be recharged promptly. If there is evidence of physical damage, shell must be subjected to hydrostatic test to determine reliability.

A. Check wire seal for security.

B. Check that pressure gage reading is in operable range.

6. Dry Chemical Extinguisher — Hydrostatic Test

NOTE: Hydrostatic test must be performed by a certified overhaul agency.

A. Remove portable fire bottle and perform hydrostatic test.

B. Check wire seal for security then install portable fire bottle.

7. **Dry Chemical Extinguisher — Weight Check**

   A. Remove portable fire bottle and perform weight check.

   B. Check wire seal for security.

   C. Check that pressure gage reading is in operable range.

   D. Install extinguisher.

8. **Portable Halon Fire Extinguisher — Inspection**

   A. Ensure that pressure indicator reads in "green range".

   B. Ensure that indicator disk is not fractured.

   C. Ensure that locking pin is intact.

   D. Check for physical damage.

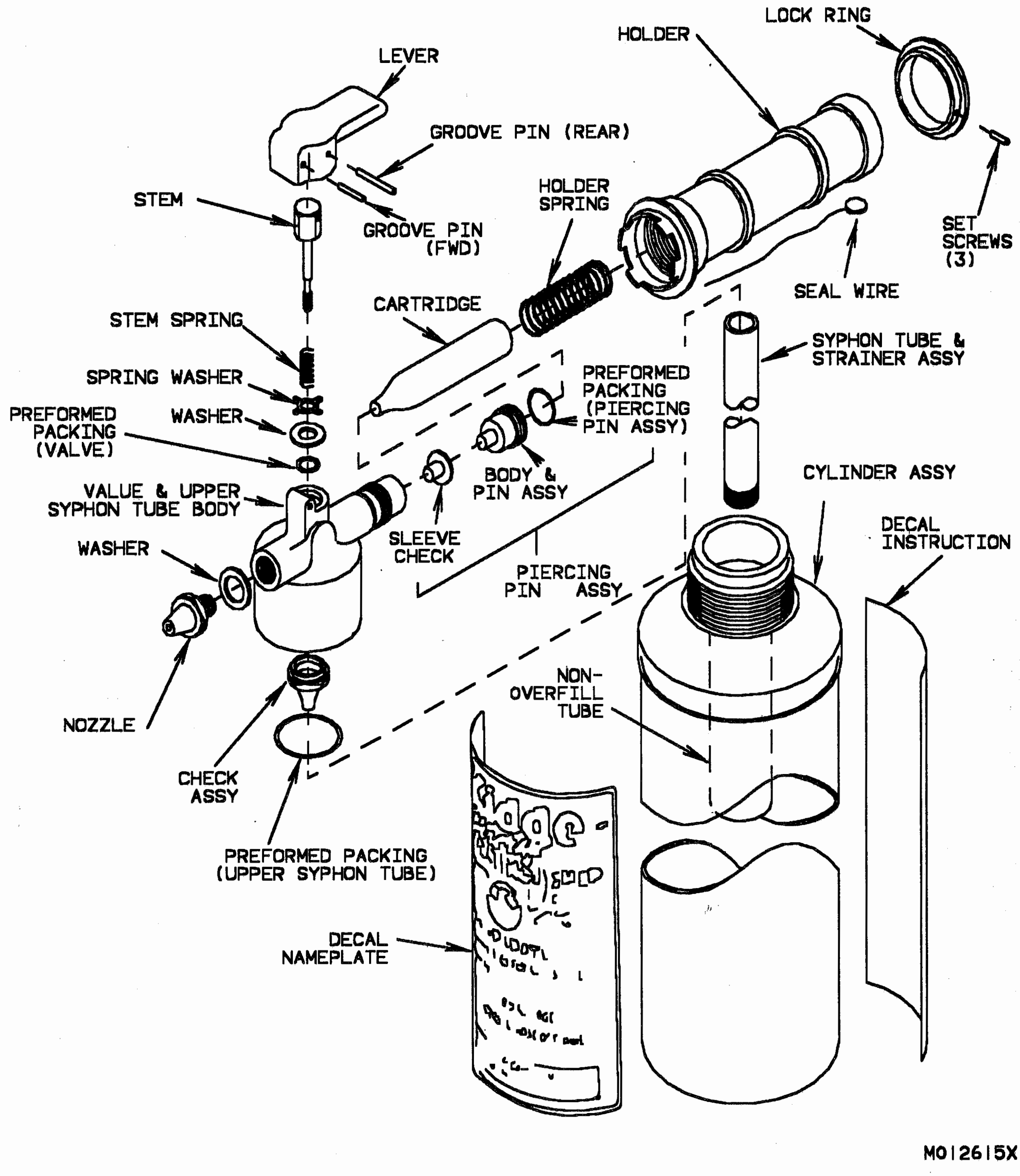
NAME	PART NUMBER (KIDDE)
Preformed Packing (Piercing Pin Assembly)	202331
Preformed Packing (Valve)	1820-1
Preformed Packing (Upper Syphon Tube)	1820-19
Sleeve Check	202381

Other expendable items required for servicing extinguisher are:

NAME	PART NUMBER (KIDDE)
Cartridge, CO_2	201386
Seal, Wire	204960
Agent, Anti-Freeze	800341

Information concerning all of the above, and any component in the portable water fire extinguisher can be obtained from the manufacturer:

Walter Kidde & Company, Inc.
675 Main Street
Belleville, New Jersey 07109



Portable Water Fire Extinguisher — Exploded View
Figure 201.

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FLIGHT CONTROLS

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CHAPTER 27

FLIGHT CONTROLS

FLIGHT CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

The aircraft is equipped with a manual control system which incorporates conventional ailerons, elevators, and rudder. All control surfaces are aerodynamically balanced. Aircraft trim about all three axes is provided by mechanically operated tabs.

The primary control system consists of bellcranks, pushrods, and straight cable runs operating in a low friction system. The lockclad cable used in the aircraft is a high strength cable with a coefficient of expansion close to that of the structure. As a result the problem of cable sag due to temperature changes is minimized in the system.

Single slotted, Fowler flaps provide lift augmentation for takeoff and landing. The flaps extend along the wing trailing edge from the fuselage to the ailerons. Each wing flap is of one-piece construction operating on four tracks. Two mechanical drives per side, powered by a single hydraulic motor and central gearbox in the wing center section, operate the flaps. The hydraulic motor and drives are common to both normal and emergency flaps. A mechanical stop is provided on the flap tracks, to prevent over travel. Four positions are provided for normal flap operation. These positions, selected by the pilot or copilot by the operation of the normal flap control are:

- UP - 0°
- TAKEOFF - 12.5°
- APPROACH - 20°
- FULL DOWN - 33°

NOTE: Aircraft having ASC 252 have the capability of selecting 6.5° takeoff and 12.5° approach if needed.

Emergency flaps are manually controlled by the copilot.

The elevators are interconnected at the fuselage and move through a range of 25° up and 14° down. Two trim tabs are provided for longitudinal trim. The control columns are connected by a torque tube and feed into a single elevator control system. Control forces are fed from the control column into the cables by a pushrod and bellcrank system. The cables carry these forces back to another pushrod and bellcrank system at the elevator, resulting in elevator movement.

The rudder moves through a range of 22° right and left of the neutral position. The rudder pedals are connected to a torque tube which in turn feeds into a single rudder control system. Control forces are fed into a combination of cables, pushrods, and bellcranks resulting in rudder movement.

Lateral control is provided by ailerons moving through a range of 16° up and 12° down. A trim tab is provided on the left aileron only. The ailerons are controlled by a manual system of pushrods, bellcranks, and cables originating at the control columns under the cockpit floor. Bellcranks, fitted to the control columns and interconnected by a pushrod pick up two cables, one on each side of the aircraft. These cables run back through the fuselage to the wing where they again pick up bellcranks which are connected across the fuselage by a cross-over pushrod. Cables from these bellcranks run outboard to the ailerons for lateral control movements.

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FLIGHT CONTROL SYSTEM — MAINTENANCE PRACTICES

1. Mass Balance

NOTE: The following procedures apply to all of the flight control surfaces. For specific information concerning tolerances for mass balance of a particular flight control surface, refer to applicable flight control section in this chapter.

All control surfaces may be balanced off the aircraft. The aileron and elevator may be balanced on the aircraft. The rudder and its tab must be balanced off the aircraft. To do this, the surface must be supported on two hinges, preferably the end ones, with the hinge line horizontal. See to figure 201 for a simple part made of commercial grade lumber that will support one hinge.

A. Causes of the Static Balance Becoming Out of Tolerance:

- Excessive layers of paint on the surface: Aircraft paint shops should never overcoat control surfaces without removing the previous coat. It is desirable to keep control surfaces a single color to avoid overcoating.
- Repairs, particularly at the trailing edge: Control surfaces are generally lightly built and are, therefore, easily damage. Weight added by a repair can be a problem. Typically, an ounce added aft of the hinge requires two ounces of weight added to balance it.
- Moisture and dirt accumulation inside the surface.

B. Measuring Control Surface Static Balance

(1) Remove control surface from the aircraft.

(2) Using the end hinges, support the control surface on two posts as shown in Figure 201.

NOTE: All tabs, access panels, static wicks with their attaching hardware must be in place, but the attaching hardware need not be screwed down tight. All hardware for attaching the leading edge curtain must be in place. The trim tab actuator (elevators and ailerons) must be in place, with its cables disconnected and unwound from the drum. The hinge support fittings on any hinges not being used to support the surface during balancing must have their weight supported during the balancing. It is acceptable to have a helper hold them up while the balancing is being done.

(3) Determine whether the surface is nose or tail heavy.

(4) If the surface is tail heavy, apply balance weights forward of the hinge line to balance the surface. For surfaces with a horn, such as the rudder and elevator, work at the outboard end since adding weight to the horn gives the largest moment arm and hence the lowest weight. If the surface is nose heavy, it may be balanced from either end, since ample moment arm exists.

(5) At or near the end of the surface at which you are going to work, set up a board or piece of stiff cardboard vertically at the trailing edge, but not touching the control surface. Mark on it the position of the trailing edge with chordline horizontal.

(6) Place weights forward or aft of the hinge as required.

NOTE: The out-of-balance moment is the product of the weights added time s its distance from the hinge line. Use either a fixed known weight and obtain a balance by varying the distance from the hinge line or have a fixed attachment for the weight and vary the amount of weight.

Placing weights on top of the surface is less satisfactory than hanging them below the surface since with the chord horizontal in the supposedly balanced condition, the c.g. of the surface and the weight may be above the hinge point causing an unstable condition. The ideal equilibrium position is one in which a small nose up displacement becomes more nose up, and a small nose down displacement becomes more nose down.

A simple method of hanging small weights to the leading or trailing edge is shown in Figure 202. In the case of the rudder and elevator, the closing rib flange of the horn

balance may be used and a wire hook could bear directly on it. Everything shown in Figure 202. should be included in the weight when calculating the out-of-balance moment.

(7) Measure the moment arm from the c.g. of the weight to the hinge line to within 0.25 inch. Measure the weight to within 0.10 pound.

NOTE: If the surface is being rebalanced because of weight added for repair to known damage, and it is found that extra balance is required, then the additional weight must be added at the same spanwise station as the repair. The requirements are that the weight must not interfere with (or reduce) the surface angular movement and that the attachment of the weight and the backup structure to which it is attached must be strong enough to meet the required `g' forces. It is recommended that Gulfstream Aerospace Engineering assistance be sought in this situation.

(8) After a surface has been balanced record the actual out-of-balance value in inch pounds in the aircrafts logbook, with the date and the signature of the responsible inspector.

C. Determining Cause of Unbalance When Surface Found Out of Balance Beyond Prescribed Limits

The reason for excessive unbalance must be found and removed. Causes of unbalance are given in paragraph 1A above, however, it is not acceptable to restore a control surface that is outside the balance limits given to the right limits by simply adding weights.

The spanwise location of the weights is also important. The additional balance weight must be added at the same spanwise station as the weight causing the unbalance.

(1) Inspect the surface for signs of repair action. A check on the corresponding surface of another Gulfstream I might be helpful.

(2) Check the paint thickness. If in doubt, strip the paint and rebalance.

(3) Remove the tab and weight it. The tab weights should be:

- Rudder: 19.4 pounds primed 21.0 pounds painted
- Elevator: 4.7 to 5.0 pounds
- Aileron Spring Tab: 7.3 pounds
- Aileron Trim Tab: 2.1 pounds

(4) X-ray the surface to detect internal repairs, accumulation of moisture and other debris, as well as corrosion and fatigue cracks.

(5) If the cause of the unbalance is still undetermined, it is suggested that the surface be removed and sent to Gulfstream Aerospace for rework. Loaner surfaces are available.

NOTE: The risk of continued flying with an unbalanced surface is that the speed margin for flutter safety over and above normal maximum operating limits (V_{MO}) that was designed into the aircraft be significantly reduced. If the aircraft has been flying with this surface in the condition found, future flying should be restricted to a speed 50 knots slower than the normal V_{MO} until the surface is changed or reworked.

If there is a need to repair damage to the surface where the damage is great enough to render the surface unairworthy without repair, then the surface must be repaired and rebalanced before further flight. Gulfstream Aerospace will be happy to assist as required in the design and installation of such a repair and the installation of required extra balance. Loaner surfaces are available and it is recommended that the damaged surface be shipped to Gulfstream Aerospace for rework.

2. **Cable Pressure Seal — Lubrication**

NOTE: Clean seals prior to lubrication and wipe seals and area clean of excess lubricant upon completion.

A. Lubricate cable pressure seal (See Chapter 12)

- Gust lock – FS 561
- Gust lock – BL 19
- Rudder – FS 561
- Elevator and rudder tab – FS
- Aileron tab – BL 19
- Left engine controls – SL 19
- Right engine controls – BL 19
- Power steering – FS 67

3. **Flight Control Cable — Damage Limits**

The allowable limits for Flight Control Cable damage are as follows:

A. Primary Control Cables (3 1/16 inch diameter 7 X 19).

(1) Three broken wires per strand are allowed in any running inch of cable, with a maximum of six broken wires per inch. A maximum of six broken wires is allowed in any two consecutive inches of cable run.

(2) A maximum of three broken wires per inch is allowed in any length of cable passing over pulleys and drums or through fairleads.

(3) Any wire worn through in excess of 50% of its diameter shall be considered a broken wire.

(4) No kinking, untwisting, or bird caging of the cable is allowed.

B. Secondary Control Cables (3/32 inches diameter 7 X 7).

(1) Two broken wires per strand are allowed in any running inch of cable, with a maximum of three broken wires per inch. A maximum of three broken wires is allowed in any two consecutive inches of cable run.

(2) A maximum of three broken wires per inch is allowed in any length of cable passing over pulleys and drums or through fairleads.

(3) Any wire worn through in excess of 50% of its diameter shall be considered a broken wire.

(4) No kinking, untwisting or bird caging is allowed.

C. If cable damage is in excess of the limits specified, it is recommended that the damaged cable be replaced rather than repaired. The cause for damage (misaligned or damaged pulleys, drums and fairleads or structural interference) should be found and corrected immediately.

After reworking structure to provide increased clearance, the reworked area should be painted. This is normal in any case for corrosion protection and the area should be inspected after further operation to see if the paint shows any sign of rubbing. If there are indications of rubbing, the clearance should be increased still further. Premature wear of control cables indicates a requirement for a check for misalignment, and, in the interest of control cable life and the total friction level in the control system, it is worthwhile making an effort to improve the situation.

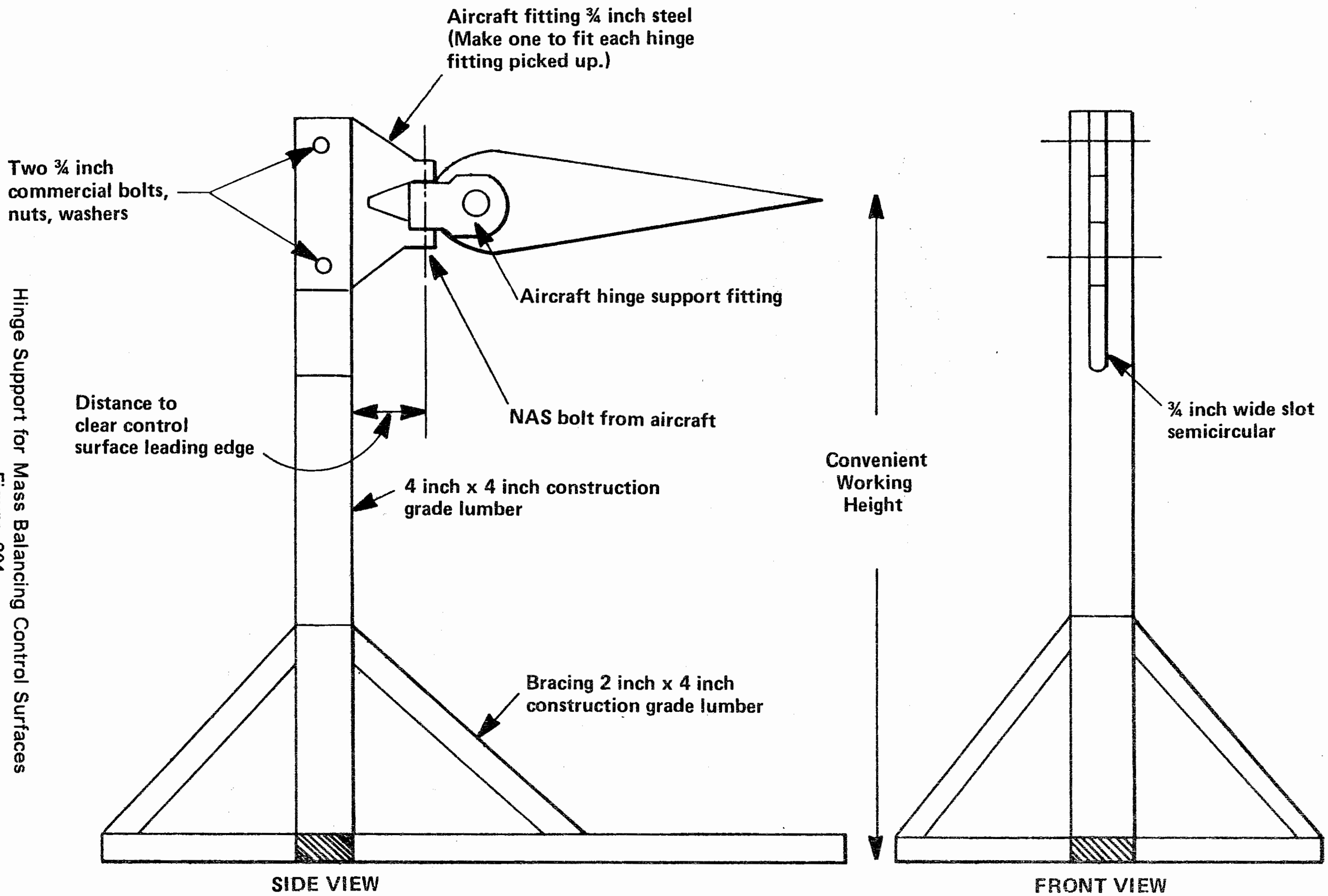
It should be borne in mind that deflections of the structure in flight can remove what appears to be an adequate clearance on the ground. If adequate clearance cannot be easily achieved inform the Gulfstream Aerospace Engineering Department.

4. Close Tolerance Hardware In Movable Joints — Removal / Installation

Upon removal of close tolerance bolts and/or hardware passing through similar or dissimilar metals (movable joints such as ailerons, elevators, hinges, control pivots, etc.), an inspection should be made for signs of surface corrosion. If metal substrate is exposed or any areas exhibit surface corrosion, it is recommended that the close tolerance bolts and/or hardware be replaced with identical replacement hardware. All close tolerance bolts and/or hardware must be coated with a compound conforming to Specification MIL-C-16173, Grade 1, or wet zinc chromate primer conforming to Specification MIL-P-6585 prior to installation.

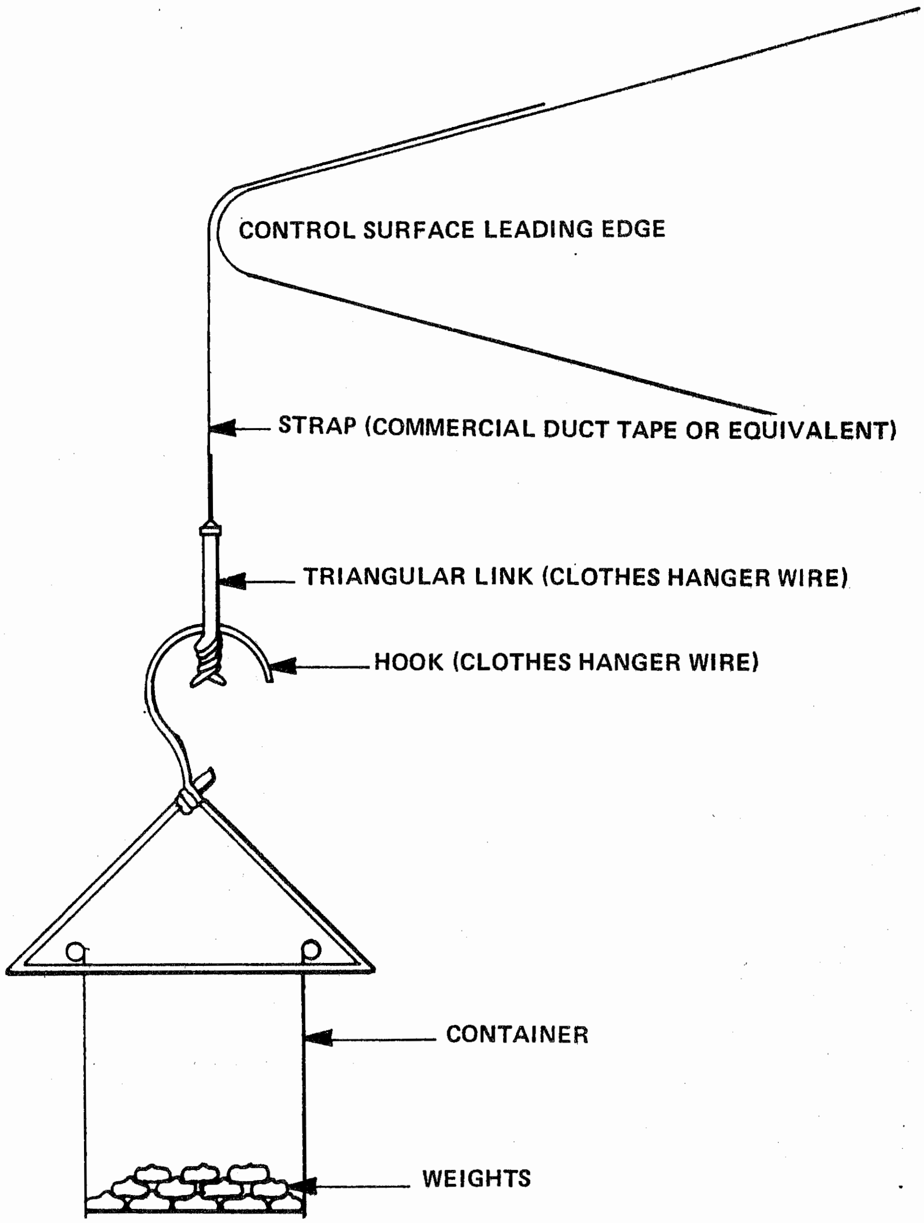
5. Turnbuckle Safetying

(See Figure 203)



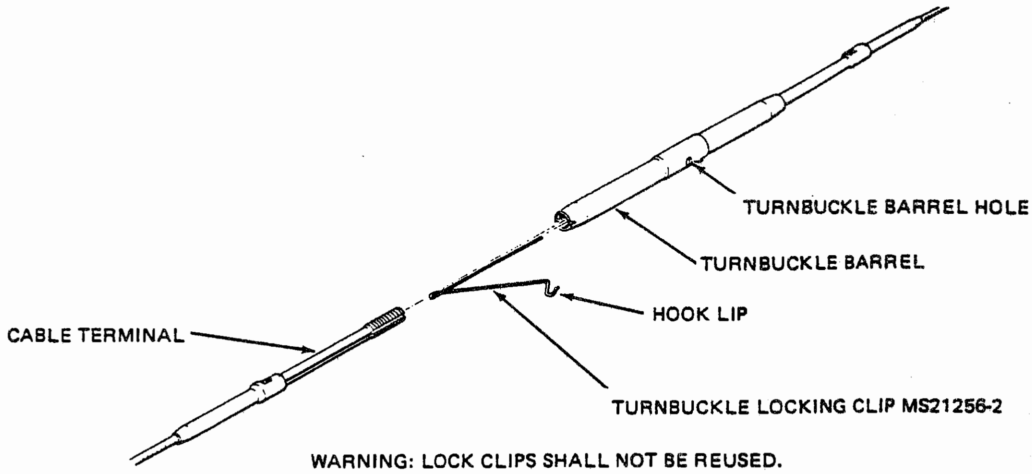
NOTE: Different adapter fittings must be made for each hinge point.

Hinge Support for Mass Balancing Control Surfaces
Figure 201.



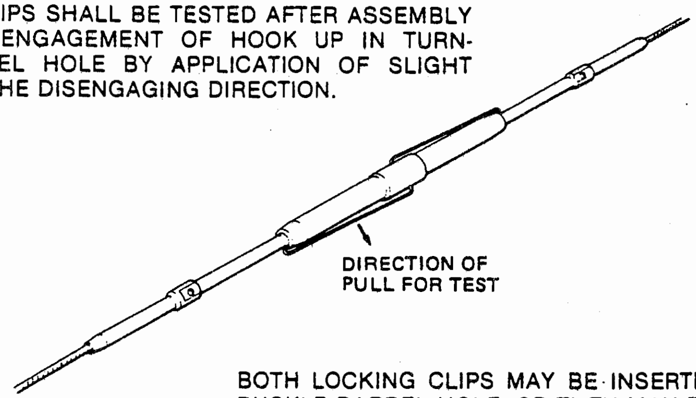
Suggested Method of Hanging Weights on Control Surface Leading Edge
Figure 202.

TO SAFETY WIRE TURNBUCKLE, ALIGN THE SLOT IN THE BARREL WITH THE SLOT IN THE CABLE TERMINAL. HOLD THE LOCK CLIP BETWEEN THE THUMB AND FOREFINGER AT THE END LOOP AND INSERT THE STRAIGHT END INTO THE APERTURE FORMED BY THE ALIGNED SLOTS. BRING HOOK END OF LOCK CLIP OVER THE HOLE IN THE CENTER OF THE TURNBUCKLE BARREL AND SEAT THE HOOK LOOP INTO THE HOLE. APPLICATION OF PRESSURE TO THE HOOK SHOULDER WILL ENGAGE THE HOOK LIP IN THE TURNBUCKLE BARREL. COMPLETE THE SAFETY LOCKING OF ONE END. THE ABOVE STEPS ARE THEN REPEATED ON THE OPPOSITE END OF THE TURNBUCKLE BARREL.

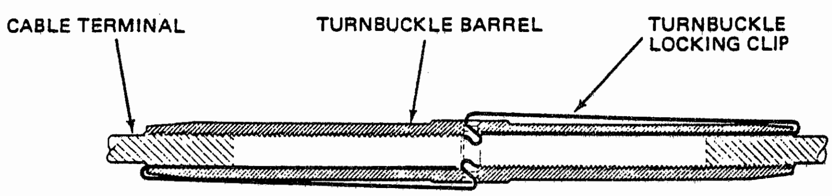


WARNING: LOCK CLIPS SHALL NOT BE REUSED.

BOTH LOCK CLIPS SHALL BE TESTED AFTER ASSEMBLY FOR PROPER ENGAGEMENT OF HOOK UP IN TURNBUCKLE BARREL HOLE BY APPLICATION OF SLIGHT PRESSURE IN THE DISENGAGING DIRECTION.



BOTH LOCKING CLIPS MAY BE INSERTED IN THE SAME TURNBUCKLE BARREL HOLE, OR THEY MAY BE INSERTED IN THE OPPOSITE HOLES.



Control System Turnbuckle Safetying
Figure 203.

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CONTROL COLUMNS — DESCRIPTION / OPERATION

1. Description and Operation

See 27-0-1 Figure 201)

The aircraft is equipped with conventional type dual control columns for both pilot and copilot. Fore and aft movement of the control column provides for longitudinal control, while rotation of the control wheel left or right provides for lateral control.

The pilots and copilots control column are mounted on a common transverse torque tube at fuselage station 100.500. The torque tube is mounted just below the cockpit floor; it is mounted in and supported by antifriction bearings near each end. There are two rotating stops located on each side of the torque tube just outboard of the bearings. The rotating stops contact adjustable bolt type stops to limit control column travel fore and aft. The longitudinal control system crank is mounted on the left side of the torque tube just outboard of the rotating stop. Rotating motion of the torque tube is imparted to this crank for elevator control. The control columns are mounted on each end of the torque tube (left and right). The torque tube acts as the pivot for the control columns (fore and aft movement implied) to provide elevator control. The control wheels are a "Yoke" type and contains an auto pilot disengage switch in the outboard grips of each wheel. The inboard grips of each wheel contain an ICS switch. Rotating motion of the control wheels is imparted to a torque shaft in the head of each control column. The torque shafts are supported by antifriction bearings at each end, where they mount in the control column head. The center of each shaft has a hollow bore and provides passage for electrical wiring. The torque shafts are linked to a vertical torque tube within the control column by a dual universal arrangement. Each torque tube is supported by and rotates in antifriction bearings. Rotation of the torque tubes imparts motion to cranks mounted at the base of each torque tube. The cranks are mounted and rotate in a horizontal plane, each crank has two arms. The arm facing rearward provides for an interconnect pushrod between the two tubes, within the control columns. The cranks facing outboard on each side provide for lateral control. A stick shaker is mounted on the right lateral crank.

Rotation of the control wheel is limited to $\pm$ 90° by non-adjustable stops within the control column. Fore and aft movement of the control column is limited to 8 inches aft and 5 inches forward approximately from neutral. This is accomplished by the rotating stops on the transverse torque tube contacting the structure adjustable bolt stops.

An electrical conduit runs down the forward face of the control column to provide protection and passage for wiring exiting the control column head.

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CONTROL COLUMNS — MAINTENANCE PRACTICES

1. Control Column/Transverse torque Tube — Removal / Installation

(See Figure 201)

A. Removal

CAUTION: PRIOR TO CONTROL COLUMN REMOVAL ENSURE THAT ELECTRICAL POWER IS OFF.

NOTE: Gust locks may be applied to lock controls during removal of control column. However in removing the transverse torque tube, the gust lock cables must be slacked off to help provide clearance for torque tube removal. In this case rig pins can be used in lateral and longitudinal cranks.

(1) Remove boots and access plates at base of left and right control columns.

(2) Slack off lateral control system cables.

(3) Disconnect lateral system control column, interconnect rod.

(4) Disconnect lateral control pushrods.

(5) Disconnect control column electrical leads.

(6) Disconnect control stick shaker from right lateral crank.

(7) Remove bolts on each side which secure control columns to transverse torque tube.

(8) Remove control columns by working them outboard.

Transverse Torque Tube

(1) Remove access panels on left and right sides of control pedestal, in order to gain access to torque tube.

(2) Remove elevator crank on left side of torque tube by removing attaching bolts and working crank outboard.

(3) Remove left and right rotating stops by removing attaching bolts on each side (which secures them to the torque tube) and working them over the ends.

(4) Remove outer torque tube sections by removing the bolts on the left and right sides that join them (access through the center pedestal). Work each section outboard for removal.

(5) Disconnect engine, trim, and gustlock cables in order to permit removal of center portion of torque tube through floor opening beneath pedestal.

B. Installation

Transverse Torque Tube

(1) Install center portion of torque tube through access in floor below control pedestal.

(2) Install left and right outer torque tube section by moving into place and aligning bolt holes (through access below pedestal) and installing bolts in each side.

(3) Install left and right rotating stops by moving them in place and aligning bolt holes. Install bolts in each stop.

(4) Install elevator crank on left side by moving it into position on torque tube and aligning bolt holes, install attaching bolts.

(5) Connect and tension engine, trim and gust lock cables. Check each system for proper operation.

(6) Install left and right control pedestal access panels.

Control Column Installation

(1) Install control columns by working into place on torque tube and aligning bolt holes. Install attaching bolts in each control column.

(2) Install control stick shaker on the right control column lateral crank.

(3) Connect control column electrical leads.

(4) Install lateral system control column interconnect rod.

(5) Adjust control column longitudinal stops to give column a travel of 5 inches forward and 8 inches aft from neutral.

NOTE: Adjust stop bolts to provide a clearance of 1/16 inch between stop bolts and crank when elevator stops contact.

(6) Install lateral control system pushrods.

(7) Tension lateral control system cables.

(8) Remove rigging pins and restraints.

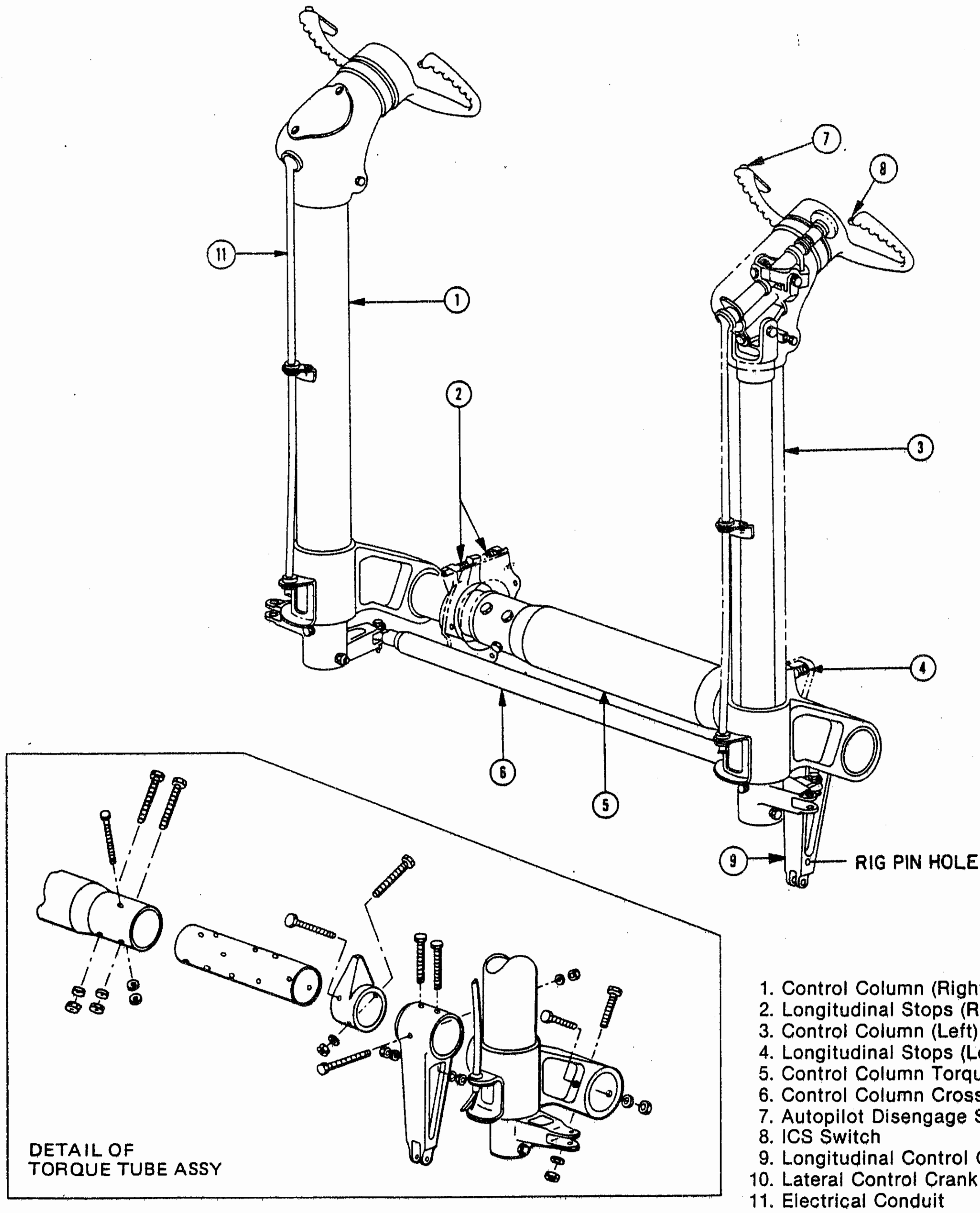
(9) With electrical power ON check operation of the following:

(a) Operation of ICS switches

(b) Operation of auto pilot cutout switches

(10) Inspect all work for proper safetying and installation.

(11) Install access plates and boots at the base of left and right control columns.



1. Control Column (Right)
2. Longitudinal Stops (Right)
3. Control Column (Left)
4. Longitudinal Stops (Left)
5. Control Column Torque Tube
6. Control Column Crossover Push Rod
7. Autopilot Disengage Switch
8. ICS Switch
9. Longitudinal Control Crank
10. Lateral Control Crank
11. Electrical Conduit

Control Column Installation
Figure 201.

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LONGITUDINAL CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

The longitudinal control system maintains longitudinal control of the aircraft by controlling the operation of the aircraft elevators. Each elevator is connected to the horizontal stabilizer at three hinge points. The elevators are of all metal construction, and connect to the longitudinal control system at a torque tube. (See Figure 1). The torque tube connects to a crank assembly at Fuselage Station 717 3/4.

The control system for the elevators is a relatively simple mechanical cable, push-pull rod, operated system, which can be maintained with a minimum of required maintenance. The longitudinal control is fully effective and develops comfortable feel forces at the control columns throughout the operating range. The system is actuated either by motion of the control columns in the cockpit enclosure or by an auto-pilot servomechanism. A manual trim system is provided to relieve stick forces resulting from an out of trim condition.

2. Description

NOTE: The following description is presented in Fuselage Station-wide order starting from the control-column, in the cockpit enclosure and working aft to the elevator.

The pilots control column on the left side of the cockpit is directly connected to the copilots column, by a horizontal torque tube. The torque tube is located at Fuselage Station 100 1/2. Two cranks are mounted on this torque tube at Butt Lines 13.09 left and right of the aircraft centerline. These cranks come in contact with the two stop bolts when maximum forward or aft motion is applied to the control column. The stop bolts are adjusted so as to limit this column motion to 5 inches forward and 8 inches aft. Each crank is designed for one full pilot effort on any control column. An output crank, also mounted on this torque tube at left Butt Line 15 1/4, transmits the pilot or copilot control column motion through a pushrod 26 15/32 inches long to a skewed bellcrank at Fuselage Station 126. The auto-pilot servomechanism is connected by cables to a sector which is mounted on the axis of this skewed bellcrank.

NOTE: The sector is designed for the breaking strength of its cables. This permits the pilot to free the system by loading the cables to their breaking point, if the servomechanism jams.

There are two 3/16 inch diameter cables which are attached to the skewed bellcrank and extend aft for a distance of approximately 35 feet. These cables tie into a horizontally mounted bellcrank at Fuselage Station 544 1/2. An idler installed between the cables at Fuselage Station 458 9/16 changes the direction of these cables for clearance purposes. A second pushrod, 39 3/32 inch long, connects the horizontal bellcrank to an input crank mounted on a torque tube at Fuselage Station 582 7/8.

A pressure box surrounding this input crank has an O-ring seal mounted on the torque tube. This seal prevents any escape of pressure from the cabin enclosure to the unpressurized portion of the fuselage. The output crank of this torque tube is mounted in an unpressurized area.

Between Fuselage Station 582 and 686 are two more pushrods which are connected to a vertical bellcrank at Fuselage Station 640. The elevator horn arrangement at Fuselage Station 717 requires two horn pushrods, each of which has been designed to support full ultimate load. Each horn pushrod originates from an output crank mounted on a torque tube at Fuselage Station 686. The elevator gust lock latch hook at fuselage station 678 engages an arm of the right output crank at Fuselage Station 686, when the gust lock handle in the cockpit is moved aft.

The connection between the elevator torque tube and surface is made through a universal joint which incorporates a stop arm at Stabilizer Station 19 3/4. The universal joint transmits torsion directly into the elevator surface, and also prevents any moment caused by elevator bending deflection from being carried into the torque tube. Two adjustable bolts mounted inside the stabilizer limit the travel of the stop arm, permitting maximum elevator rotation of $+25°$ and $-14°$. The stop arm is part of the elevator assembly and must be installed or removed with the surface.

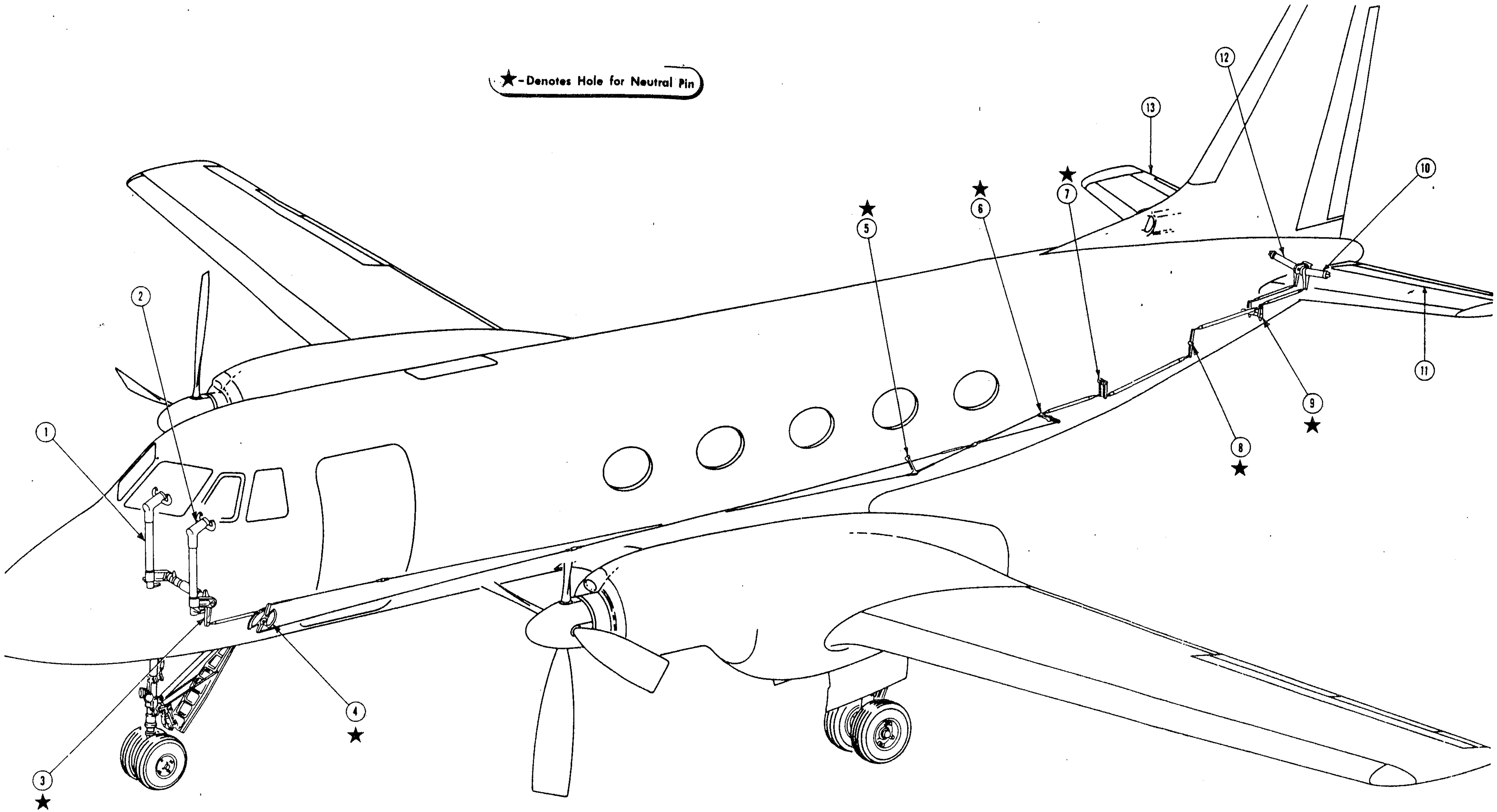
A spring booster is attached to each elevator torque tube to reduce the effort required to move the elevators to extreme positions. When the elevators are at $+14°$, the springs are at dead center and exert no torque to the elevators. From $+14°$, movement of the elevator in either direction is aided by spring tension.

The control cable assemblies are comprised of 3/16 inch stainless steel cables lock-clad to 1/4 inch. The cables are tensioned to 200 pounds $\pm 10\%$ at 70°F., and have a coefficient of expansion comparable to the fuselage structure, thereby minimizing the fluctuation in cable tension arising from extreme changes in temperature. A micarta sleeve is attached to the end of each cable assembly. These sleeves are held with a typical electrical clamp and enable the cable to move freely, ensuring that the lock-clad edge does not jam against part of the structure.

3. **Operation**

In manual operation motion of the control columns, forward or aft, directly lowers or raises the elevators. Since spring tabs or geared balance tabs are not incorporated, the pilot input motion develops a hinge moment sufficient to comfortable control the system. On the ground under a no-load condition, the elevator stops are engaged first. However, in flight, under air loads at full travel of the elevator, only the control column stops are engaged. (Due to stretch in the cable system, maximum loss of travel is 1/13.)

Autopilot operation can be performed with Sperry (SP 20) or Collins autopilot.



1. Copilot's Control Column	8. Fuselage Station 640 Bellcrank
2. Pilot's Control Column	9. Fuselage Station 686 Bellcrank
3. Control Column Longitudinal Crank	10. Left Elevator Torque Tube
4. Fuselage Station 126 Bellcrank	11. Left Elevator
5. Fuselage Station 458 Idler Assembly	12. Right Elevator Torque Tube
6. Fuselage Station 544 Crank Assembly	13. Right Elevator
7. Fuselage Station 582 Bellcrank	

Longitudinal Control System
Figure 1.

LONGITUDINAL CONTROL SYSTEM — MAINTENANCE PRACTICES

1. Longitudinal Control — Rigging

A. Special Tools Required

ITEM	DESCRIPTION
A. Tensiometer	Pacific Scientific 40 - 300 pounds
B. Rigging Pins	1/4 inch diameter bolts AN4 or equivalent
C. Throwboard	No. 159GT1005
D. Tension Scale	0 - 50 pounds Scale

B. Rigging

NOTE: Part numbers are used in this manual only as a means of identification when nomenclature alone is inadequate. This manual must not be used for identifying spare parts by number. Consult the Gulfstream I Illustrated Parts Catalog for this information.

(1) Install lockclad cable assemblies 159C10051-1, 159C10052-1, and 159C10053-1 through holes provided in Fuselage Station 85 and 119 and pass cables through their respective positions. Install cables through nose wheel well. (See Figure 201).

CAUTION: EXTREME CARE SHOULD BE TAKEN WHEN INSTALLING LOCKCLAD CABLE ASSEMBLIES. DO NOT ARC CABLES ON LESS THAN A 60 INCH RADIUS.

(2) With cable assemblies in aircraft connect forward end of cable assembly 159C10053-1 to lower end of crank assembly 159C10101-1.

(3) Install crank assembly 159C10101-1 at Fuselage Station 126 and pin in neutral position.

(4) Install idler assembly 159CM10015-1 at Fuselage Station 458.562 and pin in neutral position.

(5) Install crank assembly 159CM10014-1 at Fuselage Station 544.500 and pin in neutral position.

(6) Install cable assemblies 159C10051-1, 159C10052-1, and 159C10053-1 between Fuselage Station 126 and 544.500. Set cable tensions at 200 pounds $\pm$ 10% (at room temperature, 70°F) and safety turnbuckles.

NOTE: For each 5° above 70°F add 10 pounds to rigging load. For each 5° below 70°F subtract 10 pounds from rigging load.

(7) Install crank assembly 159C10071-1 at Fuselage Station 582.875 and pin in neutral position.

(8) Adjust and install pushrod assembly 159C10050-1.

(9) Install crank assembly 159CM10114-1 at Fuselage Station 640.875 and pin in neutral position.

(10) Adjust and install pushrod assembly 159C10055-1.

(11) Install crank assembly 159C10111-1 at Fuselage Station 686 and pin in neutral position.

(12) Adjust and install pushrod assembly 159C10107-1.

(13) With longitudinal trim tab at 0° trail position in relation to elevator. Use elevator throwboard 159GT1005 at Stabilizer Station 82, set elevator at 3° up position.

(14) Adjust and install pushrod assemblies 159C10050-5 between cranks on elevator torque tubes and crank assembly 159C10111-1.

(15) With columns pinned in neutral position adjust and install pushrod assembly 159C10100-1 between Fuselage Station 100.500 and 126.

(16) Remove all rigging pins and operate system. Set elevator at 25° + 1/2, − 1 1/2° up and set surface stop for this position. Set elevator at 14° + 0 − 2° down and set surface stop for this deflection. Safety surface stops.

(17) Set control column stops (beneath cockpit floor) to give 1/16 inch pinch (i.e., when the surface stops are contacted there is 1/16-inch clearance between column and column stops).

(18) At Fuselage Station 126, install and tension autopilot cables to 45 ± 5 pounds at 70°F and safety cable adjusters.

NOTE: For each 10° above 70°F add 1.5 lbs to rigging load. For each 10° below 70°F subtract 1.5 lbs from rigging load.

(19) Install control column boot assemblies 159C10108-1 on left and right columns.

(20) Cycle control column fully in both directions and check for freedom of motion. Double check elevator deflection versus control column motion (i.e., aft column - up elevator, forward column - down elevator). Without G102-484 springs installed, the control column should return from full aft to full forward position unassisted when released. Cushion column by hand before it hits the forward stops.

(21) If system does not check out, inspect every crank, pushrod, and cable fair lead over entire length of system for interference and binding.

(22) If system checks out, install springs G102-484 between aft side of bulkhead at Fuselage Station 708 and link on elevator torque tube.

(23) Rivet seal cover at Fuselage Station 85.

2. Elevator — Removal / Installation

A. Removal

(1) Secure elevator trim wheel to prevent movement.

(2) Open access cover at trim actuator on elevator.

(3) Open access covers at bottom of trailing edge of stabilizer to gain access to seal strips.

(4) Separate seal strips from stabilizer and disconnect bonding wires.

(5) Loosen trim tab cables and connectors in tail cone (Station 687), install cable clamp at pulleys in tail.

(6) Disconnect turnbuckles on all three cables, attach cord to traverse cables and secure to structure.

(7) Disconnect actuator rods from trim tab actuator.

(8) Unbolt trim tab actuator from elevator, allow actuator to hang from stabilizer.

(9) Disconnect torque tube from elevator at universal joint and bearing end, and remove.

(10) Remove rubber plugs then remove bolts from elevator hinge. (See Figure 204).

(11) Remove elevator.

B. Installation

(1) Install elevator and install bolts.

(2) Attach universal joint to torque tube of elevator and secure bearing end.

(3) Attach trim tab actuator to elevator and connect actuator rods to actuator.

(4) Secure seal to stabilizer with mounting strips and attach bonding wires.

NOTE: Steps (5) thru (7) below are to be performed on elevator replacement only.

(5) Open access covers inboard and outboard of trim actuator in both elevators to gain access to cranks. Install rig pin in each crank.

(6) Set tab neutral at Fuselage Station 82.

(7) Remove rig pins.

(8) Connect cables and remove cable clamps.

(9) Set cable tension in tab cables and transverse cable to 40 to 50 pounds. Safety turnbuckles and cable adjusters.

NOTE: When setting cable tension add 1.5 pounds for each 10°F above 70°F. Subtract 1.5 pounds for each 10°F below 70°F.

(10) With trim indicator set at 0°, check tab neutral and adjust rod as necessary.

(11) Check tab throws:

(a) Elevator trim wheel full fwd:

- Tabs up 2° ± 1/2° (Aircraft 1 - 12, 114 and Aircraft 14 - 25).
- Tabs up 3° ± 1/2° (Aircraft 26 - 112, 322 and 323, Aircraft 115 - 200).
- Elevator trim wheel full aft — tabs down 20° ± 2°.

(b) Inspect area for presents of foreign objects, security of all attachments.

(c) Install access covers and rubber plugs.

3. Elevator Bearing Fitting — Inspection

A. Remove elevator. (See Elevator — Removal / Installation, this Section)

B. Inspect control surface fittings and mating fitting for evidence of corrosion, wear, or chafing. Check for cracks adjacent to bearing and bolt attachment points.

C. Inspect rear beam web and rib attachment angle as follows:

(1) Remove seal attachment.

(2) Open covers on 3 1/2 inch diameter access holes in rear beam web.

NOTE: Use inspection mirror and flashlight.

(3) Inspect for cracks or damage to beam web, rib, rib attaching angle and fasteners.

(4) Inspect external area of rear beam, cap angles, overhang panels (top and bottom) and supporting bracketry.

D. Inspect fittings for irregular surfaces under primer and evidence of corrosion.

E. Inspect bearings for security, corrosion, and roughness when rotated under load.

F. Replace worn, chafed, corroded, deeply scratched, or loose fittings.

G. Replace defective bearings.

H. Install elevator. (See Elevator — Removal / Installation this Section)

4. Elevator Free Play — Inspection

A. Disengage gust lock.

B. Ensure both elevators are against their stops.

CAUTION: DO NOT ATTEMPT TO CHECK FREE PLAY BY HOLDING THE TRIM TAB FOR THESE CHECKS. HOLD THE ELEVATOR AT ITS TRAILING EDGE.

C. Hold one elevator against its stop; attempt to move other elevator.

D. Measure elevator free play at its trailing edge of opposite elevator.

NOTE: The maximum allowable elevator free play is 1/8 inch trailing edge displacement. If this measurement is exceeded an investigation of elevator linkage and torque tube stop fittings will be required. An abnormal movement or sound is an indication of wear.

E. Repeat Step 4.B. thru 4.D. above on other elevator in the up and down positions.

F. Engage gust lock.

5. Elevator Hinge Lateral Play — Inspection

A. Remove elevator hinge access panels.

B. Using a feeler gage, determine vertical play at all elevator hinge attach points. See Figure 202. Maximum allowable elevator hinge vertical play is 0.015 inch.

6. Mass Balance

A. When aircraft is in primed condition, the acceptable mass balance range for elevator is 0 to 12 inch pounds nose heavy. After custom painting, acceptable range is 3 to 20 inch pounds tail heavy. Upper surface is at a constant angle of 4.73° relative to chord plane; therefore a clinometer may be used at any station location.

B. Elevator trim tabs are not mass balanced, but any changes in weight of a trim tab will affect mass balance of elevator.

C. Balance elevator according to the following procedure:

(1) Disconnect torque tube from inboard end of elevator.

(2) Disconnect center hinge by removing hinge pin.

(3) Disconnect trim tab cables from actuator and set tab to 0° trim..

(4) Disconnect curtain from leading edge of the elevator. Replace all hardware that attached curtain to elevator.

NOTE: As an alternative, remove the elevator from the aircraft and support it in a horizontal position from the inboard and outboard hinges. See Mass Balance (Control Surface Static Balance) Section 27 - 0.

(5) Establish a neutral elevator position with a clinometer as shown in Figure 203.

(6) Mark this position, either by reference to a fixed horizontal tail structure or faring or by positioning a piece of cardboard vertically adjacent to the trailing edge and mark a line on it.

(7) Determine whether elevator is nose heavy or tail heavy.

(8) Working at outboard end, determine how much weight has to be added at what distance from hinge line to bring the horn balance into line with the horizontal surface.

(9) Multiply weight added by distance from the hinge line to get out-of-balance hinge moment in inch pounds, nose up or nose down.

NOTE: The exact balance point is not easy to determine because the c.g. of the elevator is displaced vertically from the hinge line. It is easier to obtain the balance point if a weight is hung below the surface rather than placed on top.

(10) Make up a hook with wire (such as clothes hanger wire) and hook it into one of the lightening holes in the horn balance inboard rib and then attach varying weights until a balance is obtained or attach a fixed weight to wire and hook it over lower flange of inboard rib of horn balance. Ensure wire does not rub on horizontal tail.

NOTE: When a small nose up displacement becomes more nose up while a small nose down displacement becomes more nose down, the surface is close to having zero out-of-balance. The added weight must be known within 5% (include the weight of the wire) and the distance from the hinge within 1/4 inch.

D. To determine why the surface is outside the range proceed as follows:

(1) Remove trim tab and weigh it. A painted tab should weight 4.7 to 5.0 pounds. A tab may weigh too much due to extensive repair work. Replacement with a new production tab may be necessary to bring elevator within tolerance.

(2) X-ray entire surface. This will disclose hidden repairs and/or accumulations of water or other foreign matter.

(3) Strip paint and rebalance in unpainted condition. In this condition the surface should be 0 to 12 inch pounds nose heavy. This will determine if excess paint thickness was the cause. Stripping paint should only be done if visual inspection indicates a thick paint film.

(4) Measure skin thicknesses ultrasonically and compare with drawing call outs of all skins on elevator and trim tab as shown on Figure 205.

(5) If cause of the out-of-balance condition still cannot be determined, one of skins will have to be removed by drilling out rivets. This will enable internal structure to be visually inspected.

7. **Elevator Control System — Operational Test**

WARNING: ENSURE PERSONNEL AND EQUIPMENT ARE CLEAR OF ALL CONTROL SURFACES BEFORE OPERATION.

CAUTION: IF BINDING, JAMMING OR SCRAPING OCCUR DURING FOLLOWING PROCEDURES, STOP OPERATION IMMEDIATELY AND INVESTIGATE.

A. Disengage gust lock.

B. Operate elevator in both directions with control column until surface stops are contacted.

C. Operate elevator trim wheel in both directions until surface stops are contacted as follows:

Elevator trim wheel full fwd:

- Tabs up 2° ± 1/2° Aircraft 1 - 25 and 3° ± 1/2° Aircraft 26 - 200, 322 and 323.
- Elevator trim wheel full aft: tabs down 20° ± 2°

D. Check that cockpit indicator corresponds to tab positions.

E. Return tab to 0° on indicator.

8. **Elevator / Elevator Tabs — 20,000 Hr. Inspection**

(See Figure 202 thru Figure 206)

A. Elevator

(1) Remove elevator. (See Elevator — Removal / Installation, this section)

(2)

(3) Inspect hinge fittings and mating fitting for evidence of corrosion, wear, or chafing. Check for cracks adjacent to bearing and bolt attachment points. Horizonal stabilizer supporting structure for elevator.

(4) Inspect fittings for irregular surfaces under primer and evidence of corrosion.

(5) Inspect bearings for security, corrosion, and roughness when rotated under load.

(6) Replace worn, chafed, corroded, deeply scratched, or loose fittings.

(7) Replace defective bearings.

(8) Upper and lower skin between Elevator Station 117.0 and 127.0.

(9) Upper skin, upper cap of front and rear spars and sheer web at front spar between Elevator Station 90 and 110.

Inspect rear beam web and rib attachment angle as follows:

(1) Remove seal attachment.

(2) Open covers on 3 1/2 inch diameter access holes in rear beam web.

NOTE: Use inspection mirror and flashlight.

(3) Torque tube attachment fitting, bellcranks, control rods, lookout mechanism, fairleads, attachments and supporting structure.

(4) Stop fitting, attachments and supporting structure.

(5) Inspect for cracks or damage to beam web, rib, rib attaching angle and fasteners.

(6) Inspect external area of rear beam, cap angles, overhang panels (top and bottom) and supporting bracketry.

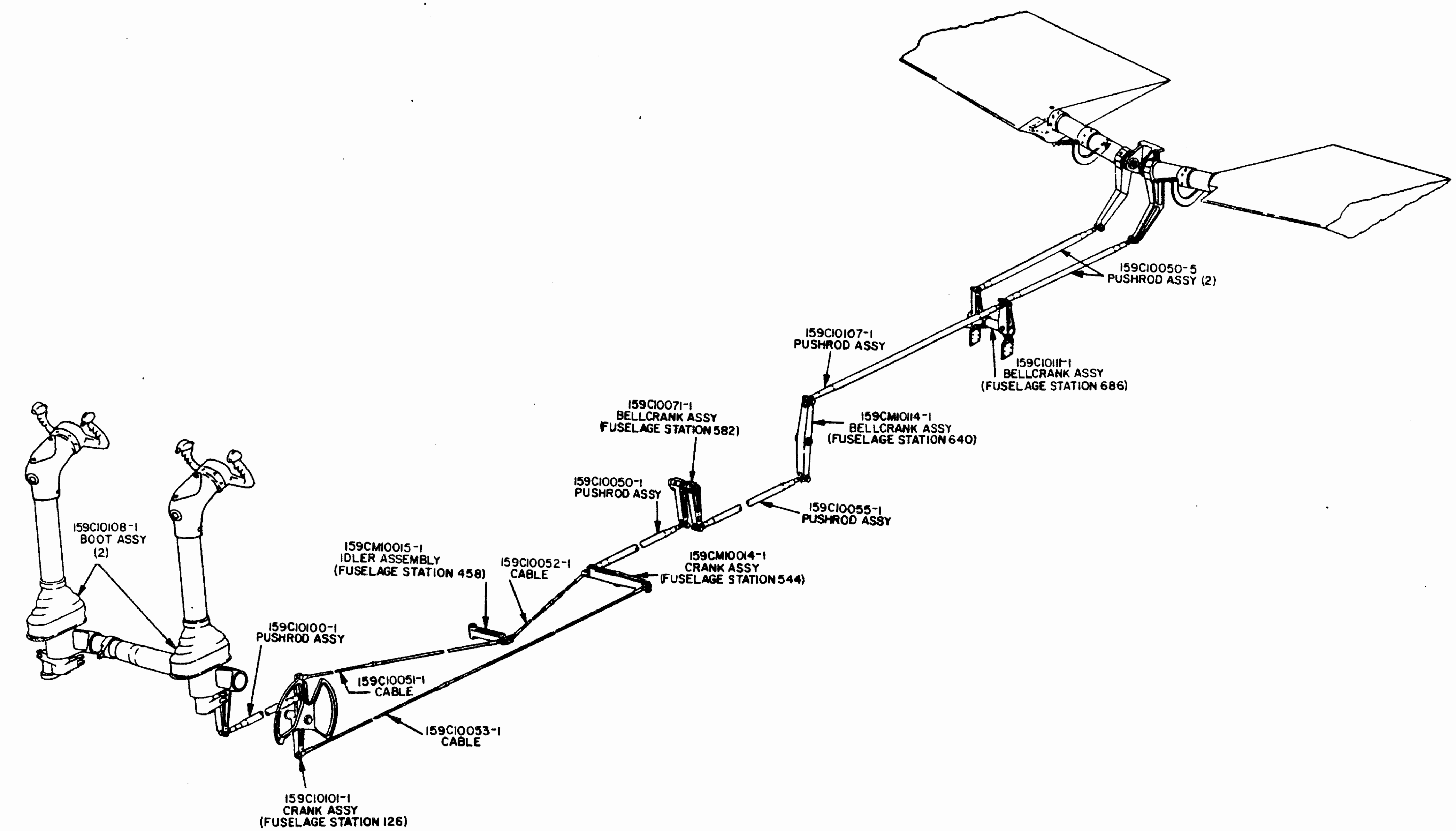
B. Elevator Trim Tab

(1) Inspect elevator trim tab hinge fittings, attachments and supporting structure.

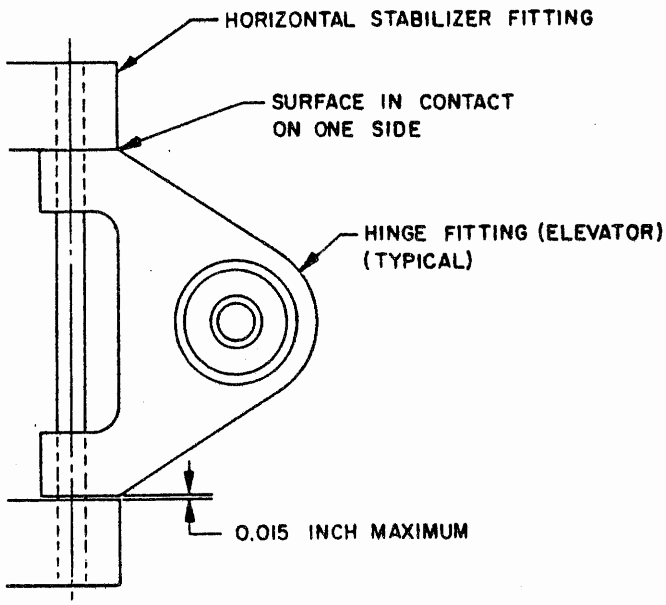
(2) Replace worn, chafed, corroded, deeply scratched, or loose fittings.

(3) Replace defective bearings.

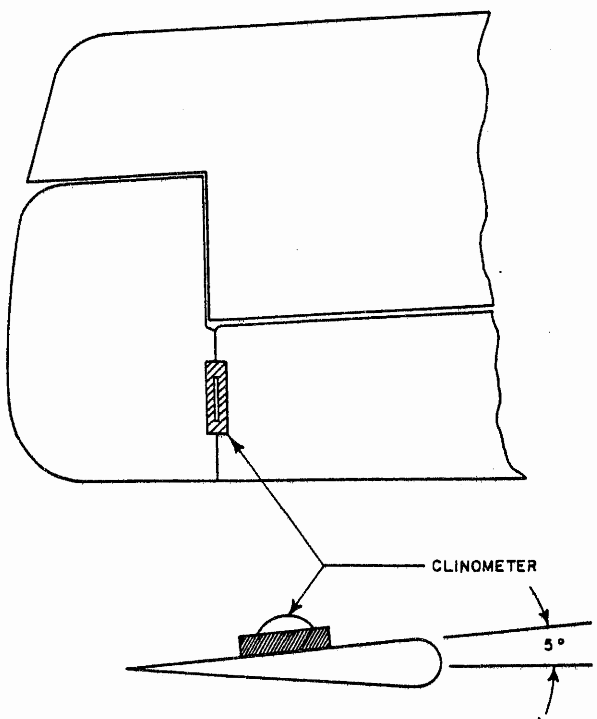
(4) Install elevator. (See Elevator — Removal / Installation, this section)



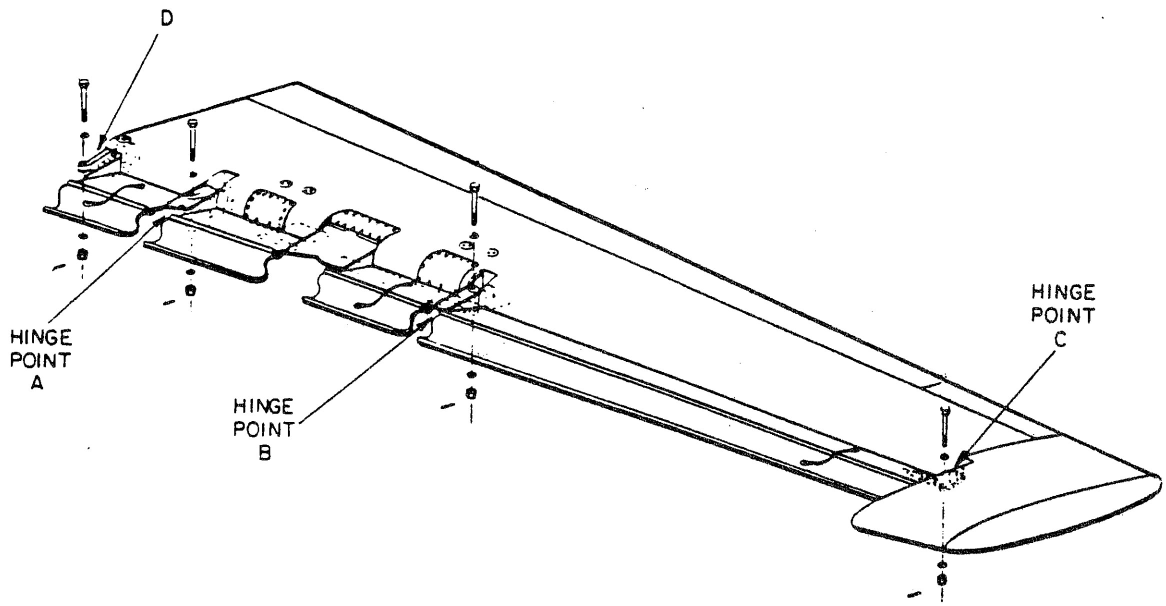
Longitudinal Control System Installation
Figure 201.



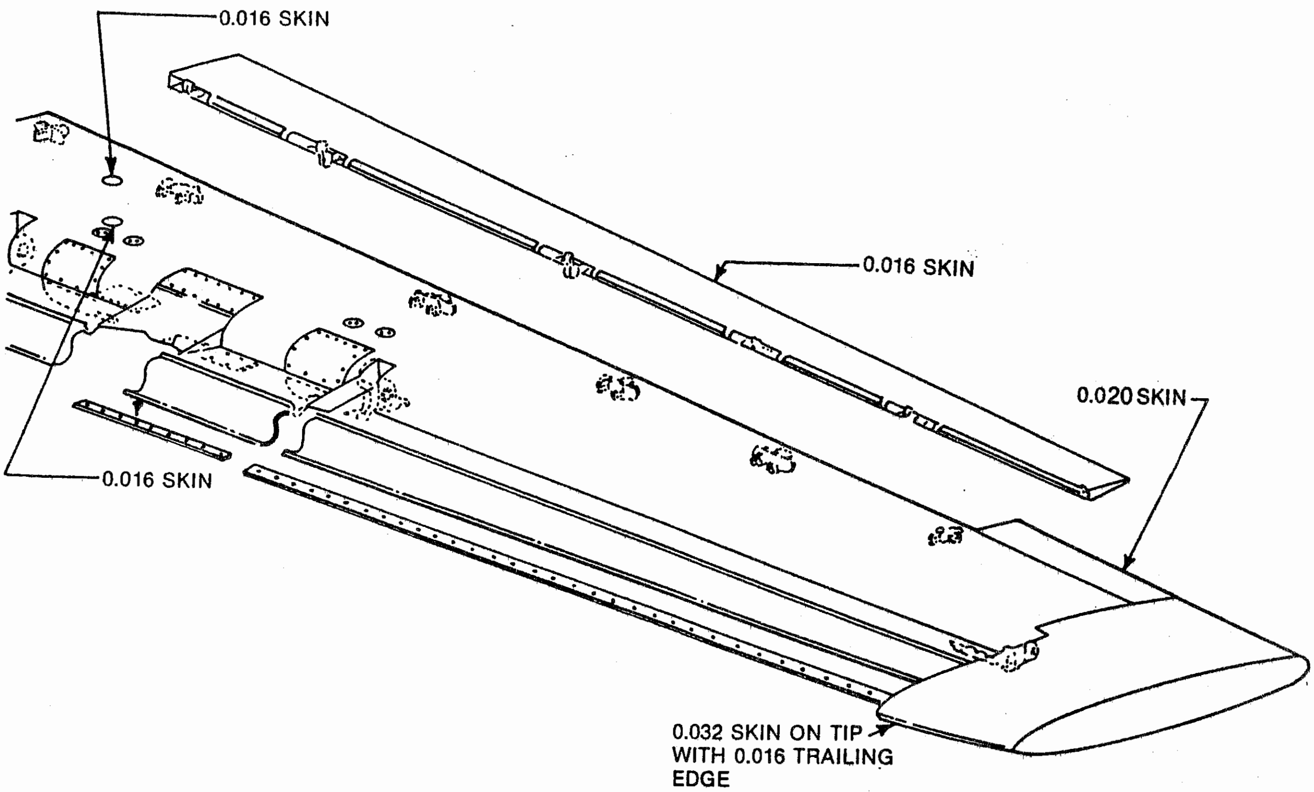
Elevator Hinge (Typical)
View Looking Inboard
Figure 202.



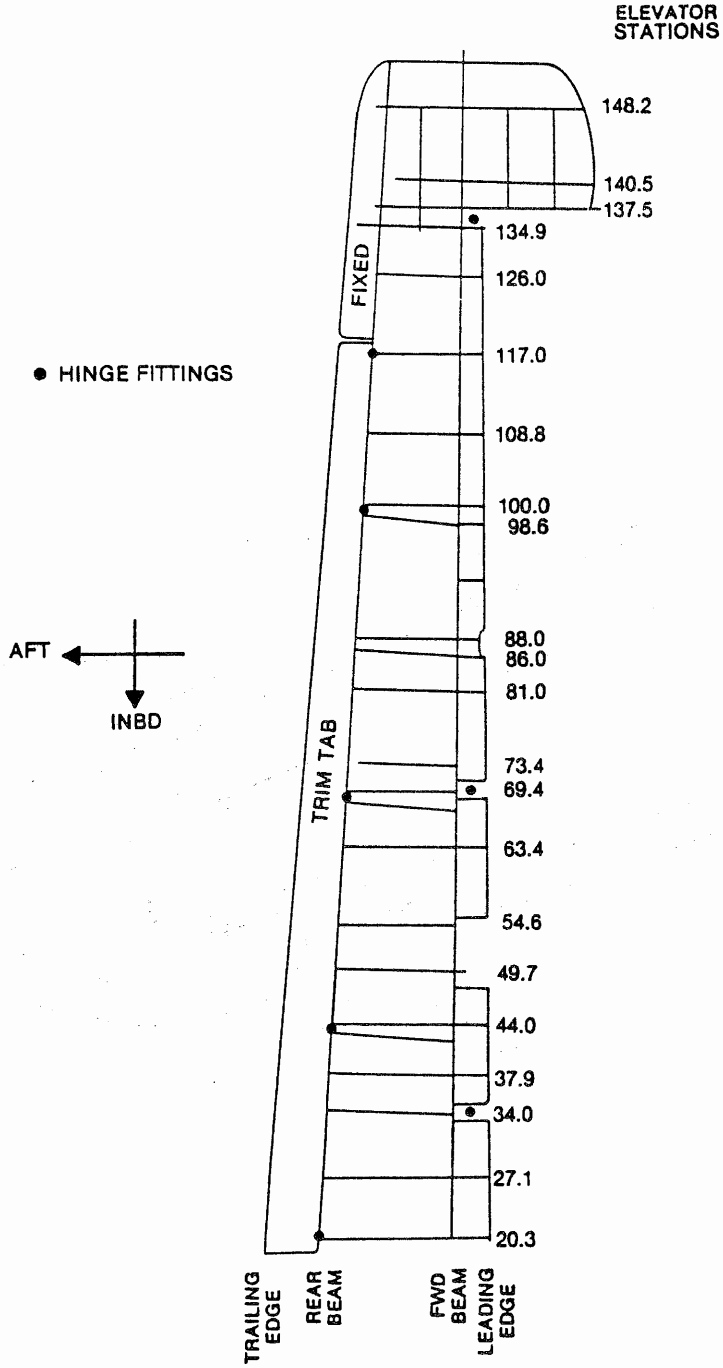
Clinometer Placement On Elevator
Figure 203.



Elevator Hinge Points
Figure 204.



Elevator Skin Thicknesses
Figure 205.



Elevator/Elevator Tab
Figure 206.

LATERAL CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

(See Figure 1)

Lateral control of the aircraft is provided by the ailerons, which are connected to the wings at four hinge points. The ailerons are of all-metal construction and are mass balanced. The control system (a mechanical, cable and push-pull rod, operated system) is actuated by movement of either control wheel located in the cockpit. Operation of system may also be accomplished by an auto pilot servomechanism. A manually operated trim tab is installed on the left aileron. Spring tabs are installed on both ailerons. Manual effort is transmitted to the ailerons from the control wheels by a system of rods, bellcranks and cables. From the control column torque tubes, rods connect to bellcranks on the left and right sides of the fuselage. Cables then lead aft to two jackshafts, one on each side, with the jackshafts being tied together by a transverse pushrod. A pushrod leads from each jackshaft to the wing inboard bellcranks. Cables lead from this bellcrank along the face of the front spar to the wing outboard upper and lower bellcranks. Tie rods running fore and aft connect the outboard bellcranks to the wing rear beam bellcrank which is then connected with a single pushrod to the aileron spring tab bellcrank. Two pushrods connect this bellcrank to the aileron spring tab. A pair of spring bungees are mounted between this crank and the aileron. This arrangement provides spring tab rotation of 15° up or down initially when the controls are moved, after which any further control movement is applied to the aileron by stop bolts. Control wheel rotation of 23 1/2° causes full travel of the spring tab providing that the aileron remains stationary.

NOTE: The following description is presented in station-wide order starting from the cockpit enclosure and working toward the aileron surfaces.

2. Control Column

The pilots control column wheel on the left side of the fuselage center-line transmits pilot effort directly to its control column torque tube by means of a double universal joint. This mechanism contains an internal stop which limits the wheel rotation to ± 90° Mounted on the axis of each torque tube is a crank whose output motion is transmitted through a pushrod to a vertical bellcrank at Fuselage Station 100.65. The point of connection between the crank and the pushrod is coincident with the axis of the horizontal torque tube connecting the two control columns when the crank is in a neutral position. This connection is made in order to minimize actuation of the lateral control components when forward or aft motions are applied to either of the two control columns.

3. Autopilot Connection

The autopilot servomechanism is connected by cables to a sector which is mounted on the axis of the right vertical bellcrank. The sector is designed for the breaking strength of its cables. This permits the pilot to free the system by loading the cables to their breaking point in the event the servomechanism should jam.

4. Fuselage Control Cables

Attached to the output arm of each vertical bellcrank is a 3/16 inch diameter 7 x 19 cable extending aft for a distance of approximately 15 feet. Each cable ties into a horizontal crank mounted on the axis of a vertical jackshaft at Fuselage Station 280. Each cable has a 200 pound preload which reduces the tendency of one cable to become slack due to an increase in tension of the opposite cable. A clockwise rotation applied to either control wheel produces tension in the right cable, while a counterclockwise rotation produces tension in the left cable. (Each cable can be considered the "down" cable for its respective aileron.)

5. **Jackshafts and Seals**

Each jackshaft at Fuselage Station 280 11/16 has an output arm attached to it which transmits the pilot or copilot effort into its corresponding wing. A pressure box surrounding each output arm has an O-ring seal mounted on its corresponding jackshaft torque tube. The seals prevent any escape of pressure from the cabin enclosure to the unpressurized portion of the fuselage.

6. **Crossover Pushrods**

The continuity between the lateral control components on the left side of the fuselage centerline and those on the right is achieved through 2 horizontal crossover pushrods at Fuselage Station 92 and 287. The first is mounted between the two control-column torque tube cranks and transmits motion directly from one control wheel to its opposite. The second crossover pushrod is mounted between the two jackshaft assemblies and transmits motion from one jackshaft to its opposite.

The pushrod between the columns is subjected to both tension and compression. The pushrod between the jackshaft assemblies is subjected to compression only. To facilitate installation, the jackshaft assembly pushrod is fabricated in three sections.

NOTE: The components of the lateral control system proceeding outboard from each jackshaft and ending at the aileron spring tab fittings are similar for each wing. The following description is for those components on the left side of the fuselage centerline.

7. **Inboard Pushrod and Bellcrank**

A pushrod extends outboard from the output arm of the jackshaft and ties into a vertical bellcrank at Butt Line 37.

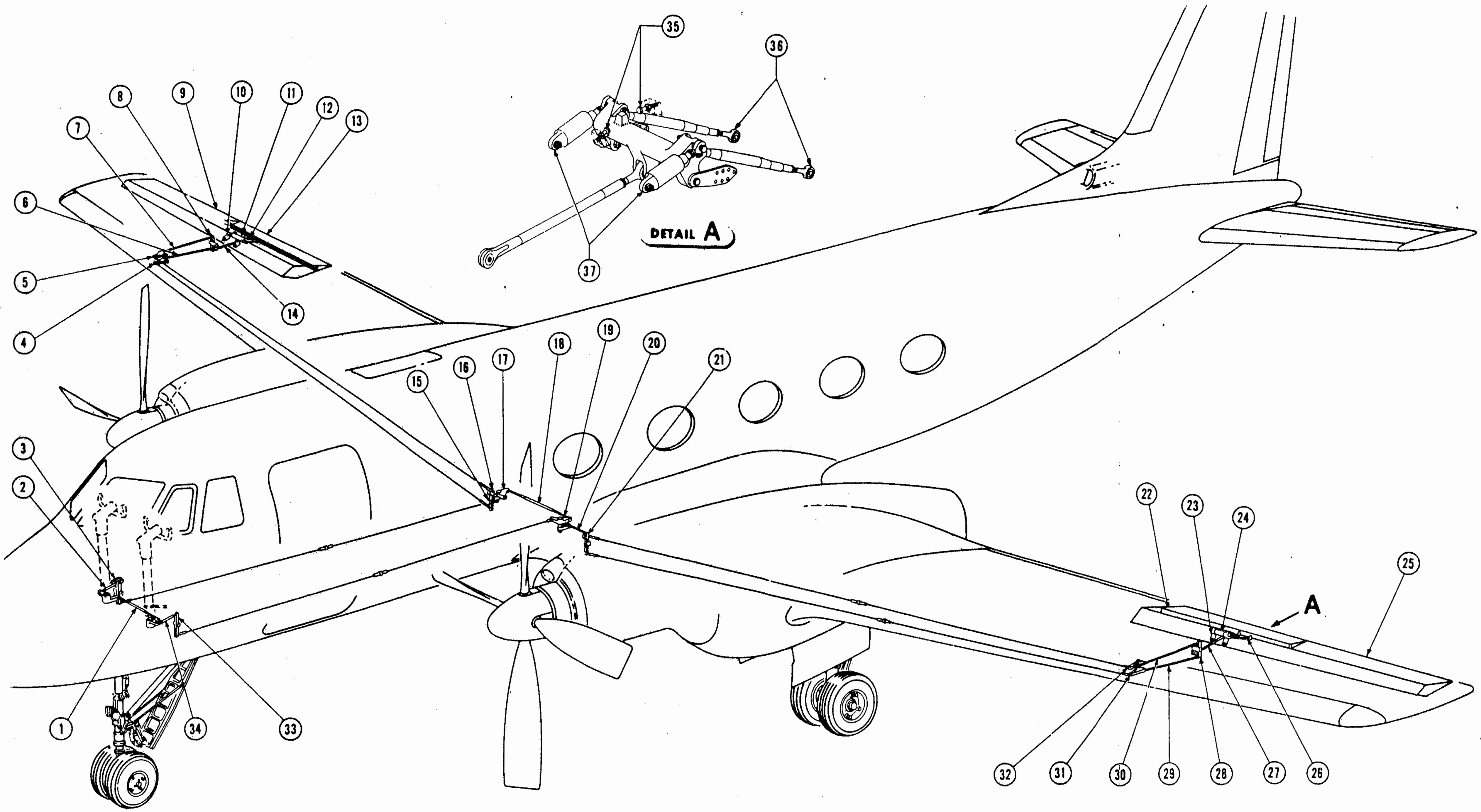
A 5/32 inch 7 by 9 stainless steel cable, lockclad to 1/4 inch, is attached to each end of the bellcrank, and is preloaded with 200 pound tension. The cables extend outboard into the wing for a distance of approximately 26 1/2 feet and tie into a horizontal crank at Wing Station 353.

8. **Outboard Linkage**

The horizontal cranks at Wing Station 353 are mounted approximately six inches apart on two separate brackets and rotate about the same vertical axis. The output arm of each crank is connected to a tie rod which in turn transmits motion aft into a single vertical bellcrank at approximately 53 percent of the wing chord.

9. **Aileron and Spring Tab**

The aileron pushrod transmits motion from the vertical crank directly into the aileron spring tab crank located at 74 1/2 percent of the wing chord. The aileron spring surface is actuated by two 12 1/2 inch long rods, each of which is connected to an aileron horn fitting. The rods are actuated by output arms mounted on the aileron spring tab crank. When the tab surface reaches its maximum travel of plus or minus 15°, a lug attached to the crank comes in contact with two tab stop bolts mounted in the aileron structure. Assuming that the aileron surface remains stationary as the tab rotates, the rotation of the control wheel is 23 1/2° at full travel of the tab. After either stop is engaged and the spring tab reaches its maximum travel, further pilot input to the aileron spring tab crank causes the entire aileron to rotate about its own axis. Two spring bungees, each of which is mounted between the output arm of the aileron spring tab crank and the leading edge of the aileron surface, brings the spring tab back to its neutral position when pilot effort to the control wheel is removed. Each bungee is designed for a maximum output of 40 pounds at full deflection of approximately plus or minus 5/8 inch. The two stop bolts for the aileron are mounted on the aileron stop fitting at 66 percent of the wing chord. These bolts come in contact with two stop lugs mounted in the wing structure when the aileron surface reaches its maximum travel. The bolts are adjusted so as to limit the aileron travel to 16° up and 12° down. Each aileron incorporates a spring tab; however, the trim is located on the left aileron only. Preload of the spring bungee reflects in a preload of two pounds at the control wheels and builds up at a rate of three pounds per degree of tab travel.



1. Control Column Crossover Push Rod	10. Spring Tab Bungees (2)	19. Fuselage Jackshaft (Left)	28. Wing Rear Beam Bellcrank (Left)
2. Control Column Push Rod (Right)	11. Aileron Spring Tab Crank (Right)	20. Inboard Push Rod (Left)	29. Wing Tie Rod (Lower)
3. Sta 100 Bellcrank (Right)	12. Spring Tab Push Rods (2)	21. Inboard Bellcrank (Left)	30. Wing Tie Rod (Upper)
4. Outboard Lower Bellcrank (Right)	13. Aileron Spring Tab (Right)	22. Aileron Spring Tab (Left)	31. Outboard Lower Bellcrank (Left)
5. Outboard Upper Bellcrank (Right)	14. Aileron Push Rod (Right)	23. Spring Tab Bungees (2)	32. Outboard Upper Bellcrank (Left)
6. Wing Tie Rod (Lower)	15. Inboard Bellcrank (Right)	24. Aileron Spring Tab Crank (Left)	33. Sta 100 Bellcrank (Left)
7. Wing Tie Rod (Upper)	16. Inboard Push Rod (Right)	25. Left Aileron	34. Control Column Push Rod (Left)
8. Wing Rear Beam Bellcrank (Right)	17. Fuselage Jackshaft (Right)	26. Spring Tab Push Rods (2)	35. Spring Tab Stops
9. Right Aileron	18. Fuselage Crossover Push Rod	27. Aileron Push Rod (Left)	36. Spring Tab Push Rods
			37. Spring Bungees

Lateral Control System
Figure 1.

LATERAL CONTROL SYSTEM — MAINTENANCE PRACTICES

1. Lateral Control — Rigging

A. Special Tools Required

ITEM	DESCRIPTION
A. Tensiometer	Pacific Scientific 30 - 300 pounds.
B. Rigging Pins	1/4 inch and 5/16 inch diameter bolts, AN4 or equivalent
C. Throwboards	159GT1006
D. Tension Scale	0-50 pounds scale

B. Rigging (See Figure 201)

NOTE: Part numbers are used in this manual only as a means of identification when nomenclature alone is inadequate. This manual must not be used for identifying spare parts by number. Consult the Gulfstream I Illustrated Parts Catalog for this information.

(1) Install crank assembly 159CM10221-1 at Butt Line 37 left and front beam of wing and pin in neutral position.

(2) Install bellcrank assembly 159C10222-1 at Fuselage Station 280 on left side of aircraft and pin in neutral position.

(3) Adjust and install pushrod assembly 159C10070-3.

(4) With tie rod assemblies 159C10213-1 and crank assemblies 159CM10201-1 and 159CM10202-1 inside wing, connect forward ends of tie rods to cranks. Install crank assemblies 159CM10201-1 and 159CM10202-1 at Wing Station 353 and pin in neutral position.

(5) Install crank assembly 159CM10203-1 at Wing Station 357 and pin in neutral position.

(6) Adjust and install tie rod assemblies 159C10213-1.

(7) Rig spring tab while aileron is on the ground. Install crank assembly 159CM10401-1, including adjustable rod end from pushrod assembly 159CM10219-1, in aileron and pin in neutral position.

(8) Using spring tab throwboard 159GT1006, set spring tab at 0° trail position.

(9) Adjust and install pushrod assemblies 159C10070-1, and bungee assemblies 159C10404-1.

(10) Adjust stop bolts for 15° + 1 degree up and down spring tab travel, and safety.

(11) Using aileron throw board 159GT1006 and spring tab throwboard, set aileron at 0° trail position.

(12) Adjust and install pushrod assembly 159CM10219-1 and remove neutral pin from crank assembly 159CM10401-1.

(13) Remove pins from crank assemblies 159CM10201-1 and 159CM10202-1 and install cable assemblies 159C10205-1 (upper cable) and 159C10205-3 (lower cable). Tension cables to 200 pounds ± 10% (at room temperature, 70° safety turnbuckles.

NOTE: For each 5° above 70°F add 10 pounds to rigging load. For each 5° below 70°F subtract 10 pounds from rigging load.

CAUTION: LOCKCLAD CABLE ASSEMBLIES MUST BE FED FROM OUTBOARD END OF WING AND CARE MUST BE TAKEN NOT TO ARC THEM ON LESS THEN A 60-INCH RADIUS.

(14) Repeat Steps B.(1) thru B.(13) for right wing.

(15) With both wings rigged and pinned determine length of crossover rod assembly 159C10225-1 between crank assemblies 159C10222-1 and -2.

(16) Remove remaining rigging pins in wings.

(17) Lengthen crossover rod assembly 159C10225-1, one half turn of adjustable end, from length determined in step 15 and install.

(18) Install crank assembly 159C10221-1 at Fuselage Station 100 on right side of aircraft and pin in neutral position.

(19) Install crank assembly 159CM10216-1 at Fuselage Station 100 on left side of aircraft and pin in neutral position.

(20) Install pushrod assemblies 159C10204-1 between crank assemblies 159C10221-1 and 159CM10216 and cranks on lower end of columns.

Remove neutral pins.

(21) Determine length of crossover push rod assembly 159C10058-1 required to satisfy step 20. Remove, shorten two full turns and install.

(22) Pin crank assembly 159C10222-2 (right side of aircraft at Fuselage Station 280).

(23) Install cable assemblies 159C10206-1 and tension to 200 pounds ± 10% (at room temperature, 70°F) and safety turnbuckles.

NOTE: For each 5° above 70°F add 10 pounds to rigging load. For each 5° below 70°F subtract 10 pounds from rigging load.

(24) At Fuselage Station 100, install and tension autopilot cables to 45 ± 5 pounds and safety cable adjusters.

NOTE: For each 10° above 70°F add 1.5 pounds to rigging load. For each 10° below 70°F subtract 1.5 pounds from rigging load.

(25) Remove all rigging pins and cycle control wheels full left and full right (travel should be 90° ± 2°). Check for freedom of motion. Double check aileron deflection versus wheel motion (wheel left, left aileron up: wheel right - right aileron up).

(26) Using aileron throwboard 159GT1006 set aileron surface stops, with wheel full left, left aileron should be 16° ± 1° up; right aileron should be 12° ± 1° down. With wheel full right, right aileron should be 160 +10 up; left aileron should be 12° ± 1° down.

NOTE: All rigging holes are 1/4 inch diameter except hole in crank 159CM10401-1 which is 5/16 inch diameter. Use AN4 bolts or equivalent except for the noted crank. In the event of an excessive right wing heavy condition, (in excess of 5° trim), droop both ailerons 1° and rerig left trim tab 2 to 2 1/2° down with cockpit trim indicator at 0° readings as follows:

(a) Install rigging pins in crank assembly 159CM1023-1 (left wing) and 159CM10203-2 (right wing).

(b) Using aileron throwboard 159GT1006, adjust pushrod 159CM10219 for 1° down aileron (left and right).

(c) Re-rig left trim tab 2 to 2 1/2° down with cockpit trim indicator at 0° reading.

(d) Remove rigging pins and cycle control wheel left and right to check for freedom of movement.

(e) Cycle lateral trim control in cockpit, while checking for freedom of movement.

(27) If procedure outlined in above note does not correct a right wing heavy condition, following alternate method may be applied:

(a) Re-rig left trim tab to 0°.(Trim control knob in cockpit also at 0°)

(b) Rig left spring tab 3° down by adjusting pushrods 159C10070-1, and right spring tab 1° up.

2. Aileron Hinge Vertical Play — Inspection

A. Remove hinge access panels.

B. Using a feeler gage, determine aileron hinge vertical play at all aileron hinges as shown in Figure 202.

Maximum allowable vertical play is 0.015 inch.

3. Aileron Mass Balance

When aircraft is in primed condition, acceptable mass balance range for aileron is 0 to 12 inch pounds nose heavy. After custom painting acceptable range is 8 to 20 inch pounds tail heavy. Aileron is of an asymmetric section; therefore, it is recommended it be mounted right side up for balancing. Upper surface of aileron should be at an angle of 11.5° relative to a level plane. A clinometer must be placed on the outboard end of the aileron only. See Figure 203.

Aileron spring tab does not require mass balancing because of high spring stiffness and dual control rod linkages. Aileron trim tabs also do not require mass balancing. Any weight changes to aileron spring tab or aileron trim tab, however, must be considered in weight changes to these tabs may affect aileron mass balance. When mass balancing aileron, tabs must be in place and in neutral condition.

NOTE: To mass balance aileron with surface off aircraft, See Mass Balance Section 27-0, To mass balance the aileron with the surface on the aircraft, proceed as follows:

A. Disconnect primary control pushrod at its aft end.

B. Remove the hinge pins from hinges at Wing Stations 364 and 402.

C. Disconnect fabric curtain between leading edge and wing rear beam at aileron edge.

D. Replace all connecting hardware in its proper location (screws may be inserted finger tight).

E. Disconnect trim tab actuator cables and unwind from drum. Tab should be in neutral position. Alignment of aileron trailing edge with wing tip and fully retracted flap determines neutral position.

F. Determine whether surface is nose or tail heavy.

G. If surface is tail heavy, apply balance weights forward of hinge line to balance surface. For surfaces with a horn, such as rudder and elevator, work at outboard end since adding weight to horn gives largest moment arm and lowest weight. If surface is nose heavy, it may be balanced from either end, since ample moment arm exists.

H. Place weights forward or aft of hinge as required.

NOTE: Out-of-balance moment is product of weights added times its distance from hinge line. Use either a fixed known weight and obtain a balance by varying distance from hinge line or have a fixed attachment for weight and vary amount of weight.

Placing weights on top of surface is less satisfactory than hanging them below surface since with chord horizontal in the supposedly balanced condition, c.g. of surface and weight may be above hinge point causing an unstable condition. Ideal equilibrium position is one in which a small nose up displacement becomes more nose up and a small nose down displacement becomes more nose down.

A simple method of hanging small weights to leading or trailing edge is shown in Figure 204. Everything shown in Figure 204. should be included in weight when calculating the out-of-balance moment.

I. Measure moment arm from c.g. of weight to hinge line to within 0.25 inch. Measure weight to within 0.10 pound.

NOTE: If the surface is being rebalanced because of weight added for repair to known damage, and it is found that extra balance is required, then additional weight must be added at same spanwise station as repair. Requirements are weight must not interfere with (or reduce) surface angular movement and attachment of weight and back-up structure to which it is attached must be strong enough to meet required 'g' forces. It is recommended GAC Engineering assistance be sought in this situation.

J. After a surface has been balanced record actual out-of-balance value in inch pounds in aircrafts log book, with date and signature of responsible inspector.

4. Aileron — Removal / Installation

A. Removal

NOTE: Before removal check and record throws.

(1) Open lower trailing edge wing access panels and hand hole cover at aileron crank.

(2) Disconnect aileron pushrod at forward end, break jam nut at aft end and unscrew forward part of pushrod while leaving aft part attached to aileron.

(3) Separate aileron leading edge seals from wing.

(4) Disconnect bonding wires.

(5) For left aileron only:

(a) Remove outboard wing leading edge.

(b) Lower trim tab cable tension and install cable clamp at any convenient station.

(c) Disconnect cable and remove pulley wheels from bracket.

(d) Attach cord to cables leading to aileron and pull cord through wing, detach and secure cords to structure.

(6) Remove rubber plugs to remove hinge bolts.

(7) Remove aileron.

B. Installation

(1) Install aileron, attach cables to cord and pull tab cables through wing.

(2) For left aileron, attach cables to cord and pull tab cables through wing.

(3) Install pulleys and connect cables; tension cables to 45 $\pm$ 5 pounds and safety-wire turnbuckle.

NOTE: For each 10° above 70°F, add 1.5 pounds to rigging load. For each 10° below 70°F, subtract 1.5 pounds from rigging load.

(4) Install six rubber grommets at hinge bolt openings; attach leading edge seals from aileron to wing.

(5) Install aileron pushrod.

(6) Insert rigging pin at crank assembly (Wing Station 357) and adjust aileron pushrod until aileron is neutral 1° down $\pm$ 1°; tighten pushrod locknut.

(7) Remove rig pin and using throwboards, check throws of aileron:

(a) Control wheel full left - Left aileron 16° $\pm$ 1° Up
- Right aileron 12° $\pm$ 1° Down

(b) Control wheel full right - Right aileron 16° $\pm$ 1° Up
- Left aileron 12° $\pm$ 1° Down

(8) Check trim tab for full travel and neutral:

(a) Trim control wheel full counterclockwise - Tab down 20° $\pm$ 2°.

(b) Trim control wheel full clockwise - Tab up 20° $\pm$ 2°.

5. Aileron Control System — Operational Test

WARNING: ENSURE PERSONNEL AND EQUIPMENT ARE CLEAR OF ALL CONTROL SURFACES BEFORE OPERATION.

CAUTION: IF BINDING, JAMMING OR SCRAPING OCCUR DURING FOLLOWING PROCEDURES, STOP OPERATION IMMEDIATELY AND INVESTIGATE.

A. Disengage gust lock.

B. Operate aileron in both directions with control wheel until surface stops are contacted.

C. Operate aileron trim wheel in both directions until trim wheel stops are contacted as follows:
   (1) Aileron trim wheel left - tab down 20° ± 2°.
   (2) Aileron trim wheel right - tab up 20° ± 2°.

D. Ensure cockpit indicator corresponds to tab position.

E. Return tab to 0° on indicator.

F. Engage gust lock.

G. Operate aileron control wheel in both directions to check spring tab throws against stops.

   NOTE: Ailerons will not move but spring tabs will go to full travel.

6. Aileron / Aileron Tabs — 20,000 Hr. Inspection

A. Open access panels shaded in Figure 205.

B. Inspect aileron seals for condition and security.

C. Inspect aileron internal for evidence of corrosion.

D. Aileron, aileron trim tab and spring tabs for evidence of corrosion, clearance and travel.

   NOTE: Visually check the clearance between the tabs and the main surface for the aileron trim and aileron spring tabs. This check should be made for the full range of tab travel in the case of the trim tab and the full range of aileron travel in the case of the aileron spring tabs. The gap should be at least 0.060 inch all along, top and bottom through the whole range of travel. If the gap is less than this value bend the trailing edge of the aileron skin up on the upper surface or down on the lower surface until the gap is at least 0.060 inch. No special equipment is necessary to do this.

E. Aileron, aileron spring tabs, aileron trim tab, trim tab actuator, tab linkages, mechanisms, control rods and cables for condition, security and freedom of operation. See 27-0 for allowable limits of flight control cables.

F. Aileron and aileron spring tab attachment fittings and bearings for condition, security and corrosion.

G. Static discharge wicks for condition, security and corrosion.

H. Inspect for presence of foreign objects then close access panels.

7. Aileron Bearings / Fittings — Inspection

A. Remove Aileron. (See Aileron — Removal / Installation, this section).

   WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN KEEP AWAY FROM OPEN FLAME.

B. Thoroughly clean areas to be inspected using cleaning solvent or equivalent.

C. Carefully inspect control surface fittings and mating fitting for evidence of corrosion, wear or chafing;

D. check for signs of cracks, especially at areas adjacent to bearing and bolt attachment points.

E. Carefully check fittings for irregular surfaces under primer and for evidence of granular corrosion.

F. Inspect bearings for security of mounting, evidence of rust, and for roughness when rotated under load.

G. Replace fittings that are worn, chafed, corroded, deeply scratched, or loose.

H. Replace bearings that are loose, rusted, or rough when rotated.

I. Install Aileron. (See Aileron — Removal / Installation, this section).

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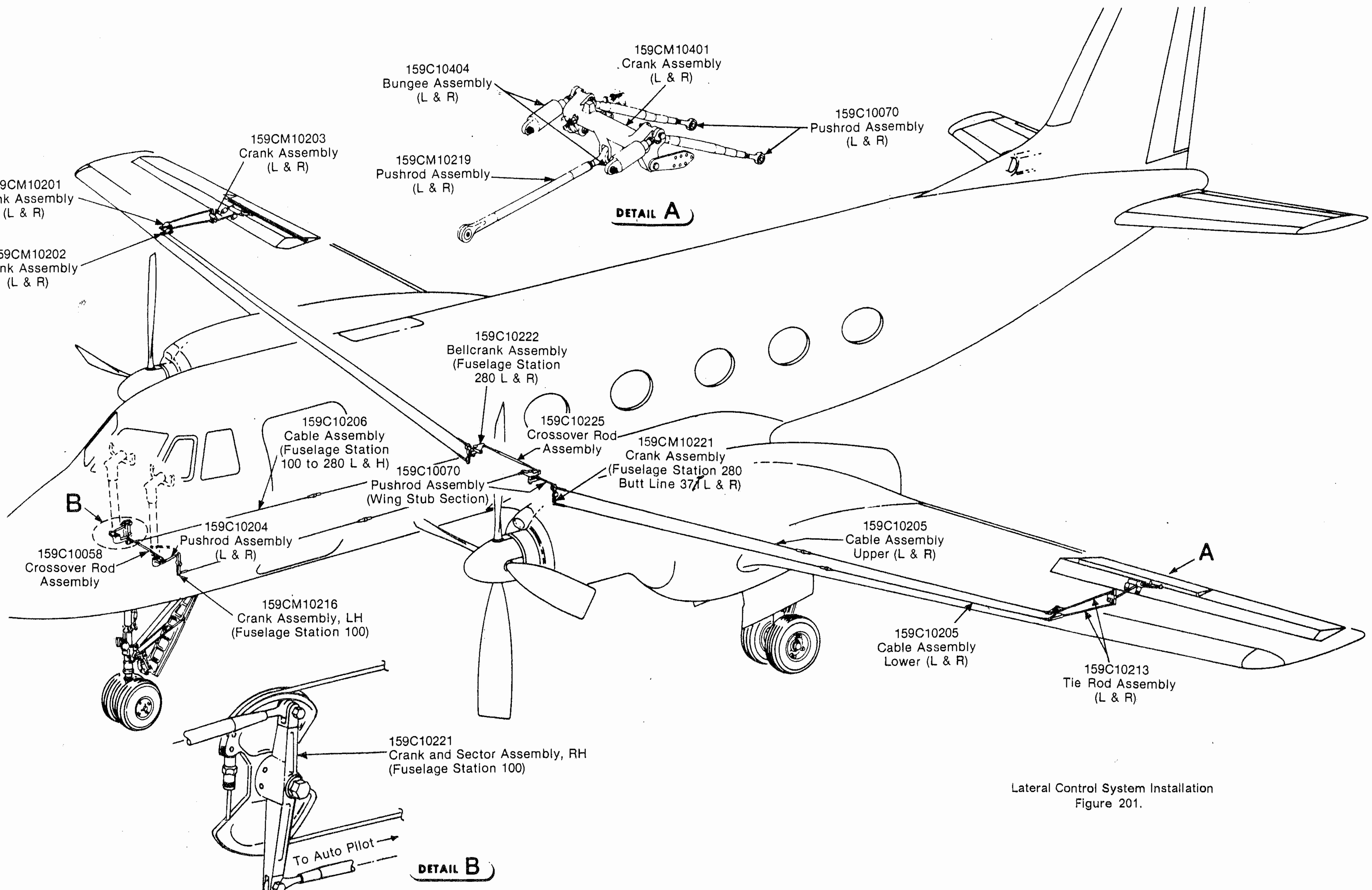
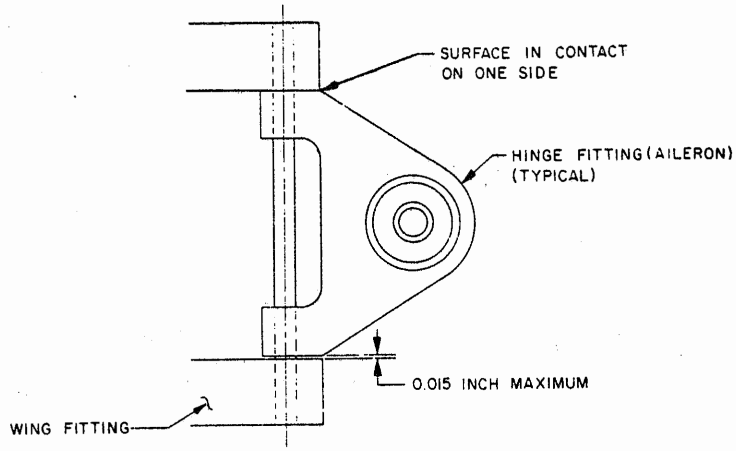
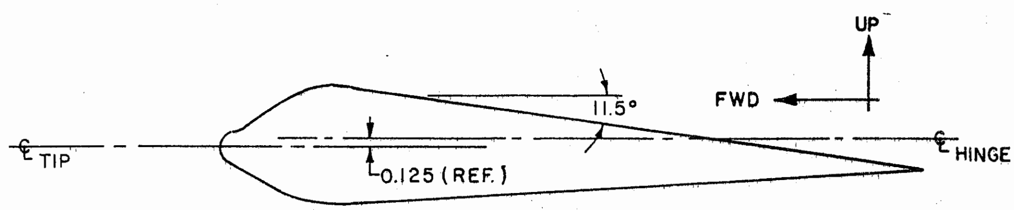


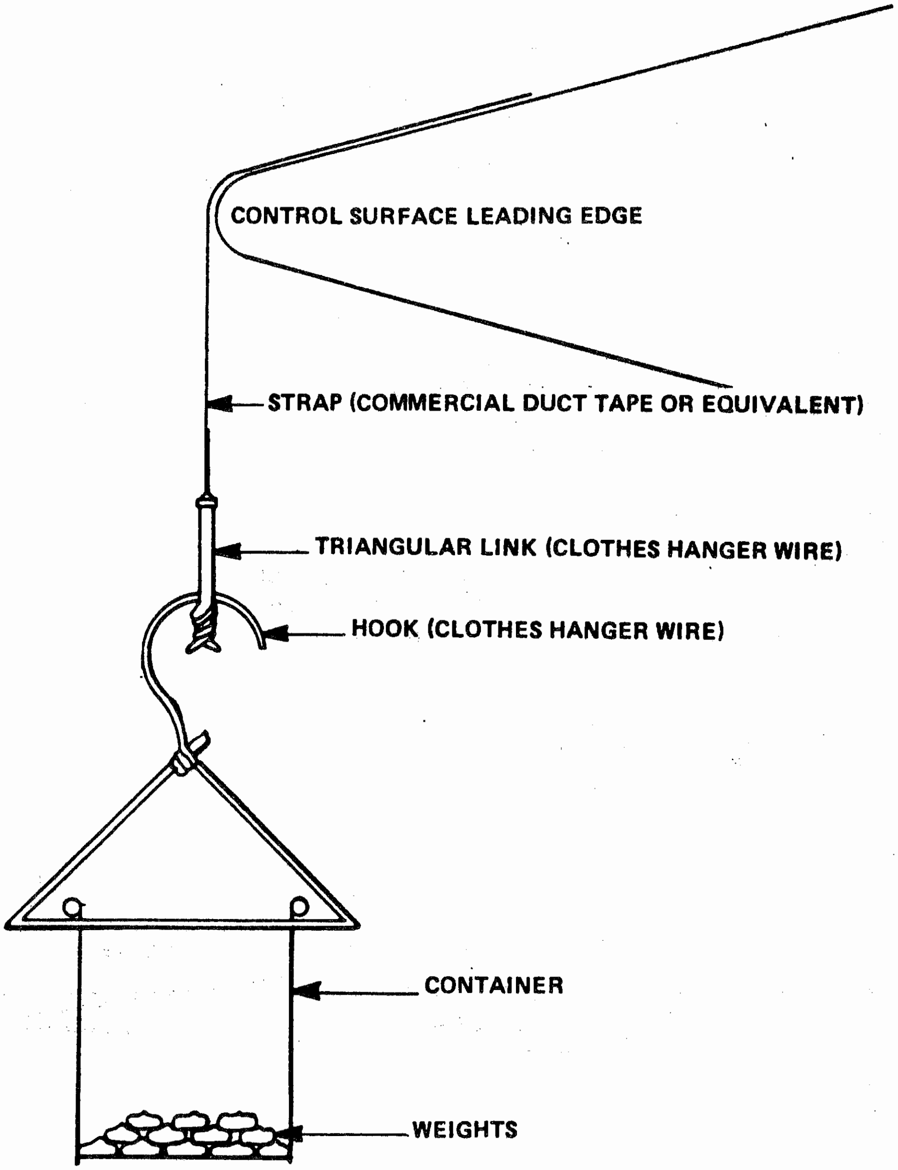
Lateral Control System Installation
Figure 201.



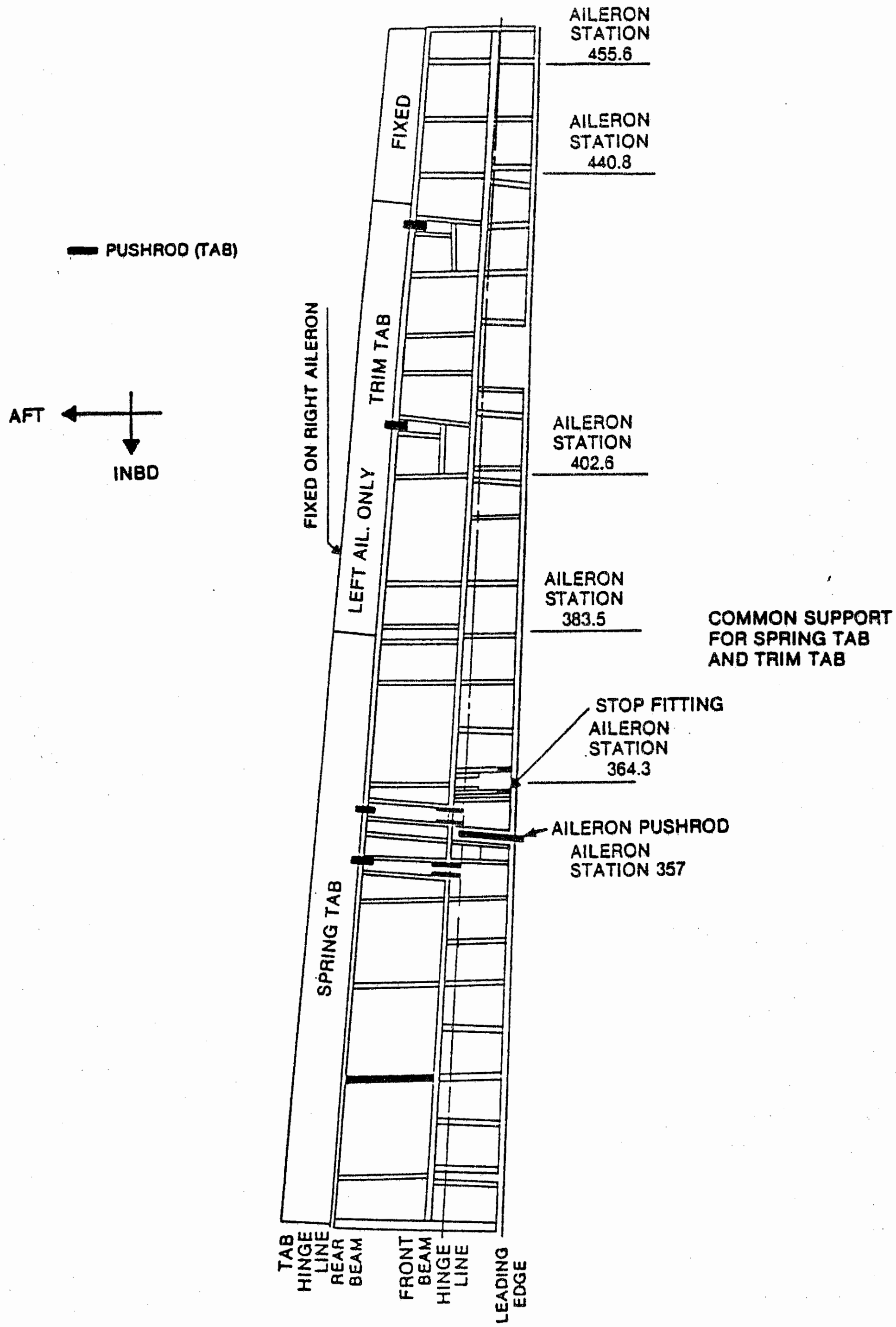
Aileron Hinge (Typical)
View Looking Inboard
Figure 202.



Aileron Mass Balance
Figure 203.



Suggested Method of Hanging Weights on Control Surface Leading Edge
Figure 204.



Aileron/Aileron Tabs
Figure 205.

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DIRECTIONAL CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

Directional control of the aircraft is provided by a single rudder. It is connected to the vertical stabilizer at 3 hinge points. The rudder is of metal construction, and connects to a torque tube. The torque tube connects to the system at a pushrod which is between Fuselage Station 678 and the base of the rudder. (See Figure 1) The control system is a mechanically operated system, incorporating cables and push-pull rods. The system is actuated by motion applied to either of the control pedal assemblies located in the cockpit enclosure. Operation of the system may also be initiated through an auto pilot servomechanism. A single tab having a dual function serves as a trim tab, a spring tab, or a combination of both. The spring tab is operative only between TAKEOFF to the full DOWN position of the flaps.

2. Rudder Pedals

(See Figure 201.

Aircraft is equipped with dual rudder pedals for pilot and copilot. The pedals are a combination rudder and brake pedal of conventional design. The pedals are mounted and supported by pedal hanger arms which are suspended and pivot on horizontal support tubes mounted spanwise in the left and right sides of the cockpit below the flight instrument panels (Fuselage Station 79.500). Just forward and centered between each set of rudder pedals is a vertical torque tube (Fuselage Station 68.500) which is universally connected to shaft and bearing supports in the cockpit floor. Each rudder pedal hanger is linked by pushrods to crank arms extending left and right (spanwise) at the top ot their respective torque tube. Induced motion of the rudder pedals causes rotation of the torque tube which is transmitted to attached cranks beneath the cockpit floor left and right (cranks extend rearward). These cranks provided a link up point for an interconnect pushrod which interconnects the left and right rudder pedal assemblies.

Output motion to the directional system is provided by a crank located beneath the cockpit floor on the right torque tube. A rotating (lug type) stop is located on the left torque tube beneath the cockpit floor. Individual simultaneous rudder pedal adjustment is provided both pilot and copilot by means of adjusting levers located between each set of pedals. The adjusting lever is linked to the top of the vertical torque tube and has a series of nine detents. The detents provide for nine positions which the pilot can adjust to. When the adjusting lever is raised, it permits the detented portion of the adjusting lever to clear the locking pawl and permits the pedals to be moved forward or backward by pushing or pulling on the adjusting handle. When the handle is released, the selected position is maintained by a latch system consisting of a lever roller and spring assembly.

3. Stop Bolts

Adjustable bolt type stops are located at the base of the left torque tube. The stops are fixed to structure just below the cockpit floor, at Fuselage Station 63. When maximum pedal travel is achieved in either direction, the rotating (lug type) stop contacts a fixed stop and limits rudder pedal travel. A similar set of adjustable bolt type stops are located at the base of the vertical fin structure and limit rudder travel.

4. Auto Pilot Connection

The auto pilot is connected by cables to a sector which is mounted on the axis of the skewed bellcrank. The sector is designed for the breaking strength of its cables, permitting the pilot to free the system by loading the cables to their breaking point, in the event the servomechanism should jam.

5. Rudder

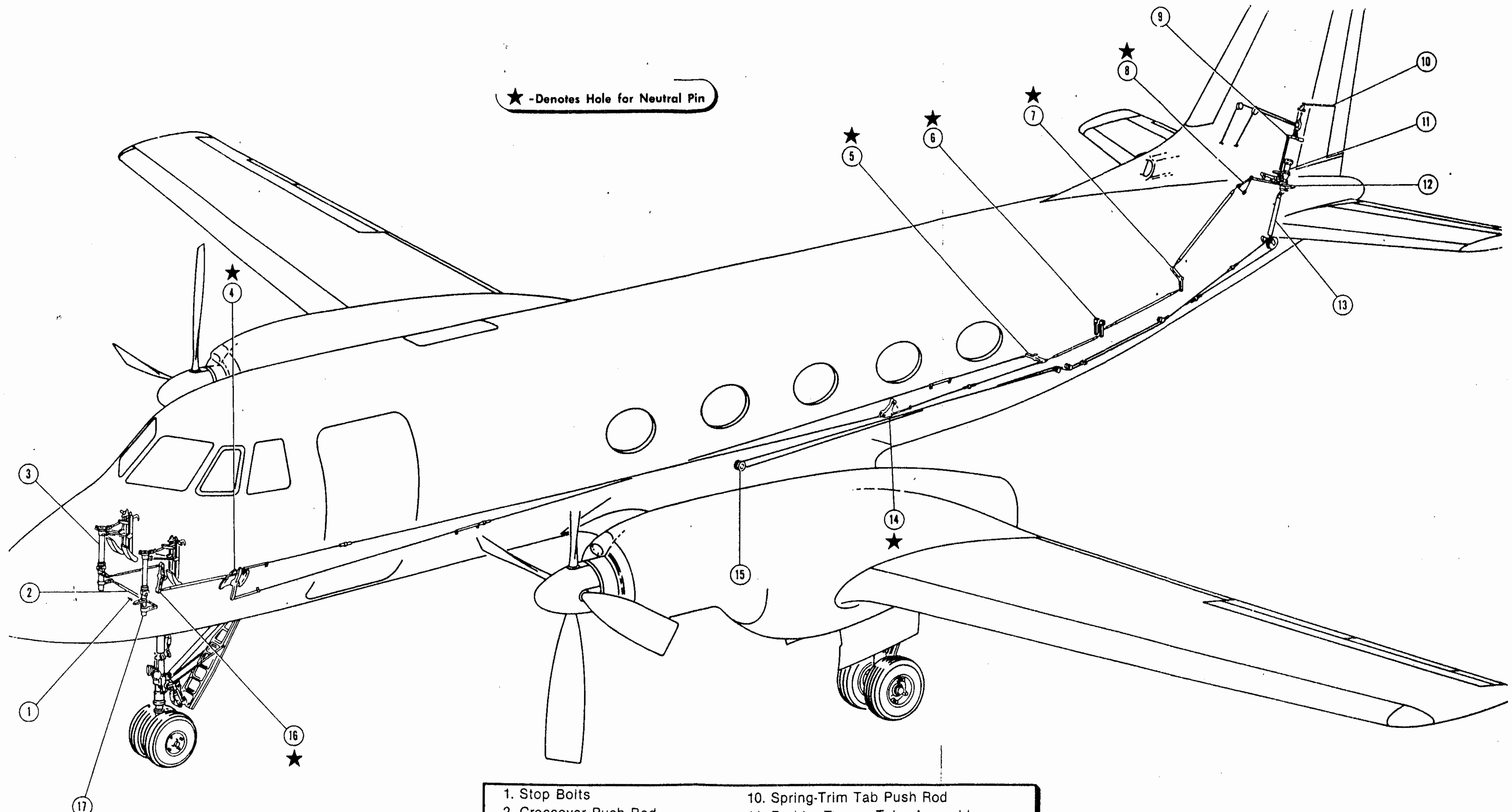
The rudder is a one piece assembly that consists of aluminum alloy beams, ribs, stringers and skins. (See Figure 202). Total weight of the rudder is 194 pounds. To minimize flutter and vibration it is dynamically balanced by weights placed in the cap forward of the hinge line. Hinges consist of steel fittings and self-aligning bearings. A tab constructed of aluminum alloy beams, ribs, stringers and skin extends from the base to waterline 243.500 and performs the functions of both a directional tab and a spring tab. By means of cranks, spring bungees, a summing link, and an actuator, rudder pedal and the trim control wheel motions are applied to the tab. The crank, actuator, and summing links are mounted on the rudder leading edge with the spring bungees and additional cranks mounted on an assembly at the base of the rudder torque tube.

6. **Spring Tab**

NOTE: The tab surface on the rudder can be employed either as a spring tab, a trim tab, or a combination of both, depending upon the control characteristics desired. Operation of the spring tab is dependent primarily upon the position of the flaps. With the flaps retracted, the spring tab is locked out and inoperative; and with the flaps deflected to the takeoff position, or any position beyond takeoff, the tab is unlocked and operative.

The spring tab stop mechanism consists of a roller mounted between a slotted surface in the rudder torque tube crank. (See Figure 203). As the rudder crank rotates, due to pilot effort, and the roller travels between the sides of the slot, relative motion between the rudder and tab surfaces takes place. (Assume that the air load on the rudder, resisting movement of the rudder, is greater than the input force applied by the pilot to the springs.) After the sides of the slot strike the roller, further pilot effort to the rudder torque tube causes the entire rudder surface to rotate as a single unit. The slotted surface is machined so as to limit the spring tab motion to $\pm$ 10°. If the pilot input effort is greater than the opposing air load on the rudder, the input motion is then directly transmitted to the rudder resulting in rudder travel.

Lockout of the spring tab is accomplished by a cable actuated eccentric cam located beneath the rudder torque tube. Motion used to operate the cam is taken from the sheave at the central gearbox in the flap drive system. The cam is interconnected, through a series of links and a bellcrank, to the spring tab stop cam (slot) which rotates against the roller. When the eccentric cam is rotated, due to retraction of the flaps, the stop cam slot is locked against the roller and prevents any pilot input motion from being transmitted to the tab. Incorporated in the tab lockout mechanism is a spring bungee which is designed to absorb full motion of the flap operated cables if the locking mechanism jams. The lockout cables are 3/32 inch and are tensioned to 45 $\pm$ 5 pounds. The aft cable pulley and the eccentric cam are designed to absorb the full travel of the lockout cables. However, lockout of the tab is realized utilizing only that portion of the travel encountered in reaching the takeoff position of the flaps.



1. Stop Bolts	10. Spring-Trim Tab Push Rod
2. Crossover Push Rod	11. Rudder Torque Tube Assembly
3. Right Torque Tube	12. Sta 693 Sector
4. Sta 126 Bellcrank	13. Spring-Trim Tab Lockout Bungee
5. Sta 544 Bellcrank	14. Sta 548 Idler Crank
6. Sta 582 Bellcrank	15. Flap Hydraulic Motor Lockout Sheave
7. Sta 640 Bellcrank	16. Sta 93 Bellcrank
8. Sta 678 Bellcrank	17. Left Torque Tube
9. Summing Link	

Directional Control System
Figure 1.

DIRECTIONAL CONTROL SYSTEM — MAINTENANCE PRACTICES

1. Directional Control — Rigging

A. Special Tools Required

ITEM	DESCRIPTION
A. Tensiometer	Pacific Scientific 30 lbs to 300 lbs.
B. Rigging Pins	1/4 inch diameter bolts, AN4 or equivalent
C. Throwboards	159GT1039
D. Tension Scale	0 - 50 lbs. Scale

B. Rigging

(See Figure 201)

NOTE: Part numbers are used in this manual only as a means of identification when nomenclature alone is inadequate. This manual must not be used for identifying spare parts by number. Consult the Gulfstream I Illustrated Parts Catalog for this information.

(1) Pin rudder pedals in neutral by means of rigging hole in crank arm of lower end of hanger assembly.

CAUTION: EXTREME CARE SHOULD BE TAKEN WHEN INSTALLING LOCKCLAD ASSEMBLIES. DO NOT ARC CABLES ON LESS THAN A 60-INCH RADIUS.

(2) Install lockclad cable assemblies 159C10051-1, 159C10052-1 and 159C10053-1 through holes provided in Fuselage Station 85 and 119 and pass cables through their respective positions. Install cables through nose wheel well.

(3) Install bellcrank assembly 159CM10313-1 at Fuselage Station 93.062 and pin in neutral position.

(4) Install crank assembly 159C10304-1 at Fuselage Station 126 and pin in neutral position.

(5) Adjust and install pushrod assembly 159C10057-1.

(6) Install idler assembly 159CM10015-1 at Fuselage Station 458.562 and pin in neutral position.

(7) Install crank assembly 159GM10014-1 at Fuselage Station 544.500 and pin in neutral position.

(8) Install cable assemblies 159C10051-1, 159C10052-1 and 159C10053-1 between Fuselage Station 126 and 544.500. Set cable tensions 200 pounds ± 10% (at room temperature, 70°F) and safety turnbuckles.

NOTE: For each 5° above 70°F add 10 pounds to cable tension. For each 5° below 70°F subtract 10 pounds from cable tension.

(9) Install bellcrank assembly 159C10071-1 at Fuselage Station 582.875 and pin in neutral position.

(10) Adjust and install pushrod assembly 159C10050-1.

(11) Install crank assembly 159CM10332-1 at Fuselage Station 640.875 and pin in neutral position.

(12) Adjust and install pushrod assembly 159C10055-1.

(13) Install crank assembly 159CM10323-1 at Fuselage Station 678.74 and pin in neutral position.

(14) Adjust and install pushrod assembly 159C10059-1.

(15) Install tube assembly 159C10011-1 at base of rudder.

(16) Using rudder throwboard 159GT1039, set rudder at 0° trail position and adjust and install pushrod assembly 159C10056-9.

(17) Adjust and install pushrod assembly 159C10058-3 between crank assemblies at Fuselage Station 93.062 and 126.

(18) Remove all rigging pins and operate system. Left pedal forward should give left rudder. Right pedal forward should give right rudder. Set rudder surface stop for 22° + 0, − 1° left and right and safety.

(19) Set rudder pedal stops (under cockpit floor) by deflecting the rudder and the rudder spring tab to their full throws, then set the stops to contact the crank arm at these position.

(20) At Fuselage Station 126, install and tension autopilot cables to 45 ± 5 pounds and safety cable adjusters.

NOTE: For each 10° above 70°F add 1.5 pounds to cable tension. For each 10° below 70°F subtract 1.5 pounds from cable tension.

(21) To rig rudder spring tab, attach cable assemblies 159C10307-1 to sector assembly 159C10417-1 and install and pin sector assembly at Fuselage Station 693.687.

NOTE: Ensure link assemblies, idler assembly 159CM10419 and pushrod assembly 159C10424 are installed before performing Step 22 below.

(22) Install and adjust pushrod assembly 159G10423-1 between crank 159CM10418-1 (on tube assembly 159C10011-1) and two link assemblies 159CM10420-1.

(23) Install and adjust spring rod assembly 159C10321-1 between sector assembly 159C10417-1 and bellcrank assembly 159CM10425-1 on the lower end of tube assembly 159C10011-1 for maximum travel of cam roller 159CM10414-1 on tube assembly.

(24) Attach forward end of two cable assemblies 159C10307 to flap drive, central gearbox assembly when flaps are in fully extended emergency down position and tension cables to 45 pounds ± 10%.

NOTE: For each 10° above 70°F add 1.5 pounds to cable tension. For each 10° below 70°F subtract 1.5 pounds from cable tension.

(25) Lock rudder with gust lock and check rudder spring tab throws. Spring tab throws shall be 10° left and right ± 1°.

(26) Check system with flaps up for lockout of spring tab. Tab may have ± 1° of motion when locked out.

2. Rudder Control Pedal — Lubrication

NOTE: Clean all fittings prior to lubrication and wipe fittings and area clean of excess lubricant upon completion.

A. Lubricate rudder pedals. (See Chapter 12)

3. Rudder Pedals — Adjustment

The left torque tube is limited, by adjustable stop bolts at Fuselage Station 63, to permit a pedal travel of approximately 4 inches. Adjust torque tube after rudder surface stops are positioned and cable tension adjustments have been completed. Adjust stop bolts to provide a clearance between stop bolts and torque tube lugs which will permit 10° of spring tab travel after rudder contacts stops.

4. Rudder Hinge Side Play - Inspection

Using a feeler gage, determine lateral play at all rudder hinge attach points. See Figure 203. Maximum allowable hinge lateral play is 0.015 inch.

5. **Rudder Tab — Mass Balance**

Since the rudder and the rudder tab are both mass balanced, the rudder tab should be checked before the rudder is checked. A tab with only the prime coat on it should be 20 to 25 inch pounds nose heavy. Balance weights should, therefore, be applied to the trailing edge. A fully painted tab should be 17 to 22 inch pounds. The tab should be mounted with the actuator lugs (on the second hinge from the bottom) pointing downward. There are three intermediate hinges that require the weight of the hinge fitting supported. Since the tab is of symmetrical section, the horizontal position may be judged by eye.

If the whole rudder is to be balanced, it is recommended that the tab be weighed before installing it on the rudder. The tab should weigh 21.0 pounds painted or 19.4 pounds primed.

6. **Rudder — Mass Balance**

The rudder mass balance is affected considerably by any weight changes in the trim tab. When mass balancing the rudder, the rudder must include the tab with all its attaching hardware and mechanisms in place. Nuts are not required to be tightened. Lights, bulbs, static wicks and all access panels must be in place with attaching hardware.

The rudder should be mounted with the tab horn and pushrod down and the tab at zero angle. Since the rudder is part of a symmetric airfoil section, the chordline horizontal may be judged by eye. It is permissible to disconnect the rudder at the torque tube base (See Figure 202, Item 6). All attaching bolts should be either in the rudder or left in the torque tube base.

The correction for removing the rudder, rudder spring tab and stop tube assembly is 26.9 inch pounds, tail heavy.

In the primed condition with the rudder, rudder spring tab and stop tube assembly in place, the acceptable mass balance is 240 to 280 inch pounds nose heavy for Aircraft 1 and 2, and 250 to 290 inch pounds nose heavy for Aircraft 3 - 200, 322 and 323.

In the painted condition with the rudder, rudder spring tab and stop tube assembly in place, the acceptable mass balance is 195 to 235 inch pounds nose heavy for Aircraft 1 and 2, and 205 to 245 inch pounds nose heavy for Aircraft 3 - 200, 322 and 323.

7. **Rudder — Removal / Installation**

A. Removal

(1) Remove the following access covers;

(a) Vertical Stabilizer - 109-RH-1, 109-TE RH-1 thru 109-TE RH-5

(b) Rudder - 110-RH-1 thru 110-RH-5

(2) Gain access to tail cone area.

NOTE: Aircraft 1 - 7 have fixed tail cones, access is through opening be low stabilizer, right side.

Aircraft 8 - 200, 322 and 323, have a removable tail cone.

(3) Remove upper spring/trim tab attaching bolts and install hoist sling to these points, remove lower bolt and install guide cables. (See Figure 204)

(4) Separate seal strips from stabilizer. (See Figure 205)

(5) Slacken cable tension and install two non-metallic clamps (to prevent slack forward of Fuselage Station 618) on trim tab cables at Fuselage Station 618 in tail compartment.

(6) Disconnect cables aft of Fuselage Station 618; remove tab actuator, then remove cables from tab actuator and secure cables to vertical stabilizer. (See Figure 205).

(7) Remove vertical (tab) pushrod between rudder and torque tube linkage. (See Figure 205)

NOTE: Before rod can be removed, parts of rudder sector may have to be removed. (See Figure 205)

(8) Disconnect navigation light at connector on rudder leading edge. (See Figure 204)

(9) Remove rubber plugs from vertical stabilizer.

(10) Take up slack on hoist sling and remove bolts from rudder hinges and torque tube. (See Figure 204)

(11) Remove shim from between base of rudder and torque tube. (See Figure 205)

(12) Remove rudder.

B. Installation

(1) Position rudder on vertical stabilizer and install hinge bolts.

(2) Set shim on torque tube (combined shim thickness not to exceed 0.090 inch), line up holes, position rudder, install bolts and safety-wire.

(3) Remove hoist sling and guide cables.

(4) Install spring/trim tab attaching bolts.

(5) Install vertical (tab) pushrod between rudder base and torque tube linkage.

(6) Connect navigation light.

(7) Install cables on tab actuator, install actuator and connect to linkage.

(8) Attach cables aft of Fuselage Station 618, take up slack and remove cable clamps.

NOTE: Comply with Step (9) below on rudder replacement only.

(9) Install rig pin in linkage neutral hole and adjust rods so that tab is aligned with fixed trailing edge of rudder.

(10) Install rudder leading edge seals to fin.

(11) Tension trim cables to 45 $\pm$ 5 pounds at 70°F.

NOTE: For each 10° above 70°F add 1.5 pounds tension; for each 10° below 70°F subtract 1.5 pounds.

(12) Using throwboard, check trim tab throws:

- Rudder trim wheel full right - tab left 10° $\pm$ 1°.
- Rudder trim wheel full left - tab right 10° $\pm$ 1°.

(13) With wing flaps up, check spring/trim tab for lockout; a motion of $\pm$ 1° is permissible.

(14) With wing flaps below takeoff position and gust lock engaged, check spring tab throws using throwboard:

- Rudder full right - tab left 10° $\pm$ 1°.
- Rudder full left - tab right 10° $\pm$ 1°.

(15) Unlock gust lock and check rudder throws:

- Rudder full left - 22° + 0 − 1°.
- Rudder full right - 22° + 0 − 1°.

(16) Energize essential dc bus and check operation of navigation light.

(17) Install rubber plugs.

(18) Close or install all access covers; de-energize essential dc bus if no longer required,

8. Rudder Bearing / Fitting — Inspection

A. Remove rudder. (See Rudder — Removal / Installation, this section)

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP SOLVENT AWAY FROM OPEN FLAMES.

B. Thoroughly clean areas to be inspected with solvent or equivalent.

C. Carefully inspect control surface fittings and mating fitting for evidence of corrosion, wear, or chafing; check for signs of cracks, especially at areas adjacent to bearing and bolt attachment points.

D. Carefully check fittings for irregular surfaces under primer and for evidence of granular corrosion.

E. Inspect bearings for security of mounting, evidence of rust, and for roughness when rotated under load.

F. Replace worn, chafed, corroded, deeply scratched, or loose fittings.

G. Replace loose, rusted, or rough bearings.

NOTE: While rudder is removed, open rear vertical beam access panels and inspect interior structure of vertical fin for cracks, corrosion, cleanliness and general condition. Replace rear vertical beam access panels.

H. Install rudder. (See Rudder — Removal / Installation, this section)

9. Rudder Pedals — Removal / Installation

(See Figure 206)

A. Removal

(1) Disconnect brake rod at pedal crank.

(2) Disconnect the pedal end of pushrod connecting rudder pedals to torque tube crank.

(3) Remove upper bolt holding pedal hanger to support tube.

(4) Disconnect rudder pedal adjustment shaft at torque tube. Hold end against spring force when removing bolt.

(5) Remove springs.

(6) Remove pivot bolt from adjustment lock crank. Remove crank and adjustment shaft.

(7) Remove bolt attaching torque tube to universal.

B. Installation

(1) Install torque tube attaching bolts at universal.

(2) Place rudder pedal adjusting shaft in position. Install locking crank and pivot bolts.

(3) Attach springs to locking crank and support tube.

(4) Attach adjustment shaft to torque tube.

(5) Install rudder pedal hangers.

(6) Pin crank arms in neutral, set adjustment in fifth notch, adjust and install pushrods between hangers and torque tube so that pedal pivot points are in line with Fuselage Station 79.5.

(7) Install brake rod linkage.

(8) Check operation of rudder pedals throughout full range of travel and adjustment.

10. Rudder Control System — Operational Test

WARNING: ENSURE PERSONNEL AND EQUIPMENT ARE CLEAR OF ALL CONTROL SURFACES BEFORE OPERATION.

CAUTION: IF BINDING, JAMMING OR SCRAPING OCCUR DURING FOLLOWING PROCEDURES, STOP OPERATION IMMEDIATELY AND INVESTIGATE.

A. Operate rudder in both directions until surface stops are contacted.

B. Operate rudder trim wheel in both directions until surface stops are contacted as follows:

- Rudder trim wheel left - tab right 10° ± 1°.
- Rudder trim wheel right - tab left 10° ± 1°.

C. Ensure cockpit indicator corresponds to tab positions.

D. Return tab to 0° on indicator.

E. Lower wing flaps and engage gust lock.

F. Operate rudder pedals in both directions to check spring tab throws against stops.

NOTE: Rudder will not move but spring tab will go to full travel.

G. Raise flaps; with operation of rudder pedals, tab should remain stationary (locked out), however tab may have ± 1° of movement.

11. Rudder / Rudder Trim Tab — 20,000 Hr Inspection

(See Figure 207)

A. Rudder

(1) Remove rudder. (See Rudder — Removal / Installation, this section)

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP SOLVENT AWAY FROM OPEN FLAMES.

(2) Thoroughly clean areas to be inspected with solvent or equivalent.

(3) Carefully inspect control surface fittings and mating fitting for evidence of corrosion, wear, or chafing; check for signs of cracks, especially at areas adjacent to bearing and bolt attachment points.

(4) Carefully check fittings for irregular surfaces under primer and for evidence of granular corrosion.

(5) Inspect bearings for security of mounting, evidence of rust, and for roughness when rotated under load.

(6) Replace worn, chafed, corroded, deeply scratched, or loose fittings.

(7)

(8) Replace loose, rusted, or rough bearings.

(9) Front and rear beam caps. Rudder station 78.5.

(10) Rib caps at front spar. Rudder station 110.92.

(11) Skin stiffeners between Rudder station 137.75.

NOTE: While rudder is removed, open rear vertical beam access panels and inspect interior structure of vertical fin for cracks, corrosion, cleanliness and general condition.

(12) Torque tube stop fitting, attachment and supporting structure.

(13) Area where torque tube connects to the thrust shear web.

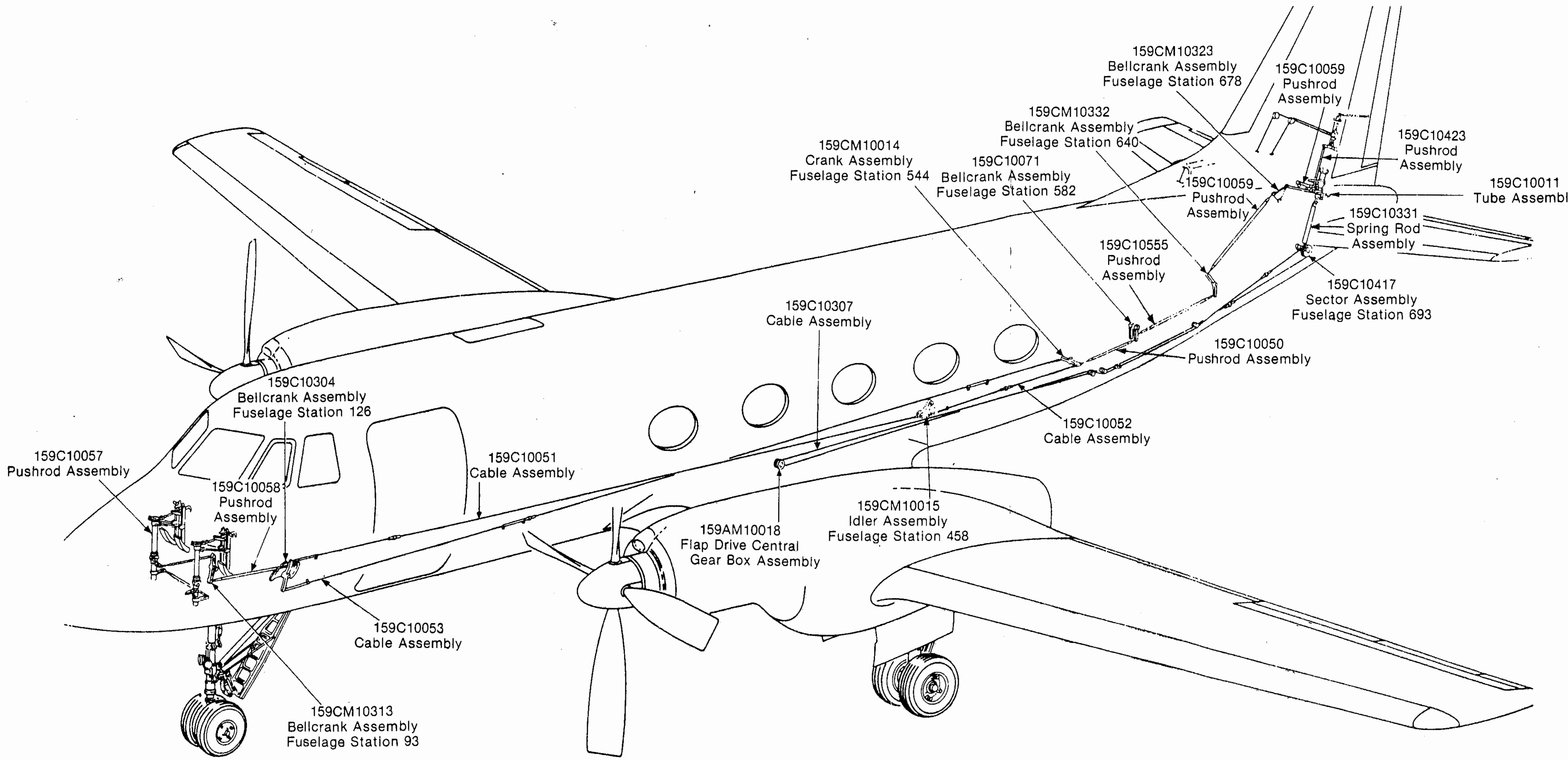
(14) Replace rear vertical beam access panels.

(15) Install rudder. (See Rudder — Removal / Installation, this section)

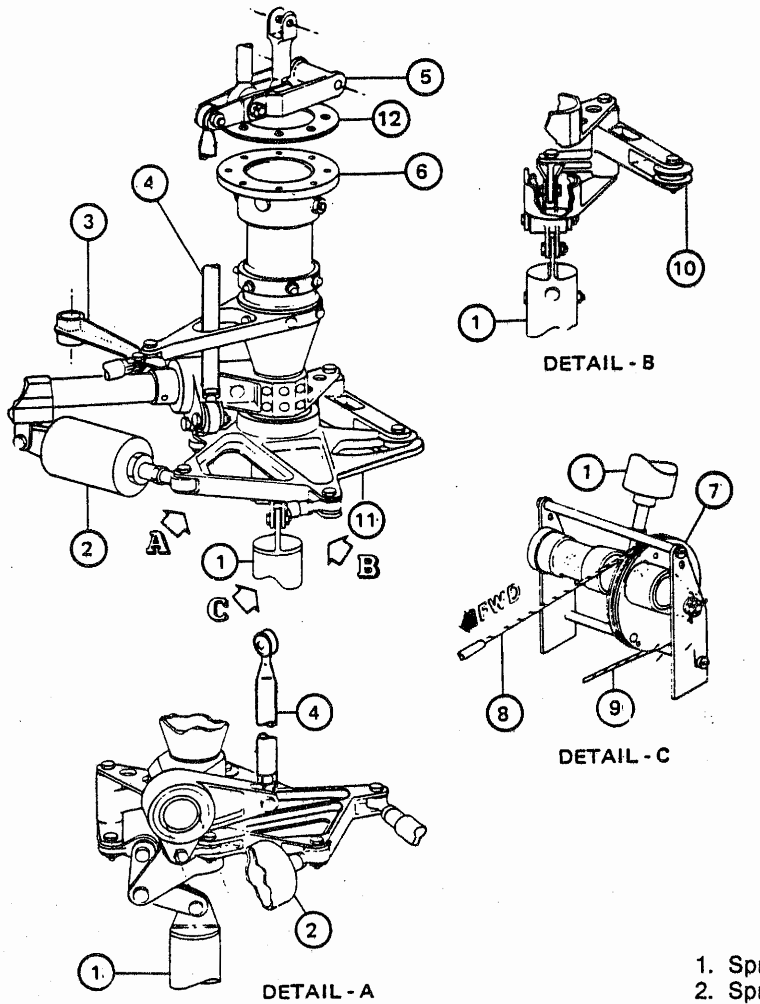
B. Rudder Trim Tab

(1) Trim tab front beam channel at Rudder station 101.29.

(2) Trim tab pushrod horn fitting, attachments and supporting structure.

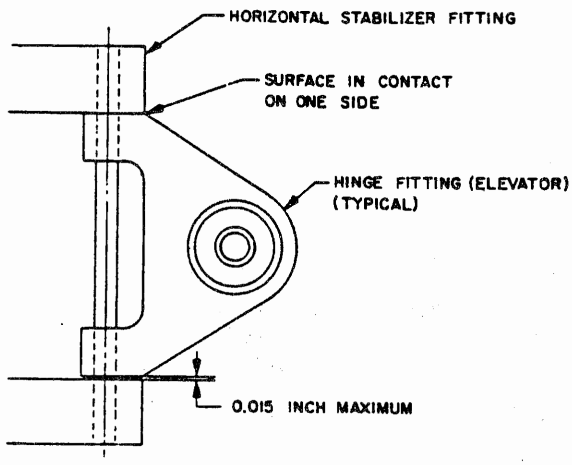


Directional Control System Installation
Figure 201.

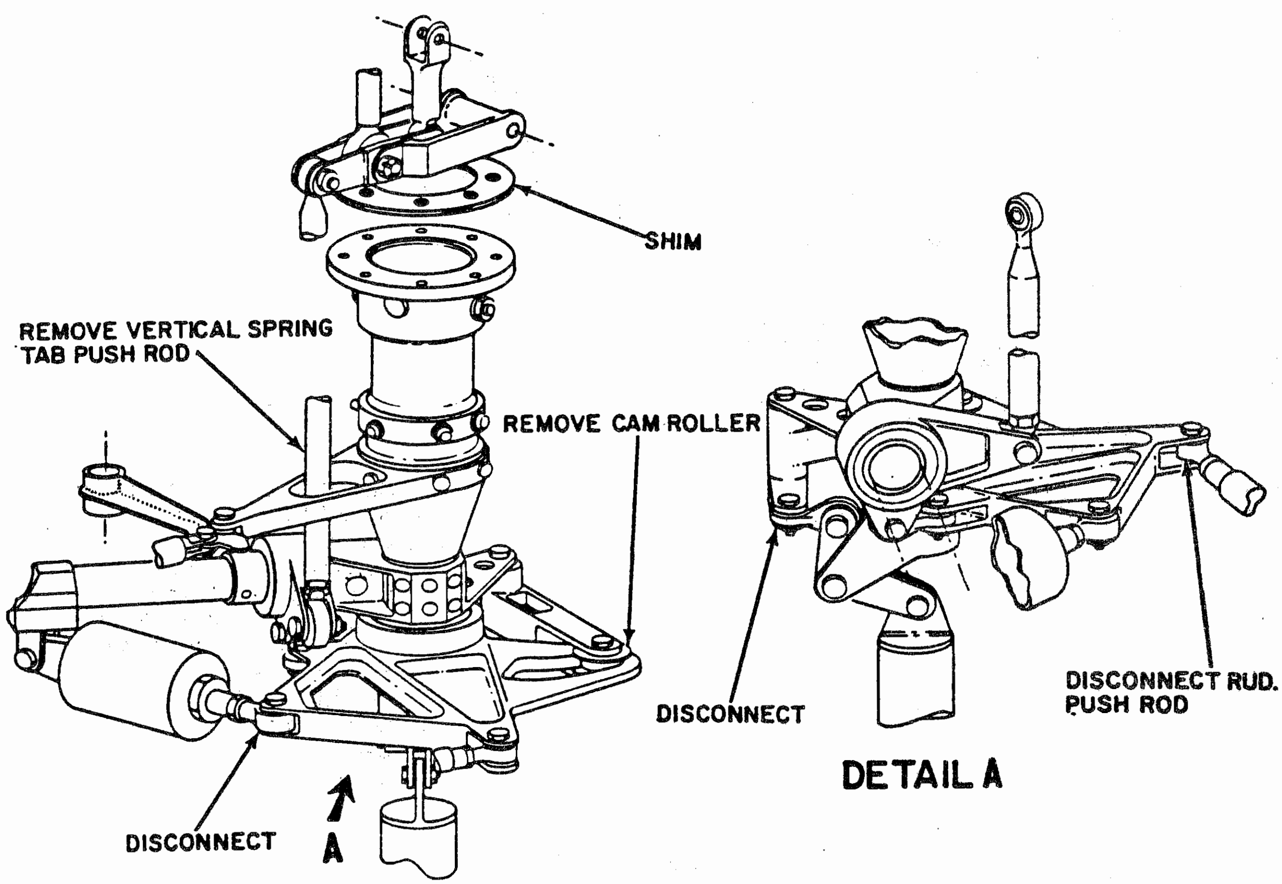


1. Spring Tab Lockout Bungee
2. Spring Tab Bungee
3. Gust Lock Latch
4. Spring Tab Push Rod
5. Summing Link Assembly
6. Rudder Torque Tube Base
7. Spring Tab Lockout Drum
8. Flaps Up Cable
9. Flaps Down Cable
10. Cam Roller
11. Cam
12. Shim

Rudder Sector
Figure 202.

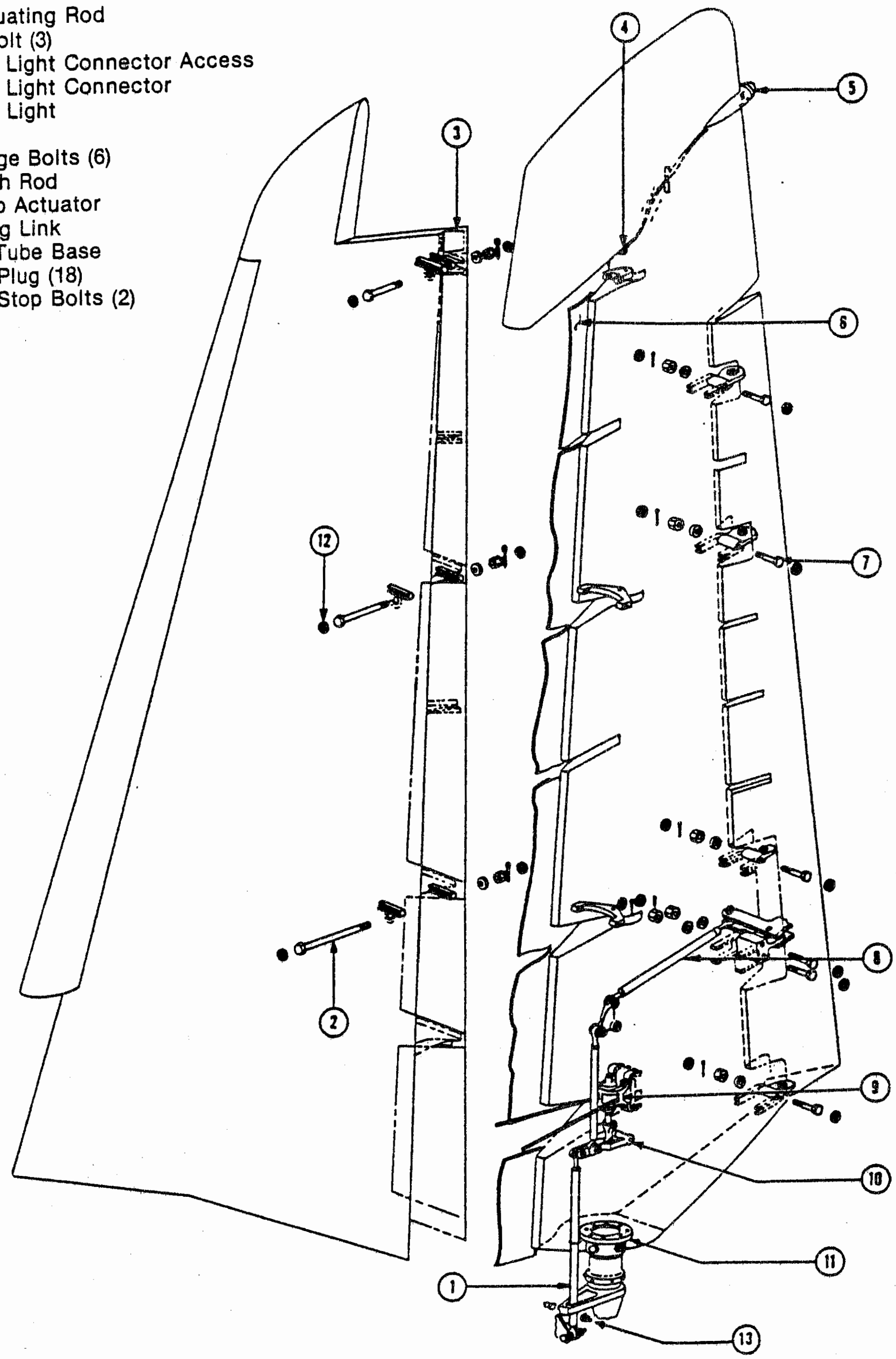


Rudder Hinge
Typical View Looking Down
Figure 203.

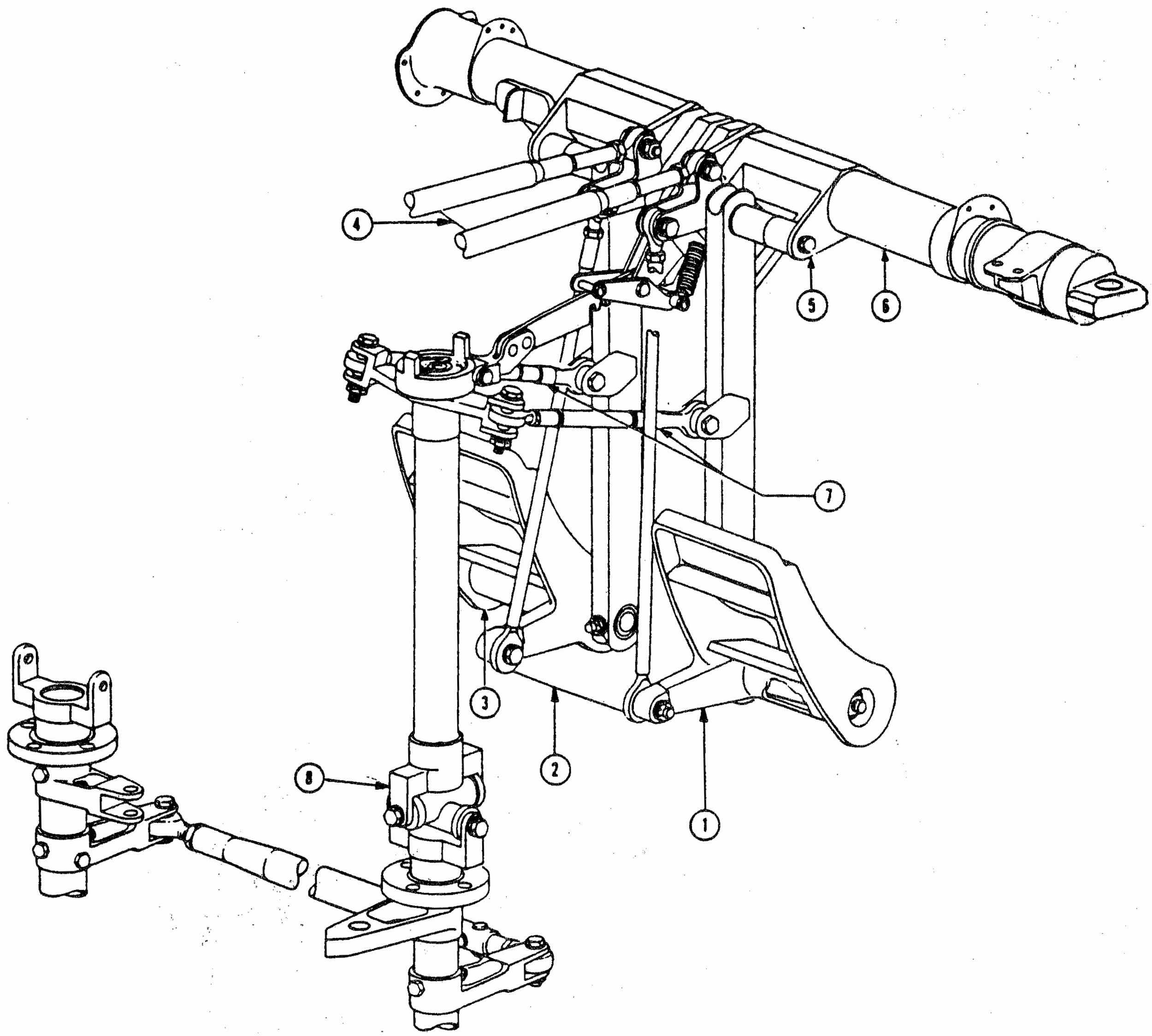


Rudder Removal / Installation
Figure 204.

1. Tab Actuating Rod
2. Hinge Bolt (3)
3. Position Light Connector Access
4. Position Light Connector
5. Position Light
6. Seal
7. Tab Hinge Bolts (6)
8. Tab Push Rod
9. Trim Tab Actuator
10. Summing Link
11. Torque Tube Base
12. Rubber Plug (18)
13. Rudder Stop Bolts (2)

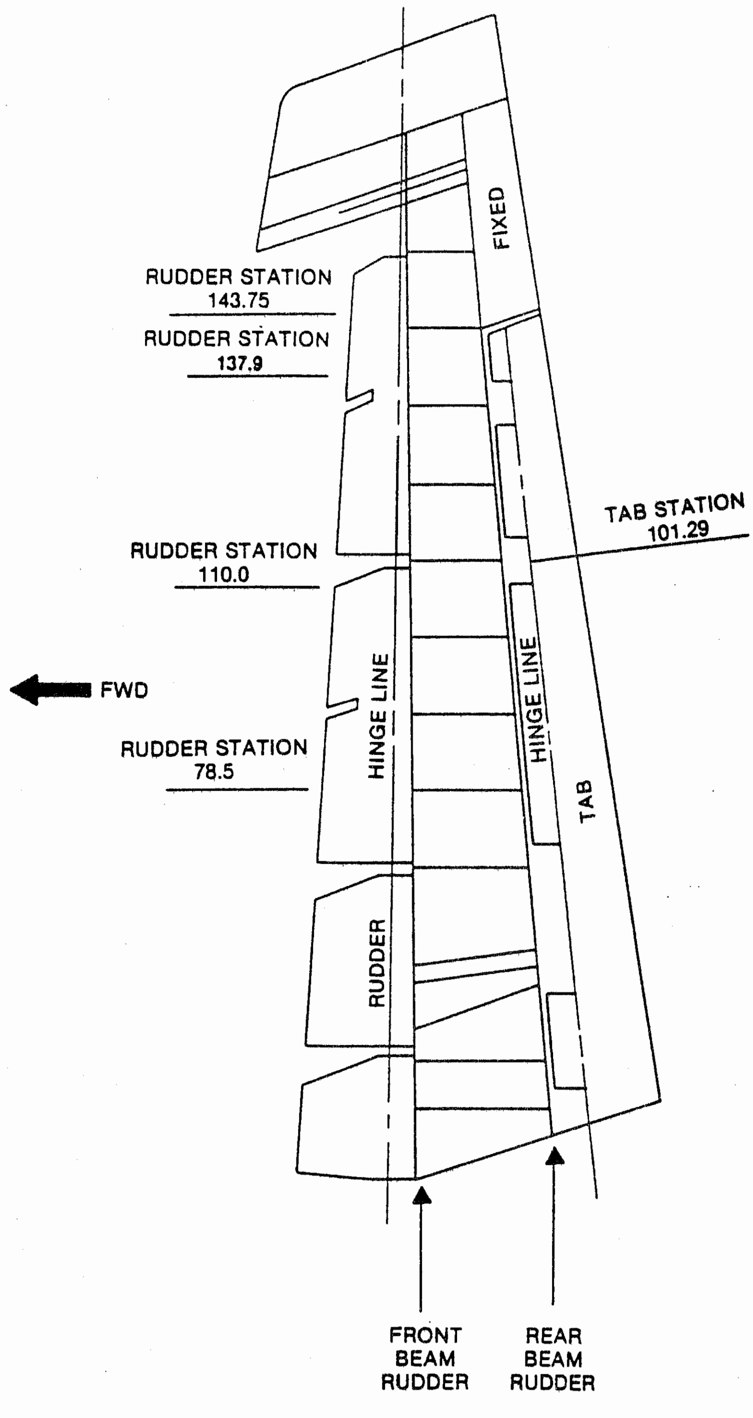


Rudder Removal / Installation
Figure 205.



1. Pilot's Left Rudder Pedal
2. Brake Rod
3. Pilot's Right Rudder Pedal
4. Brake Push Rod
5. Mounting Bolt
6. Rudder Pedal Hangar
7. Torque Tube Push Rod
8. Universal

Rudder Pedal Removal / Installation
Figure 206.



Rudder/Rudder Trim Tab
Figure 207.

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TRIM SYSTEM – DESCRIPTION / OPERATION

1. General

Trim control is accomplished by conventional trim tabs mounted on the control surfaces. The lateral trim tab is located on the left aileron, the longitudinal on both elevators, and the directional on the rudder. (See Figure 1). The rudder tab also serves as a spring tab in the range between takeoff position and full down position of the flaps. Screwjacks at each trim tab surface are driven by cables operating from 2 1/2 inch interchangeable drums below the floor at the center console. Screwjack operation is controlled by manually operated control wheels mounted on the pedestal in the cockpit. The lateral (aileron) tab control and the directional (rudder) tab control have single wheels mounted at the aft end of the pedestal and accessible to either the pilot or copilot. (See Figure 201) The longitudinal (elevator) trim tab is controlled by dual wheels interconnected by a torque shaft mounted at the forward end of the pedestal. One wheel accessible to the pilot, the other accessible to the copilot. The degree and direction of tab travel is indicated at each control wheel. Edge lights have been provided for illumination of the tab controls. Cables lead from the drums below the floor at the center console to a drum driving the acme screwjack at each tab location. Each acme screwjack has a travel of 1.08 inches and provides 1500 pounds of thrust. Stops are not provided at the screwjacks but are installed at the trim control wheels and limit the travel of the rudder and aileron control wheels to 6 5/8 turns which corresponds to a travel of $\pm$ 10° for the rudder tab and $\pm$ 20° for the aileron tab. The elevator control wheel travel is limited to 3 3/4 turns which corresponds to a travel of 20° (nose down) and 3° (nose up) for the elevator. At each control wheel, an AN456-AD4 rivet is used as a shear pin connecting the wheel to the torsion bar. The AN456-AD4 rivets shear at the ultimate screwjack load of 1500 pounds. (This load is adequate to cover calculated air loads on the tabs.) In the longitudinal system, a force of 31 pounds on the longitudinal control wheels causes the pin to shear. On the directional and lateral control wheels, 55 pounds shears the pin. The integral mechanical stops at the trim control wheels preclude inadvertently shearing the pins on the ground by applying excessive force to the trim wheels after reaching full travel of the tabs. Cables used are 3/32 inch 7 x 7 and are tensioned to 45 pounds $\pm$ 10% at 70°F. The cables are fairleaded at least every 30 inches (except the rudder trim cables in the tail turnbuckle which travel 51 inches) and unlike the primary control cables, are not interchangeable between systems. To ensure cabin pressurizations, rubber seals are installed at each point where the cables extend through the pressurized area of the fuselage.

2. Rudder Trim System

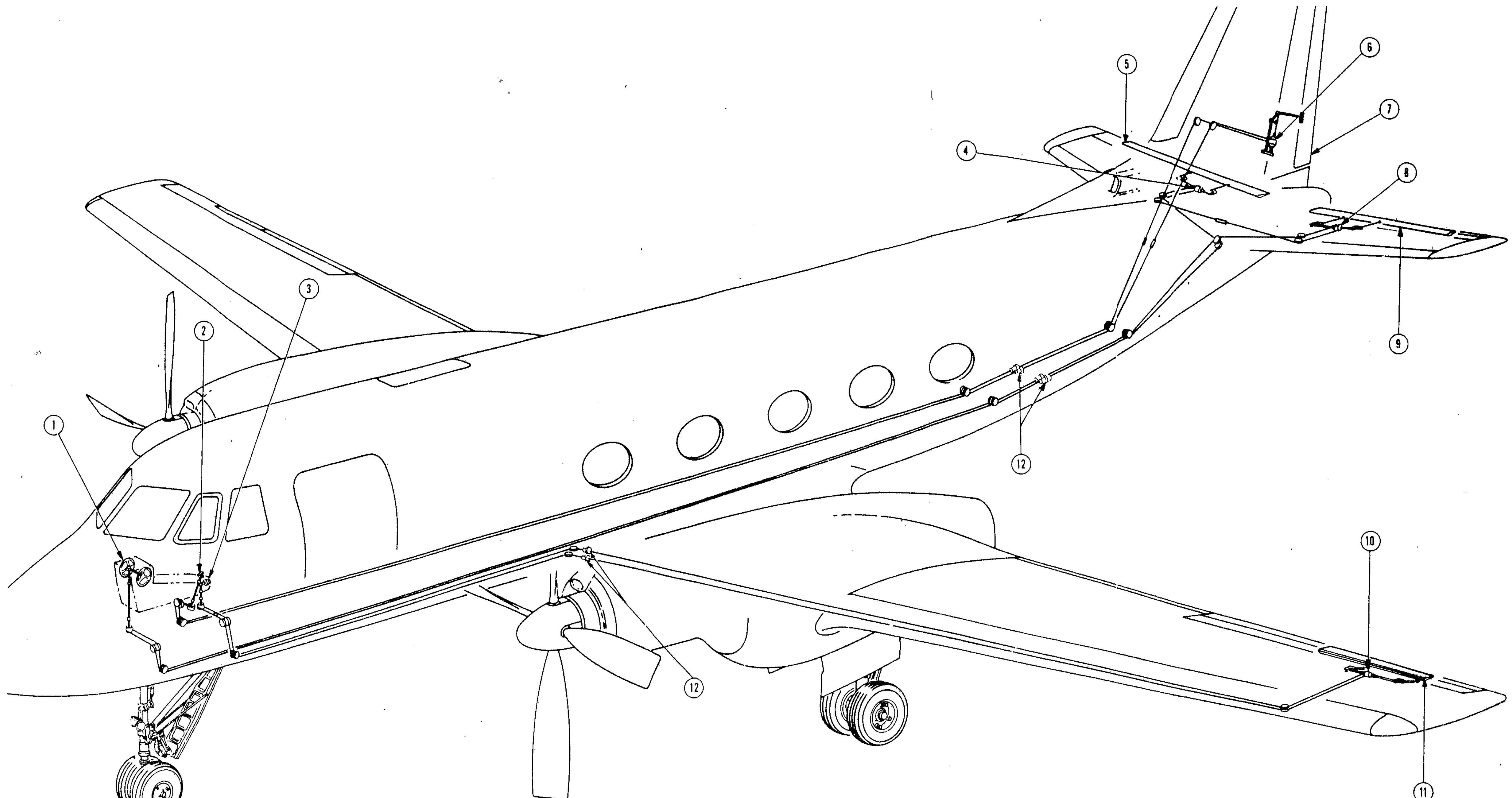
From the drum below the floor of the flight station pedestal, cables lead to the rudder trim actuator (acme screwjack) in the rudder leading edge. The trim cables wrap around the cable drum a total of nine complete turns, and are fastened with ball locks at both ends of the drum. Rotation of the drum about the acme screw produces a linear shaft travel of 1.08 inches, and produces an ultimate axial load of 1500 pounds either under tension or compression, and since no stops are in the actuators, this shaft travel is determined by the stops at the trim control wheels. The actuators are irreversible up to and including 250 cycles per second, and contain provisions which prevent the acme screw torque from being transmitted to the rod ends. The vertical movement of the actuator is applied to a summing link just below it. The summing link is connected to the rudder pedals. A spring bungee in the rudder torque tube base holds the spring tab in relation to the rudder. Movement of the actuator by the trim control wheel displaces the spring-trim tab to a position on one side or the other in which it is held by the spring bungee. This displacement places a trim correction on the rudder.

3. Aileron Trim Tab

The aileron trim tab is on the left aileron and is mounted by three hinges. Two rods connect it to the actuator located at the aileron leading edge. The cables from the actuator drum run forward, then down the front spar of the wing to the center of the fuselage. They then run forward to a drum under the cockpit floor at the center pedestal.

4. Elevator Trim Tab

Both elevators have trim tabs extending from stabilizer station 18 5/16 to stabilizer station 118. These trim tabs are operated by actuators located in the elevator. Cables lead to each actuator from a drum below the cockpit floor at the center pedestal. The elevator trim tabs have a range of movement of 3° up and 20° down.



1. Longitudinal Trim control Wheels
2. Lateral Trim Control Wheel
3. Directional Trim Control Wheel
4. Right Elevator Trim Tab Actuator
5. Right Elevator Trim Tab
6. Rudder Spring - Trim Tab Actuator
7. Rudder Trim Tab
8. Left Elevator Trim Tab Actuator
9. Left Elevator Trim Tab
10. Left Aileron Trim Tab Actuator
11. Left Aileron Trim Tab
12. Pressure Seals

Trim System
Figure 1.

TRIM SYSTEM — MAINTENANCE PRACTICES

1. Rudder Spring Trim Tab — Rigging

(See Figure 201)

NOTE: Part numbers are used in this manual only as a means of identification when nomenclature alone is inadequate. This manual must not be used for identifying parts by number. Consult Gulfstream I Illustrated Parts Catalog for this information.

A. Install pushrod assembly 159CM10453.

B. Install crank assembly 159CM10421 and pin in neutral position.

C. Adjust pushrod assembly 159CM10453 for 0° trail position of tab.

D. Attach short sections of cable assemblies 159C10428-3 and -7 to actuator 159SCC100. Set actuator for neutral position. Rotate actuator to extreme travel, back off 4 3/4 turns. Check for an equal number of wraps of each cable on actuator drum.

E. Install actuator on front beam of rudder and set in neutral position.

F. Install idler assembly 159CM10419, two crank assemblies 159CM10420, link assembly 159C10419, and pushrod assembly 159C10424 and pin in neutral position.

G. Adjust pushrod assembly 159C10424 to pick up crank assembly 159CM10421.

H. Adjust rod end on actuator to pick up link assembly 159C10419.

I. Install long sections of cable assemblies 159C10428-3 and -7 from aft end of aircraft to forward end under cockpit floor.

J. Attach cables to drum assembly 159CM10406 and install drum under floor of cockpit. Set drum for neutral position (i.e., an equal number of wraps of each cable on drum).

K. Install directional trim control wheel and torque tube assembly 159C10401 with wheel reading 0° trim.

L. Tension cables to 45 pounds ± 10% (for each 10° above 70°F add 1.5 pounds, for each 10° below 70°F subtract 1.5 pounds) and safety cable adjusters. Install cable seals GS 13F-3 at Fuselage Station 561.

M. Remove all rigging pins and cycle trim control wheel. Check for freedom of motion and tab deflection. Rotate trim control wheel to NOSE LEFT, trim tab should move to right, rotate wheel to NOSE RIGHT, trim tab should move to left.

NOTE: A shear pin has been incorporated in trim system. It is an AN456-AD4 rivet connecting torque tube assembly (159C10401) to trim control wheel. In case of failure of shear pin, check entire system for a jam condition. All rigging holes are 1/4 inch diameter. Use AN4 bolts or equivalent as rigging pins.

2. Aileron Trim Tab — Rigging

(See Figure 202)

NOTE: Part numbers are used in this manual only as a means of identification when nomenclature alone is inadequate. This manual must not be used for identifying spare parts by number. Consult Gulfstream I Illustrated Parts Catalog for this information.

A. Install two crank assemblies 159CM10407 in left aileron and pin in neutral position.

B. Install pushrod assemblies 159C10056-1, -3 and adjust for 0° trail position of tab using throwboard 159GT1005-9 at Aileron Station 82.

C. Install short sections of cable assemblies 159C10428 on actuator 159SCC100. Set actuator for neutral position. Rotate actuator to extreme travel, back off 4 3/4 turns. Check for an equal number of wraps of each cable on the actuator drum.

D. Install actuator on front beam of aileron and set in neutral position.

E. Install two pushrod assemblies 159C10056 and adjust rod ends on actuator.

F. Install long sections of cable assemblies 159C10428. These cables run along front beam of wing and turn forward to run under cockpit floor.

G. Install cables on drum assembly 159CM10406 and install drum under floor of cockpit. Set drum for neutral position (i.e., an equal number of wraps of each cable on drum).

H. Install lateral trim control wheel and torque tube assembly 159C10401 with wheel reading 0° trim.

I. Tension cables to 45 pounds ± 10% (for each 10° above 70°F add 1.5 pounds, for each 10°F below 70°F subtract 1.5 pounds) and safety turnbuckles. Install cable seals GS 13F-3 at Fuselage Station 285 (left side only).

J. Remove all rigging pins and cycle trim control wheel. Check for freedom of motion. Double check tab deflection versus wheel motion.

K. In event of an excessive right wing heavy condition, in excess of 5° trim, droop both ailerons 1° and left trim tab as follows:

(1) Install rigging pins in left and right wing crank assembly 159CM10203.

(2) Using aileron throwboard 159GT1006, adjust pushrod 159CM10219 for 1° down aileron (left and right).

(3) Re-rig left trim tab 2 to 2 1/2° down with cockpit trim indicator at 0° reading.

(4) Remove rigging pins and cycle control wheel left and right to check for freedom of movement.

(5) Cycle lateral trim control in cockpit, while checking for freedom of movement.

NOTE: If procedure outlined above does not correct a right wing heavy condition, following alternate method may be applied:

(6) Rig trim tab to 0° (Trim control knob in cockpit also at 0°.)

(7) Rig left spring tab 3° down by adjusting pushrods 159C10070-1, and right spring tab 1° up.

3. Elevator Trim Tab — Rigging

(See Figure 203).

NOTE: Part numbers are used in this manual only as a means of identification when nomenclature alone is inadequate. This manual must not be used for identifying spare parts by number. Consult Gulfstream I Illustrated Parts Catalog for this information.

A. Install 4 crank assemblies 159CM10407, 2 in left and right elevators and pin in neutral position.

B. Install pushrod assemblies 159C10056 in left and right elevators and adjust for 0° trail position of tab.

C. Install short sections of cable assemblies 159C10426 on actuators 159SCC100. Set actuators for neutral position. Rotate actuators to extreme travel, back off 4 3/4 turns. Check for an equal number of wraps of each cable on actuator drum.

D. Install actuators, 1 each on front beam of left and right elevators.

E. Install 4 pushrod assemblies 159C10056, 2 in each of left and right elevators and adjust rod ends on actuators.

F. Install long sections of cable assemblies 159SC10426 from aft end of aircraft to forward end under cockpit floor.

G. Install cables on drum assembly 159CM10406 and install drum under floor of cockpit. Set drum for neutral position (i.e., an equal number of wraps of each cable on drum).

H. Install torque tube assembly 159C10401 with wheels reading 0° trim.

I. Tension cables to 45 pounds ± 10% (for each 10° above 70°F add 1.5 pounds, for each 10° below 70°F subtract 1.5 pounds) and safety turnbuckles and cable adjusters. Install cable seals GS13F-3 at Fuselage Station 561.

J. Check for clearance of cable adjusters, on cable assemblies 159C10426, and frames at Fuselage Station 618 and 687.

K. Remove all rigging pins and cycle trim control wheel. Check for freedom of motion. Double check tab deflection versus wheel motion.

4. Aileron Trim Tab — Removal / Installation

A. Preparation (Before initiating aileron trim tab removal, perform following)

WARNING: ENSURE PERSONNEL AND EQUIPMENT ARE CLEAR OF CONTROL SURFACES BEFORE OPERATING.

(1) Fair tab surface with upper surface of aileron.

(2) Install pendulum protractor and set to 0°.

(3) In cockpit, adjust aileron trim tab wheel to 0° and record trim tab preset.

(4) Rotate trim wheel to full counterclockwise - (tab down) and record reading.

(5) Rotate trim wheel to full clockwise - (tab up) and record reading.

(6) Remove pendulum protractor.

B. Removal

(1) Raise tab and expose bolts that connect rods to tab.

(2) Remove hinge bolts.

(3) Disconnect bonding lead from tab.

(4) Remove tab.

C. Installation

(1) Move tab into position and align with attaching hinge fittings.

(2) At inboard hinge point install NAS1104-I4D bolt with AN960D416 washer under bolt head. Install AN960D416 washer and AN320-4 nut. Tighten hand tight and safety with cotter key.

(3) At outboard hinge point install NAS1104-I4D bolt with AN960D416 washer under bolt head. Install AN960D416 washer and AN320-4 nut. Tighten hand tight and safety with cotter key.

(4) Connect bonding lead.

(5) Connect inboard pushrod with NAS1104-14 bolt, AN960D416 washer under bolt head AN960D416 washer and AN320-4 nut. Tighten hand tight and safety with cotter key.

(6) Connect outboard pushrod with NAS1104-14 bolt, AN960D416 washer under bolt head, AN960D416 washer and AN320-4 nut. Tighten hand tight and safety with cotter key.

(7) With trim indicator in cockpit on 0° adjust rods until tab aligns with trailing edge of aileron. Tighten jam nuts.

NOTE: If trim tab was previously rigged with preset, return trim tab to that preset.

(8) Using throwboard check tab throws:

- Full up - 20° $\pm$ 2°
- Full down - 20° $\pm$ 2°

5. Aileron Spring Tab — Removal / Installation

A. Removal

(1) Apply gust lock in cockpit.

(2) Move control wheel to raise tab.

WARNING: ENSURE MAN IN COCKPIT HOLDS CONTROL WHEEL IN POSITION DURING REMOVAL OF ACTUATOR ROD BOLTS. FAILURE TO DO SO CAN RESULT IN SERIOUS INJURY TO PERSON REMOVING BOLTS.

(3) Disconnect actuator rods from tab.

(4) Remove tab.

B. Installation

(1) Install tab.

(2) Move control wheel in direction to raise tab.

(3) Install actuator rods being careful to warn man in cockpit to hold control wheel during installation of bolts.

(4) After bolts are installed, return aileron controls to neutral.

(5) Install throwboard.

(6) With gust lock still engaged, rotate aileron control fully in both directions. Spring tab travel must be 15° ± 1° UP, 15° ± 1° DOWN.

(7) Check neutral position: left tab 1° DOWN, right tab 1° UP.

(8) Remove throwboard.

6. Aileron Spring Tab Bearing / Fitting — Inspection

A. Remove spring tab. (See Aileron Spring Tab — Removal / Installation, this Section)

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP SOLVENT AWAY FROM OPEN FLAME.

B. Thoroughly clean areas to be inspected using cleaning solvent or equivalent.

C. Carefully inspect control surface fittings and mating fitting for evidence of corrosion, wear or chafing; check for signs of cracks, especially at areas adjacent to bearing and bolt attachment points.

D. Carefully check fittings for irregular surfaces under primer and for evidence of granular corrosion.

E. Inspect bearings for security of mounting, evidence of rust and for roughness when rotated under load.

F. Replace fittings that are worn, chafed, corroded, deeply scratched or loose.

G. Replace bearings that are loose, rusted. or rough when rotated.

H. Install spring tab. (See Aileron Spring Tab — Removal / Installation, this Section)

7. Aileron Trim Tab Bearing / Fitting — Inspection

A. Remove aileron trim tab. (See Aileron Trim Tab — Removal / Installation, this Section.)

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP SOLVENT AWAY FROM OPEN FLAME.

B. Clean areas to be inspected with cleaning solvent.

C. Control surface fittings and mating fittings for evidence of corrosion, wear or chafing. Check for signs of cracking, especially at areas adjacent to bearing and bolt attachment points.

D. Fittings for irregular surfaces under primer and granular corrosion.

E. Bearings for security of mounting, corrosion and roughness when rotated under load.

F. Replace fittings that are worn, chafed, corroded, deeply scratched or loose.

G. Replace bearings that are loose, rusted or rough when rotated.

H. Install aileron trim tab. (See Aileron Trim Tab — Removal / Installation, this Section.)

8. Aileron Trim Tab Actuator — Removal / Installation

A. Removal

(1) Remove access covers at aileron trim tab actuator.

(2) Remove left outboard leading edge, loosen cables and install cable clamps.

(3) Disconnect trim tab cables at connectors (Wing Station 414) and remove two pulleys.

(4) Attach cord to both tab cables leading to aileron.

(5) Disconnect both pushrods on either side of trim actuator.

(6) Unbolt actuator from aileron spar and remove actuator.

NOTE: When removing actuator, pull tab cables through wing until cords attached to cables are exposed. Untie cords from cables and secure cords.

(7) Rotate actuator drum fully in one direction, remove Allen screw and cable; rotate actuator drum fully in opposite direction, remove Allen screw and remaining cable (mark cables top and bottom for correct installation).

B. Installation

NOTE: Trim tab actuator (P/N 159SCC100-11) is to be used on elevator tab position only and is not interchangeable with either aileron or rudder trim tab positions.

(1) Position actuator on aileron and secure.

(2) Tie cable ends from actuator to cord ends; pull cables through to leading edge of wing and install pulleys.

(3) Connect tab cables and tension to 45 $\pm$ 4.5 lbs.

NOTE: For each 10° above 70°F add 1.5 lbs. For each 10° below 70°F subtract 1.5 lbs.

(4) Connect pushrods to both terminals of actuator; with trim indicator on 0° in cockpit, adjust terminals to neutral tab.

(5) Using throwboards check following:

- Trim control wheel fully counterclockwise - tab down 20° $\pm$ 2°.
- Trim control wheel fully clockwise - tab up 20° $\pm$ 2°.

(6) Safety turn buckle.

(7) Install leading edge and access covers.

9. Elevator Trim Tab — Removal / Installation

A. Preparation (Before beginning elevator trim tab removal, perform following procedure)

CAUTION: AVOID EXCESSIVE FORCE AGAINST LIMIT STOP.

(1) Fair elevator trim tab with upper surface of elevator.

(2) Install neutral board (protractor) and set to 0°.

(3) In cockpit, adjust elevator trim tab wheel to 0° and record trim tab preset.

(4) Rotate trim wheel to full tab down (nose up) position and record setting.

(5) Rotate trim wheel to full up (nose down) position and record setting.

(6) Remove neutral board (protractor).

B. Removal

NOTE: Apply a liberal amount of penetrating oil to mounting hardware before removal, if possible.

(1) Lower trim tab and disconnect pushrods at tab.

(2) Remove tab hinge bolts and bonding wires.

(3) Remove tab.

C. Installation

NOTE: Coat bolts with MIL-G-21164 (MoS_2) before installation.

(1) Position tab on elevator, install bonding wires.

(2) Insert bolts (NAS1104-10D) with washer (AN960D416) under each bolt head at extreme inboard and outboard hinges.

(3) Insert bolts (NAS1104-15D) with washer (AN960D416) under each bolt head into each of remaining hinge fittings.

(4) Install washers (AN960D416) and nuts (AN320-4) on each bolt. Torque nuts to 30-40 inch pounds and safety with cotter pins (MS24665-153).

(5) Connect pushrods to tab. Torque nuts to 30-40 inch pounds and safety with cotter pins.

NOTE: To increase trim tab travel, shorten rod end. To decrease travel, extend rod end.

(6) Using throwboard, set neutral at Stabilizer Station 82 by adjusting pushrods. Check and record tab throws as follows:

- Tabs full UP 2° ± 1/2° (Aircraft 1 - 25 and 114)
- Tabs full UP 3° ± 1/2° (Aircraft 26 - 200, 322 and 323)
- Tabs full DOWN 20° ± 2°

10. Elevator Trim Tab Horn Linkage — Inspection

A. Inspect elevator trim tabs for free play by checking the vertical movement at tab trailing edge in the vicinity of each pushrod. Maximum allowable movement is 0.030 inch relative to trailing edge of elevator. If movement is greater than 0.030 inch, check through the linkage between tab and actuator to determine cause of excessive free play.

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP SOLVENT AWAY FROM OPEN FLAME.

B. Clean horns carefully, ensuring that cleaning solvent does not contaminate grease in rod end bearings.

C. Inspect the horns for cracks visually, in good lighting using a 10X magnifier. A tab with a cracked or broken horn should be corrected before further flight.

NOTE: If a tab is reworked or replaced, the surface mass balance must be checked.

11. Elevator Trim Tab Bearing / Fitting — Inspection

A. Remove elevator trim tab. (See Elevator Trim Tab — Removal / Installation, this Section)

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP AWAY FROM OPEN FLAME.

B. Thoroughly clean areas to be inspected with approved cleaning solvent.

C. Carefully inspect control surface fittings and mating fitting for evidence of corrosion, wear or chafing, check for signs of cracks especially at areas adjacent to bearing and bolt attachment points.

D. Carefully check fittings for irregular surfaces under primer and for evidence of granular corrosion.

E. Inspect bearings for security of mounting, evidence of rust and for roughness when rotated under load.

F. Replace fittings that are worn, chafed, corroded, deeply scratched or loose.

G. Replace bearings that are loose, rusted or rough when rotated.

H. Install trim tab. (See Elevator Trim Tab — Removal / Installation, this Section)

12. Elevator Trim Tab Actuator — Removal / Installation

A. Removal

(1) In cockpit, tag or tie down elevator trim tab control wheel at 0° to prevent movement.

(2) Remove access covers in area of trim tab actuator.

(3) Install cable clamps at pulleys, disconnect trim and traverse cable.

(4) Loosen left or right elevator trim tab cable at connector in tail compartment.

(5) Attach cord to ends of traverse cables and secure to structure.

(6) Separate pushrod from actuator and remove bolts.

(7) Remove actuator.

(8) Unreel upper tab cable at actuator while maintaining tension on lower tab cable to prevent fouling lower tab cable on drum.

(9) Remove socket head screw and remove cable.

(10) Unreel lower tab cable, remove socket head screw and remove cable.

(11) Tag cables and secure to stabilizer structure.

B. Installation

(1) Extend cable drum fully in one direction, install lower tab cable on actuator and secure with socket head screw. Reel drum until cable is fully wound on drum.

(2) Install upper tab cable, secure with socket head screw. Reel drum by pulling lower cable until both cables are midway on drum.

(3) Install actuator with bolts and connect actuator pushrod to actuator.

(4) With elevator trim at 0° in cockpit, using neutral board at Elevator Station 82, adjust terminals on actuator to obtain tab neutral.

(5) Connect cables and remove all cable clamps.

NOTE: Add 1.5 pounds for each 10°F above 70°F. Subtract 1.5 pounds for each 10°F below 70°F.

(6) Adjust cable tension to 45 ± 5 pounds.

(7) Check elevator tab throws:

- Elevator trim wheel full fwd - tabs UP 2° ± 1/2° (Aircraft 1 - 25 and 114), 3° ± 1/2° (Aircraft 26 - 200, 322 and 323).
- Elevator trim wheel full aft - tabs DOWN 20° ± 2°

(8) Safety turnbuckles.

(9) Install access covers.

13. Rudder Spring / Trim Tab — Removal / Installation

A. Removal

(1) Using trim wheel, crank tab to full nose right. Tab will move left to expose pushrod bolt.

(2) Disconnect pushrod from tab crank, remove rubber plugs and bolts.

(3) Hold tab away from rudder, remove bonding wires and remove tab.

B. Installation

(1) Hold tab in position and install bonding wires.

(2) Install tab with bolts and install rubber plugs.

(3) Attach pushrod to tab crank and set rudder trim wheel to 0° Tab should be neutral and align with fixed trailing edge of rudder.

(4) Using throwboard check trim tab throws:

- Rudder trim wheel full right - tab left 10° ± 1°
- Rudder trim wheel full left - tab right 10° ± 1°

14. Rudder Spring / Trim Tab Bearing / Fitting — Inspection

A. Remove spring / trim tab. (See Rudder Spring Tab / Rudder Trim Tab / Removal / Installation, this Section)

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP SOLVENT AWAY FROM OPEN FLAME.

B. Clean areas to be inspected with cleaning solvent.

C. Control surface fittings and mating fitting for evidence of corrosion, wear or chafing. Check for cracks especially at areas adjacent to bearing and bolt attachment points.

D. Fitting for irregular surface under primer and corrosion.

E. Bearings for security, evidence of corrosion and roughness when rotated under load.

F. Replace defective parts.

G. Install rudder tab. (See Rudder Spring Tab / Rudder Trim Tab / Remove / Installation, this Section)

15. Rudder Trim Tab Actuator — Removal / Installation

A. Removal

(1) Open bottom access door on fin; open access plate on right side of rudder at trim tab actuator.

(2) Remove cable tension and install two clamps on trim tab cables at Fuselage Station 618 to prevent slack in cables forward of this point.

(3) Disconnect cables at cable adjusters aft of Fuselage Station 618.

(4) Disconnect terminal at lower end of actuator, remove mount bolts.

(5) Remove actuator.

(6) While maintaining tension on one tab cable, unreel mating cable from actuator; remove socket head screw and remove cable.

(7) Unreel and remove socket head screws and remaining cable from actuator; tie cable ends to structure.

B. Installation

NOTE: Trim tab actuator (P/N 149SCC100-11) is to be used on elevator tab position only and is not interchangeable with either aileron or rudder trim tab positions.

(1) Install one tab cable with socket head screw on actuator and reel drum until cable is fully wound.

(2) Install remaining cable with socket head screw and reel drum by pulling first cable until both cables are midway on drum.

(3) Install actuator with bolts, connect cables at Fuselage Station 618, remove cable clamps and tension cables to 45 ± 5 pounds.

NOTE: For each 10°F above 70°F, add 1.5 pounds. For each 10°F below 70°F, subtract 1.5 pounds.

(4) Set trim wheel to 0°; adjust bottom terminal of actuator to line up with linkage, and install bolt.

(5) Safety cable connectors and check all hardware safeties.

16. Trim Tab Actuator — Lubrication

NOTE: This procedure is identical for all trim tab actuators. See Chapter 12 for approved lubricates.

A. Remove trim tab actuator. (See Rudder Trim Tab Actuator — Removal / Installation, this section)

B. Remove screws on both sides of actuator that secure rubber boots to housing.

C. Separate boots and retainer rings from housing.

D. Operate actuator to extend Acme screw fully to one side; apply a light coating of lubricant to screw.

E. Extend actuator Acme screw fully to other side and apply MIL-M-7866 to screw.

F. Secure rubber boots and retainer rings to housing with screws; safety-wire screws in pairs.

G. Install trim tab actuator. (See Rudder Trim Tab Actuator — Removal / Installation, this section)

17. Rudder Spring Tab Lockout Cable — Removal / Installation

A. Removal

(1) Energize main and essential dc buses.

WARNING: ENSURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR BEFORE OPERATING FLAPS.

(2) Lower flaps to full down position.

(3) Remove center floorboards 15 and 26.

(4) Open tail compartment access door on right side of fuselage just under leading edge of stabilizer.

(5) De-energize main and essential dc buses and pull FLAP CONT and MAIN DOOR CONT circuit breakers.

(6) Install rigging pin in sector at Fuselage Station 693.

(7) Loosen turnbuckles to lockout cables aft of Fuselage Station 640 and remove aft cables.

(8) Remove pulleys or cable guides as necessary throughout cable run.

(9) Attach cord to aft end of forward cables, disconnect cables at flap gearbox and carefully pull through pressure seals.

(10) Disconnect cables from cord.

B. Installation

(1) Attach cables to cord and pull through cable run; attach to flap gearbox and secure pulleys and guides.

(2) Install cables on sector at Fuselage Station 693.

(3) Connect cables with turnbuckles and tension to 45 $\pm$ 5 pounds; remove rigging pin at Fuselage Station 693. (Rigging pin should be free to remove without binding.)

NOTE: For each 10°F above 70°F, add 1.5 pounds. For each 10°F below 70°F, subtract 1.5 pounds.

(4) Lock rudder with gust lock and check that rudder spring tab hits stops.

(5) Depress FLAP CONT and MAIN DOOR CONT circuit breakers, and energize main and essential dc buses. Raise flaps using emergency flap control handle, then return handle to neutral postion.

(6) Reset flap dump valve at Fuselage Station 169 bulkhead. (Aircraft 1 - 5 have two dump valves each.)

(7) With flaps full up, check that rudder spring tab does not move more than 1° when rudder pedals are applied.

(8) Install floorboards and access covers removed.

18. Rudder Spring Tab Lockout and Gust Lock Cable — Inspection

(See Figure 204)

A. Inspect rudder spring tab lockout cable.

NOTE: Step A.(1) - (7) not required on Aircraft having ASC 211. Proceed to Step B.

(1) Gain access to cables in pressurized compartment through Floorboard 24 in baggage compartment.

(2) Inspect cables forward of pressure bulkhead for evidence of frayed, worn, rusted or corroded strands (See Figure 201). Close inspection is necessary in these areas because internal strands may be broken and not obvious.

WARNING: ENSURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR BEFORE OPERATING FLAPS.

(3) Operate flaps to full down position and inspect that portion of cable that was covered by the pressure seals.

(4) Gain access to unpressurized area through the controls access plate on lower left side of fuselage at approximately Fuselage Station 570. Cable will be under the air conditioning duct at approximately left Butt Line 2.

(5) Inspect cables in unpressurized area for evidence of frayed, worn, rusted or corroded strands. Close inspection is necessary in these areas because internal strands may be broken and not obvious.

(6) Raise flaps to full up position and inspect that portion of cable that was covered by the pressure seals. If cable requires replacement.

(7) Inspect area for presents of foreign objects, security of all attachments.

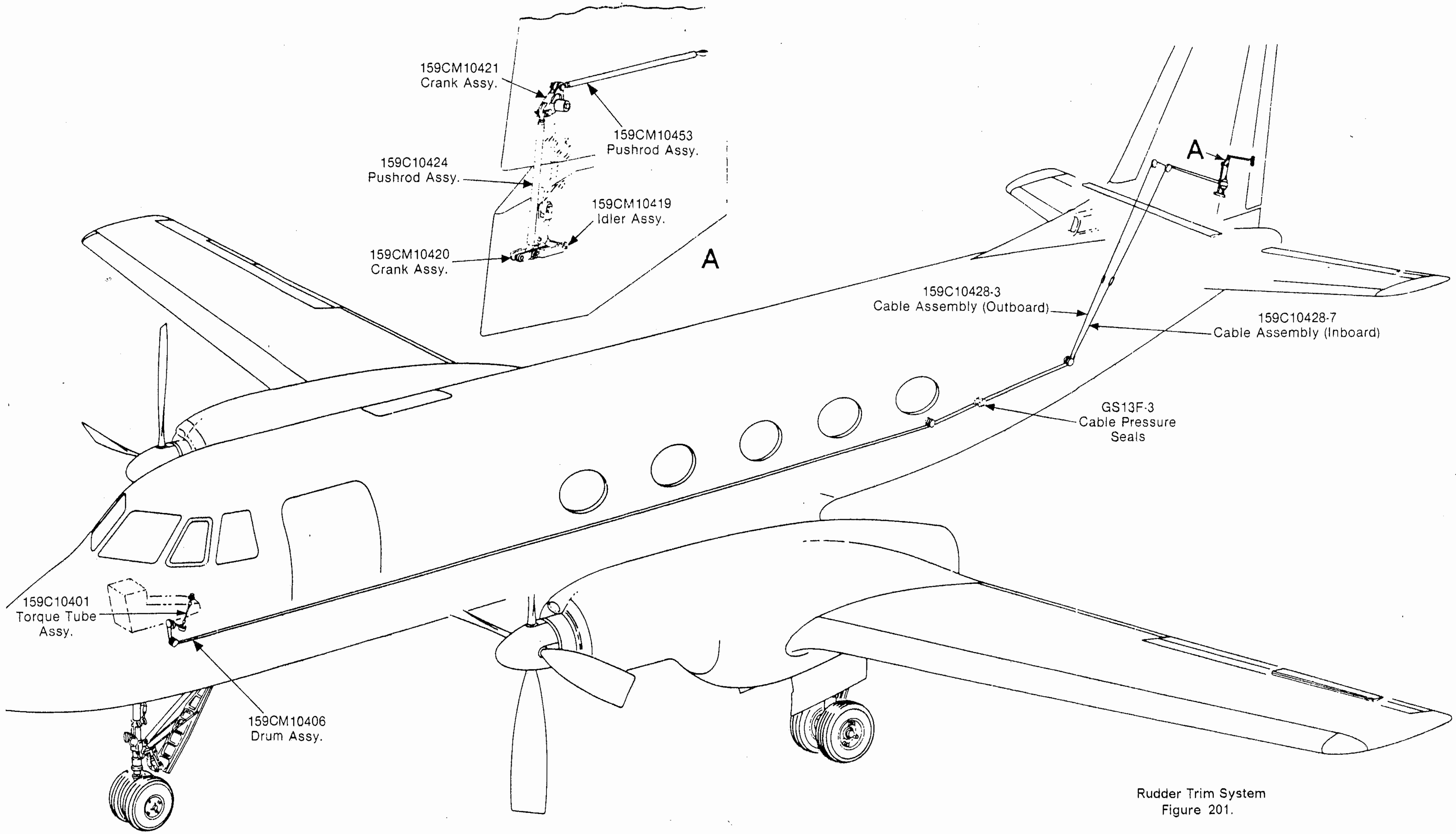
(8) Replace access plate on left side of fuselage and floorboard in baggage compartment.

19. Gust Lock Cable — Inspection

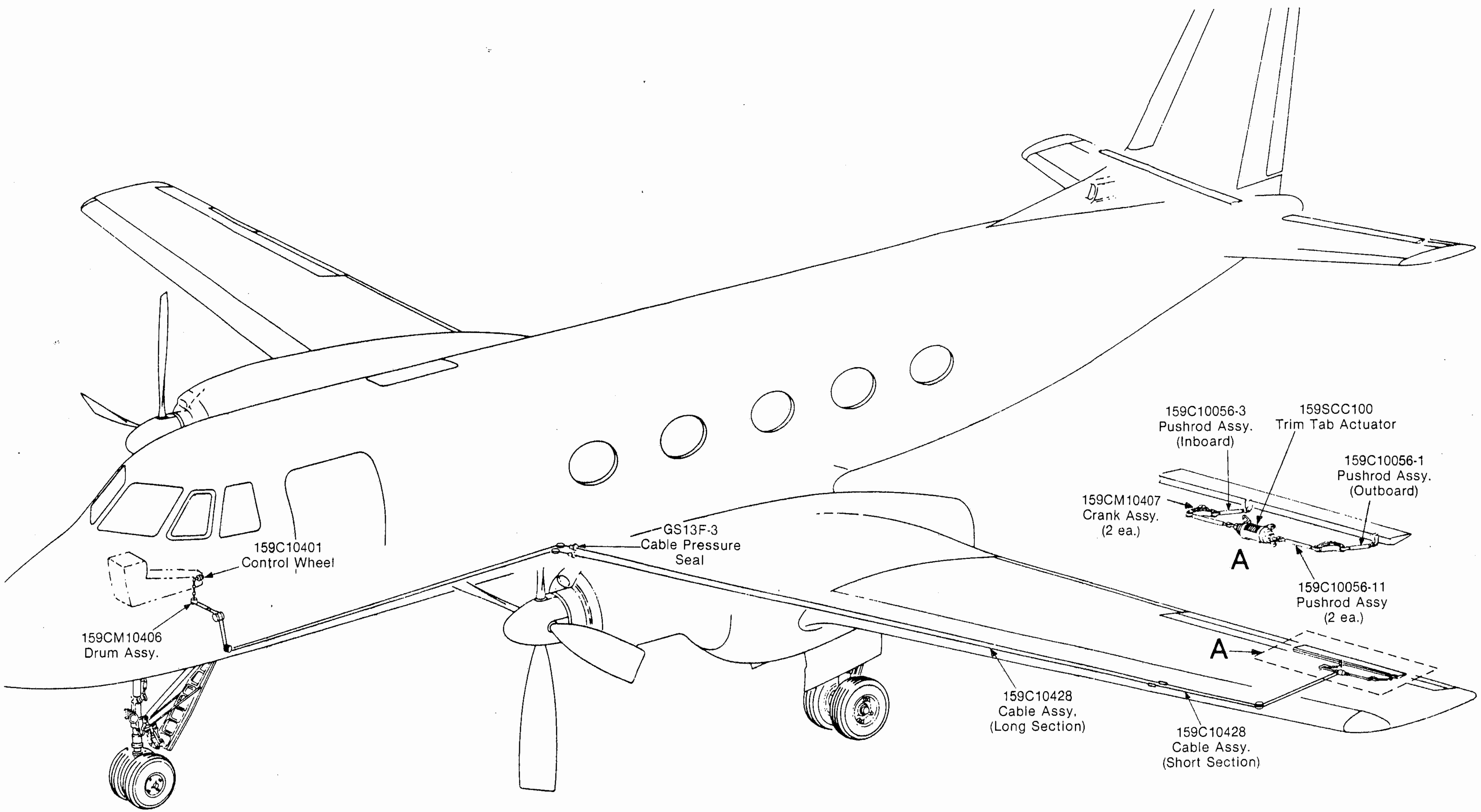
A. Gain access to cables in pressurized compartment through Floorboard 24 in baggage compartment.

B. Inspect gust lock cable for frayed, worn, rusted or corroded strands. Close inspection is necessary where the cables travel over the dual pulleys at Fuselage Station 554.

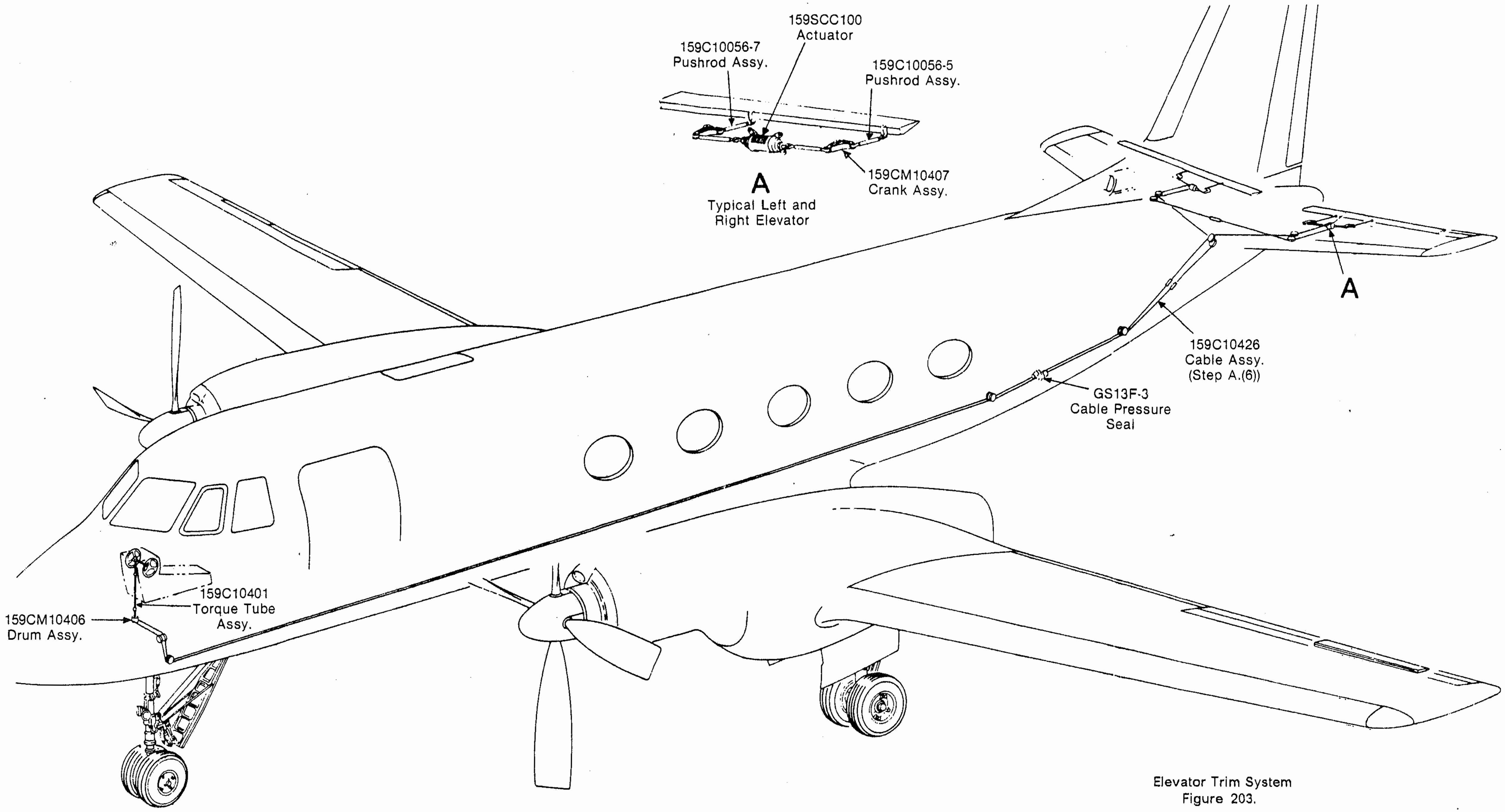
C. Replace access plate on left side of fuselage and the floorboard in the baggage compartment.



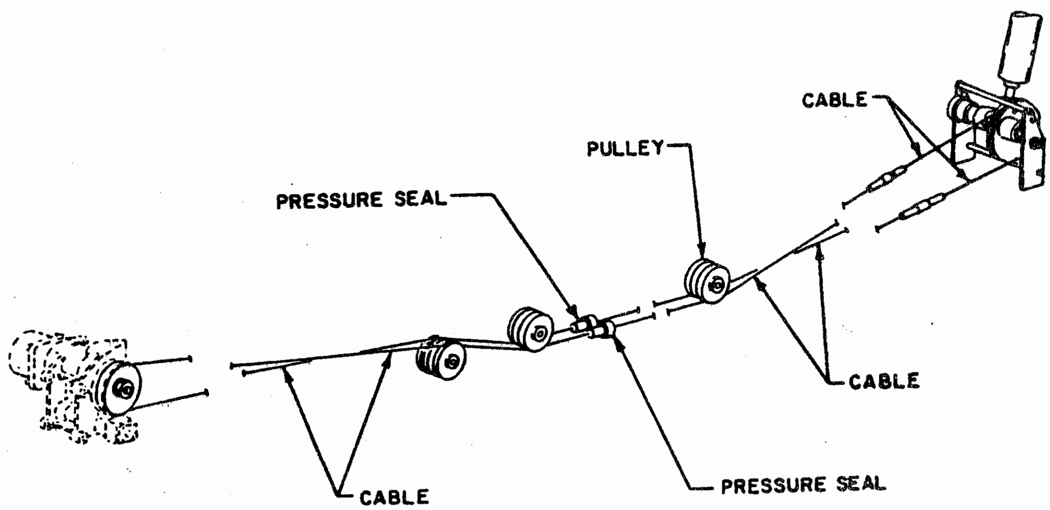
Rudder Trim System
Figure 201.



Aileron Trim System
Figure 202.



Elevator Trim System
Figure 203.



Rudder Spring Tab Lockout Cable
Figure 204.

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GUST LOCK SYSTEM — DESCRIPTION / OPERATION

1. **Description**

The gust lock system is a ground safety device which neither affects the flight performance of the aircraft, nor receives any flight loads. The system consists of cables and pushrods. The gust lock locks the control surfaces against gust loads of up to 60 knots when the aircraft is on the ground.

Gust lock latches are engaged by aft motion of a control lever in the cockpit. The GUST LOCK lever motion is transmitted to the latches through a cable circuit with the final input to the latch being made through a spring bungee in each case. The latches engage the surfaces in neutral position, holding them against any ground gust loads. In case a surface should not be in neutral when the gust locks are applied, the latches each have a cammed lead-in surface. As the control surface moves towards neutral, it progresses along the cam, deflecting the spring bungee. As the surface reaches neutral, the bungee load engages the latch. In the unlocking direction of motion, the bungee acts as a solid link, minimizing any possibility of the locks being engaged or jammed without the pilots knowledge.

The latch point for each surface is made as close to the surface itself as practicable. The latch points for the ailerons are located on the leading edges of the control surfaces. (See Figure 1) The rudder latch point is combined with the rudder stop fitting mounted on the rudder torque tube. The elevator latch point is on the last crank in the control system at Fuselage Station 686. This is the first common point connecting both elevators.

2. **Operation**

(See Figure 1)

NOTE: The GUST LOCK control lever is a two position lever mounted on the upper surface of the center pedestal console. The lever is part of the fine pitch lock selector - power levers interlock system. Moving lever forward to the CONTROLS UNLOCKED position, releases the lock latches, moving the lever aft to the CONTROLS LOCKED position, engages the lock latches. However, operation of a springlatch at the lever knob is required to move the lever to either position. The interlocking device is manually operated by means of the propeller FLIGHT FINE PITCH LOCK SELECTOR handle.

A. With the propeller FLIGHT FINE PITCH LOCK SELECTOR handle in the GROUND (aft) position, the following conditions exist:

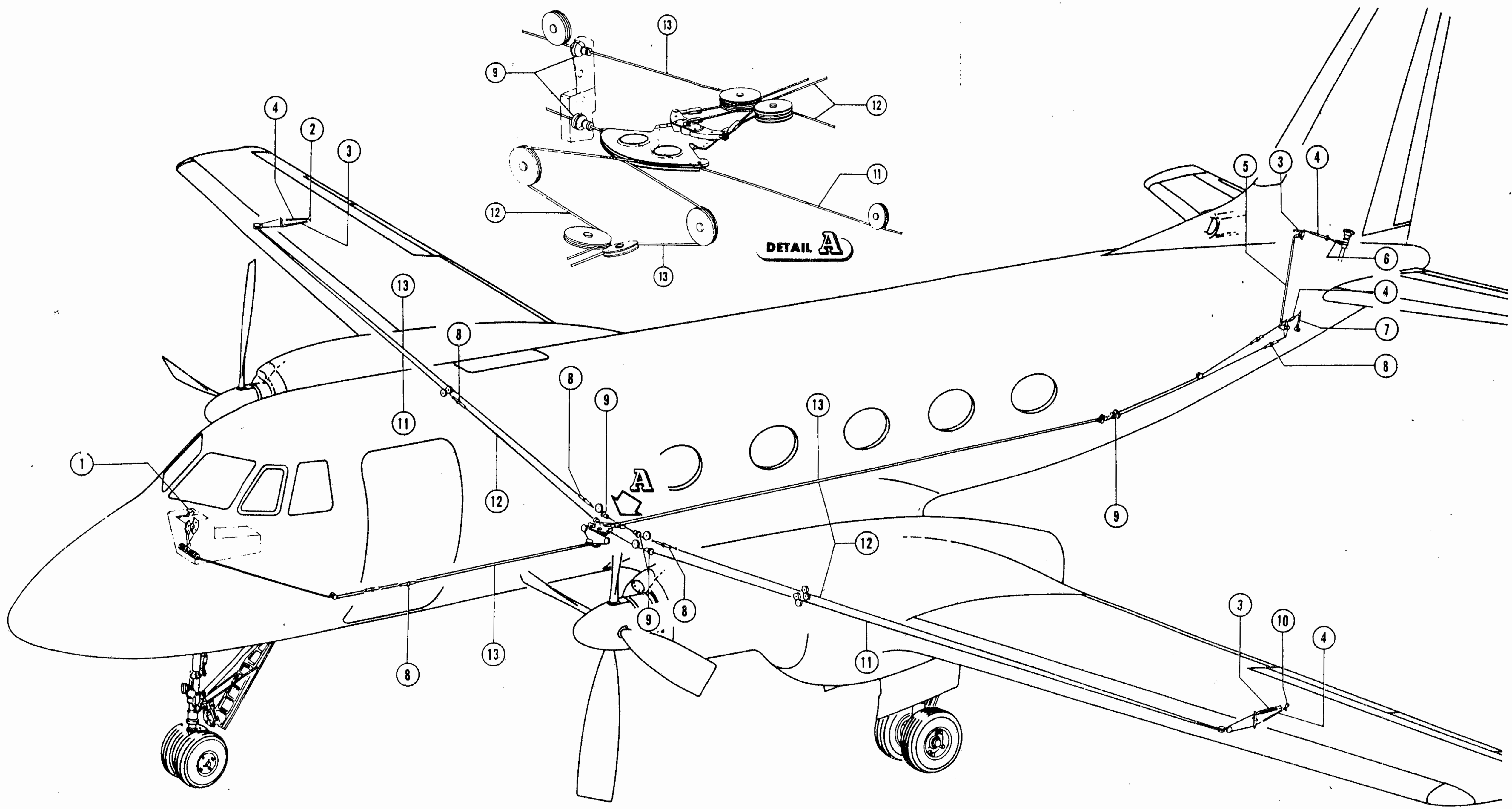
- It is not possible to move both power levers together beyond the 30° forward (11,000 rpm) position with the GUST LOCK lever in the CONTROLS LOCKED position (aft). This prevents takeoff with the controls locked.
- Force applied to advance both power levers does not cause them to override the gust lock interlock. Advancing one power lever to the maximum rpm position automatically retards the other to below the 30° forward setting. This retarding as the result of advancing action enables full power checks on one engine at a time - not both with the controls locked.
- Movement of the GUST LOCK lever into the CONTROLS LOCKED position (aft) is permitted. This movement locks the FLIGHT FINE PITCH LOCK SELECTOR handle in the GROUND INTERLOCK position. The locking device is a fail safe design, whereby it is impossible to move the fine pitch lock handle forward with the GUST LOCK lever aft in the CONTROLS LOCKED position. As a result, inadvertent propeller action is prevented.

B. With the propeller FLIGHT FINE PITCH LOCK SELECTOR handle in the FLIGHT position, the following conditions exist:

- It is impossible to move the GUST LOCK lever aft to the CONTROLS LOCK position. This prevents locking the controls in flight.
- Movement of the power levers is unrestricted from IDLE to MAXIMUM POWER.

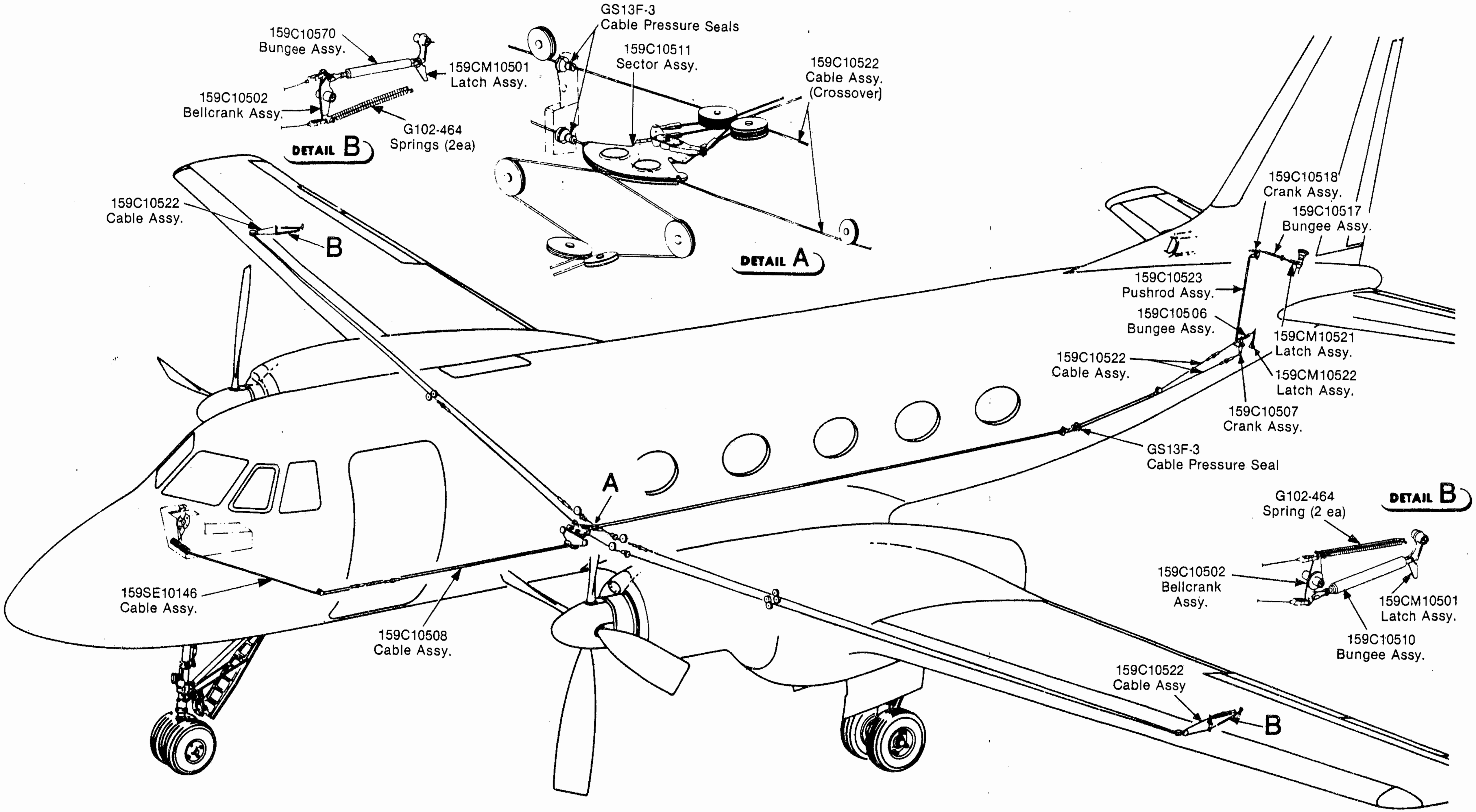
NOTE: Moving the FLIGHT FINE PITCH LOCK SELECTOR handle into the GROUND position with the power levers above 30°, will retard both power levers back to the 30° setting. This is another propeller function safety feature.

With the GUST LOCK lever in the CONTROLS UNLOCKED position, advancement of both power levers past the 30° forward setting engages a mechanism which moves the FLIGHT FINE PITCH LOCK SELECTOR lever from the GROUND position to the FLIGHT position. This movement operates a latching device on the FLIGHT FINE PITCH LOCK SELECTOR handle, locking it in the FLIGHT position.



1. Gust Lock Handle	7. Elevator Latch
2. Right Aileron Latch	8. Turnbuckles
3. Retracting Springs (3)	9. Seals
4. Spring Bungee	10. Left Aileron Latch
5. Push Rod	11. Tie Cable
6. Rudder Latch	12. Lock Cable
	13. Unlock Cable

Gust Lock System
Figure 1.



Gust Lock System Installation
Figure 201.

WING FLAP SYSTEM — DESCRIPTION / OPERATION

1. General

The flaps are four position fowler type flaps which extend from wing station 40 7/8 to 307. They are of the single slotted trailing type having a slot between the flap and the wing when extended. Each flap is extended by two screwjack actuators which are installed at wing station 113.0 and 225.063. They are operated by line shafting extending from the outer actuator gearboxes to a central gearbox at the centerline of the fuselage. (See Figure 201) A hydraulic motor, powered by either the main or auxiliary hydraulic system, drives this gearbox. A solenoid control valve energized through microswitches actuated by the flap control lever and through contacts in the flap programmer (Aircraft having ASC 252) and follow-up switch, controls hydraulic fluid to the motor from the main pressure system. (See Figure 1). A follow-up switch located on the central gearbox, de-energizers the solenoid valve when the flaps reach the selected position, and a transmitter sends information to the flap position indicator on the instrument panel. If there is a loss of pressure in the main hydraulic system or an electrical malfunction of the normal flap system, the auxiliary system must be put into operation so that the flaps may be operated by the emergency flaps control lever. This control, accessible only to the copilot, is mechanically connected directly to the flaps emergency control valve. Shuttle valves are provided to pass the hydraulic fluid from the emergency flap selector valve to the hydraulic motor, returning to normal when main hydraulic power is restored. At the input of the emergency flap selector valve is an electrically controlled shutoff valve. This valve is closed whenever the asymmetry switches located at the outboard end of each drive shaft detect a 0.268 inches differential of ball nut travel between screwjacks. This prevents the emergency system from being used if there has been a failure in the flap system causing asymmetry and therefore, prevents the lowering of a single flap if a broken shaft occurs. A dump valve is provided to shut off normal pressure from the selector valve and remove any hydraulic pressure remaining in the flap system main hydraulic lines. This valve must be manually reset after use of the emergency flap system to restore the system to normal (using main system hydraulic pressure.)

2. Flaps

Each flap is of all metal construction made into a one piece unit. Each flap is mounted by means of four carriage assemblies which roll along support tracks. The carriage assemblies run on tracks riveted to ribs at wing stations 66.5, 133.276, 205.25 and 274.75. Cam followers in each of the center pairs of carriages control the deflection of the flap in a tilted (downward) direction. The outer and inner carriages contain spring bungees which exert a flap hinge movement to hold the cam followers in contact with the cams.

3. Flap Actuators

The flap actuators located at wing station 113 and 255.063 on the left and right wings each consist of a screwjack which drives a traveling nut, the nut being connected to the flap structure. Rotation of the jackshaft either drives the nut aft (extending the flaps) or forward (retracting the flaps). The basic principle of operation of the inboard and outboard actuators is the same, but the outboard actuator output is 5/6 the speed of the inboard actuator with the same input from the drive shafts. This is necessary to allow the outboard actuator to travel a shorter distance than the inboard actuator with the same input in a given time. This enables the flaps to keep in a proper relationship with the wing. The working stroke from full up to full down is accomplished in 30 seconds of jackshaft operation with normal hydraulic pressure of 1500 psi. The travel of the actuator is controlled by the follow-up switches. Non-jamming mechanical stops at the ends of the actuator jackscrew shafts allow the traveling nuts to bottom during emergency operation at 40 degrees of flap travel. The flap actuators contain torque limiters, and limit the loads to 1780 pounds and 1260 pounds outboard/inboard, respectively.

4. **Overload Brakes**

To avoid damage to the flap or structure caused by jammed mechanism or lowering flaps against excessive air loads, a torque limiter is provided at each actuator. This consists of balls held in sockets by springs which pop outward stalling the mechanism if 125 percent of lowering torque is exceeded during the extension cycle. Application of sufficient torque to the input shafting of the actuator displaces the balls outward of their pockets, causing a small amount of shaft displacement which locks the input shaft. This is transmitted back through the drive shafting to the central gearbox. The overload feature acts to prevent the possibility of lowering the flaps into excessive air loads which might incur structural damage to the aircraft. A jam occurring during flap retraction places greater loads on the system than during extension due to the air pressure on the flap. For this reason, retraction torque is limited to 40 percent of that required to extend. This is accomplished by a friction plate at the forward end of the actuator screw shaft which is operated by the displacement of the screw shaft due to tension at the flap connecting link.

5. **"No Back" Device**

To prevent the flaps from backing off (the actuator spinning freely) if the shafting fails while in the extended position, a "no back" device is fitted to each actuator gearbox. This consists of a friction plate which brakes the flap actuators and prevents them from releasing the flap should the shafting fail while the flap is down. The application of torque to the shaft releases the friction plate and allows freedom of movement. The locking action applies only to rotation in the retracting direction, since it is in this direction that air loads react upon the actuators. Mechanical stops at the ends of the screwjacks stop the flaps at the emergency position of 40°.

6. **Flap Control Lever**

NOTE: On Aircraft 2 - 93 and 114 having ASC 108A, Part I and Aircraft 94 - 200, 322 and 323, moving the flap control handle to the APPROACH or FULL detent position with the landing gear not down and locked, the landing gear warning horn will activate and stay activated until all three gear (nose and main) are down and locked. Use of the horn cut-out switch will not silence the horn. (See Figure 2).

Flap control lever, on the right side of center pedestal console, selects 4 positions of flaps; UP, TAKEOFF (at the 12° position), APPROACH (at 20°), and FULL (at 33°). Lever moves vertically into four detented positions, operating microswitches which energize flap control valve solenoids. (See Figure 1).

7. **Flap Programmer Switch**

(Aircraft having ASC 252)

Flap programmer switch is a toggle type switch mounted on center instrument panel, above and to left of flap position indicator. Switch has 2 positions, they are as follows:

Up Position;

T/O 6.5°, APP 12.5° - used for abnormal conditions, (i.e. high altitude / hot day, runway length not limiting).
Down Position;

T/O 6.5°, APP 20° - used for normal conditions. The switch is used in conjunction with the flap control lever.

8. **Flap Control Valve**

The flap control valve is a three position, four way, solenoid operated valve, located in the hydraulic compartment. When energized, it directs hydraulic fluid to either extend or retract the flaps. When de-energized, it is closed to hydraulic fluid flow.

Switches in the follow-up switch assembly mounted on the central gearbox, operate in conjunction with the limit switches in the flap control lever and switches in the flap programmer (Aircraft having ASC 252). When the flaps reach the desired position the flap control valve is de-energized, this stops the hydraulic motor and holds the flaps in the selected position.

9. Restrictor and Filter

A restrictor is located in the hydraulic line at Fuselage Station 169 below the hydraulic reservoir. It is connected at the input of the flap control valve to regulate the flow of hydraulic fluid into the flap control valve and hence, the speed of the hydraulic motor. A filter is installed in each of the two restrictor ports. The restrictor permits a flow of 600 cubic inches of fluid per minute at 1500 psi.

10. Dump Valve

A dump valve is connected between the flap control valve and the two shuttle valves. Whenever auxiliary hydraulic pressure is directed to one of the hydraulic motor ports, the dump valve shuttle is shifted to open a passage from the shuttle valves to the hydraulic reservoir and from the flap control valve to the hydraulic reservoir. The dump valve shuts off all main system hydraulic fluid flow from the flap control valve to the hydraulic motor (up and down pressure at the same time). The auxiliary system may then take over operation of the hydraulic motor. After use of the auxiliary system to operate the flaps, the dump valve must be manually reset. It is located on Fuselage Station 169 bulkhead below the hydraulic reservoir. Before resetting this dump valve, the emergency flap handle must be in neutral position. After re-set is accomplished, the flaps will move to that position selected by normal flap handle at the next application of main system pressure.

NOTE: Aircraft 2 - 6 have two dump valves each with a reset button.

To resume normal operation, switch off auxiliary hydraulic pump, return emergency flap control lever to neutral position, and press reset button on side of flap dump valve or valves. The wing flap system is now in normal configuration. With the normal hydraulic and electrical systems operating, cycle the flaps to insure normal system is operating properly.

11. Shuttle Valves

There are two shuttle valves in the flap control system, one in the retract the other in the extend line, located downstream of the flap control and emergency flap selector valves. When auxiliary hydraulic pressure is released through the auxiliary flap control valve, it shifts the shuttle to close the port to the dump valve. This directs fluid into the motor returning through the other shuttle valve to the dump valve and the reservoir. The shuttle valves are located under floorboard 15 just aft of the hydraulic motor.

12. Emergency Flap Control Lever

The emergency flap control lever, located on the copilots side console can be used by the copilot only to operate the flaps from the auxiliary hydraulic system should main hydraulic pressure fail or an electrical malfunction occur in the normal flap control system. The auxiliary system must be put into operation by starting the auxiliary hydraulic pump. Moving the emergency flap control lever down lowers the flaps; raising it retracts the flaps, control is detented in NEUTRAL or OFF position only. The flap indicator shows the position of the flaps. In addition, the flaps can be moved into an emergency position beyond the FULL position normally used. The maximum emergency down position is 40°.

13. Auxiliary Hydraulic Pump Switches

These switches are located as follows: the pilots switch is at the forward inboard side of the side console, and the copilots switch is at the lower right side of his outboard skirt panel between the hydraulic pressure gages.

14. Asymmetry Switches

A flap asymmetry system is in the aircraft flap system. Asymmetry switching attached to outboard ends of the shafting on each wing are connected in such a manner to form a complete circuit during normal operation of the flaps. (See Figure 3). If movement of either flap is interrupted, it will leave the switching device cams on the affected side static. Continued operation of the opposite wing flap and switching device opens the circuit. Opening the circuit deactivates power to the flap control solenoid valve, stopping the flaps and allows no further use of the flaps. The asymmetry switches also activate a shutoff valve, located in the auxiliary hydraulic supply line making emergency operation of the flaps impossible.

There are three switches in each device. The switches are attached to the case and are set 120° apart around a cam. The cam is rotated by the flap shafting, and is cut with a 180° lobe. The cam actuates the switches. When one flap stops, cam rotation stops. Further rotation of the cam in the device attached to the uneffected

flap drive shaft will break the electrical circuit and deactivate the system. Flap movement will stop in a maximum of 0.268 inches of screwjack ball nut travel.

15. Flap Shutoff Valve

An electrically operated shutoff valve is located on the input side of the emergency flap selector valve. It is normally deenergized and open. Whenever the asymmetry switches detect asymmetry greater than 0.268 inches, the shutoff valve will be energized and close. This prevents the emergency system from being used in case of a failure in the flap system causing asymmetry, and therefore prevents the lowering of a single flap. The shutoff valve is located in the nose wheel well on the right side.

16. Emergency Flap Selector Valve

The manually operated emergency flap selector valve takes fluid from the auxiliary hydraulic system and directs it through shuttle valves to the ports of the hydraulic motor. The emergency flap selector valve is controlled by the emergency flap control lever and can be turned to two positions to provide the two directions of rotation of the hydraulic motor. The emergency flap control valve is located in the nose wheel well on the right side.

17. Hydraulic Motor

The hydraulic motor is attached directly to the gearbox at the centerline of the fuselage under floorboard 15. At 1500 psi, hydraulic pressure, the motor develops 2 HP with the torque increased by the 8.25 to 1 ratio of the gears connecting it to the drive shaft. The drive shaft operates at 727 rpm.

18. Central Gearbox

The central gearbox is driven by a hydraulic motor. Hydraulic pressure drives the motor at 6000 rpm. The speed of the hydraulic motor is reduced by gearing in the central gearbox at a ratio of 8.25 to 1. Torque is transmitted at a speed of 727 rpm via the spanwise shaft to the flap drive actuators. The rudder lockout sheave located on the central gearbox is geared so that when the flaps reach TAKE-OFF position, or any position below take off, the rudder spring tab is unlocked and operative. Also located on the central gearbox are the flap position transmitter and the follow-up switch assembly. The flap position transmitter, followe-up switches and rudder tab lockout sheave, rotate 285° from full up to full down position. On aircraft having ASC 108A, part I or part II, the rudder lockout sheave is modified to a cam. This cam operates a microswitch which introduces flap position as a means of activating the landing gear warning horn system. Aircraft 2 - 93 and 114 having ASC 108A, part I (only) and Aircraft 94 - 200, 322 and 323 see Figure 4, Aircraft 2 - 93 and 114 having ASC 108A, part II (only) (See Figure 5).

19. Drive Shaft

The drive shaft runs from the central gearbox out to actuators at each flap. It is divided into short lengths for ease of installation, and is supported approximately every 3 feet to prevent whip lash. The first support point is where the shaft leaves the fuselage. This opening has a pressure seal and the shaft has universal joints just inside and outside of the seal. The next support point is a pillow block 3 feet out from the fuselage contour having additional universal joints. From here, the shaft runs out to a gearbox at each inner actuator. The use of universal joints allows the system to be free running and compensates for all conditions of wing flexing.

20. Operation

Flap normal operation is controlled by a flap handle located in the pedestal. The handle has four positions, UP, TAKE-OFF, APPROACH and FULL. A switch for each of these four positions is provided in the pedestal. The corresponding switch is actuated when the handle is moved to a particular position.

NOTE: Flap normal operation for Aircraft having ASC 252 is controlled by the flap handle and a flap programmer switch located on the instrument panel. The programmer switch has two positions, takeoff 6.5° and 12.5° and approach, 12.5° and 20°.

The flap asymmetry switches prevent operation of the flaps unless the flaps are in the same position (symmetrical). This is to prevent a lateral aerodynamic unbalance. The asymmetry switches will only allow a maximum of 3 degrees of flap separation.

A typical operation of the flap control system follows:

A. Aircraft not having ASC 252 (See Figure 6).

Assume the flaps are in the UP position, and the pilot selects APPROACH. If the flaps are symmetrical (both up and in the same position), a circuit is completed through the asymmetry switches to the coil of the flap control relay located in the mid relay box under FLR 8. This action holds the relay energized. A circuit is now made through A and B contacts of the relay, through the flap control handle APPROACH microswitch. (This was closed by the placing of the handle in the APPROACH position.) The power then goes through the No. 5 switch of the approach switches in the flap follow-up switches (on the central gearbox under FLR 15), and finally to the extend side of the flap control valve. The valve in the hydraulic compartment opens and the flaps are powered hydraulically to extend. When the flaps reach the APPROACH position, a cam in the flap follow up switch assembly breaks the No. 5 switch. This opens the circuit to the flap control valve, neutralizing the valve and stopping hydraulic flow. The flaps are now, and will remain, in the position selected until another position is selected with the pedestal control handle.

B. Aircraft having ASC 252 (See Figure 7)

Assume the flaps are in the UP position, and the pilot selects APPROACH with the flap control handle and APP. 12.5° on the flap programmer switch. If the flaps are symmetrical, a circuit is completed through the asymmetry switches to the coil of the flap control relay. Their action holds the relay energized. A circuit is now made through A and B contacts of the relay, through the flap control handle APPROACH microswitch. The power then goes through the APP. 12.5° contacts of the flap programmer switch and the 12.5° contacts of the No. 3 T/O and APP switch in the flap follow-up switch, and finally to the extend side of the flap control valve. The valve opens and the flaps are power hydraulically to the extend. When the flaps reach 12.5° position, a cam in the flap follow up switch assembly breaks the No. 3 switch. This opens the circuit to the flap control valve, neutralizing the valve and stopping flow. The flaps are now, and will remain, in the position selected until another position is selected with the pedestal control handle and the flap programmer (Takeoff and approach only).

During any operating cycle, should the flaps become asymmetrical (as in the case of a failure of spanwise shaft), the asymmetry switch circuit is broken and the flap control relay becomes de-energized, halting flap operation. When the flap control relay is de-energized, the following actions take place:

- The normal flap control circuit to the pedestal switches is opened by breaking B2-B1, and A2-A1 contacts, preventing flap movement by normal control.
- A circuit is completed through contacts B2-B3 to the emergency flap stop solenoid valve in the nose wheel well. This valve, when energized, closes, preventing flap movement by means of the emergency flap control handle.

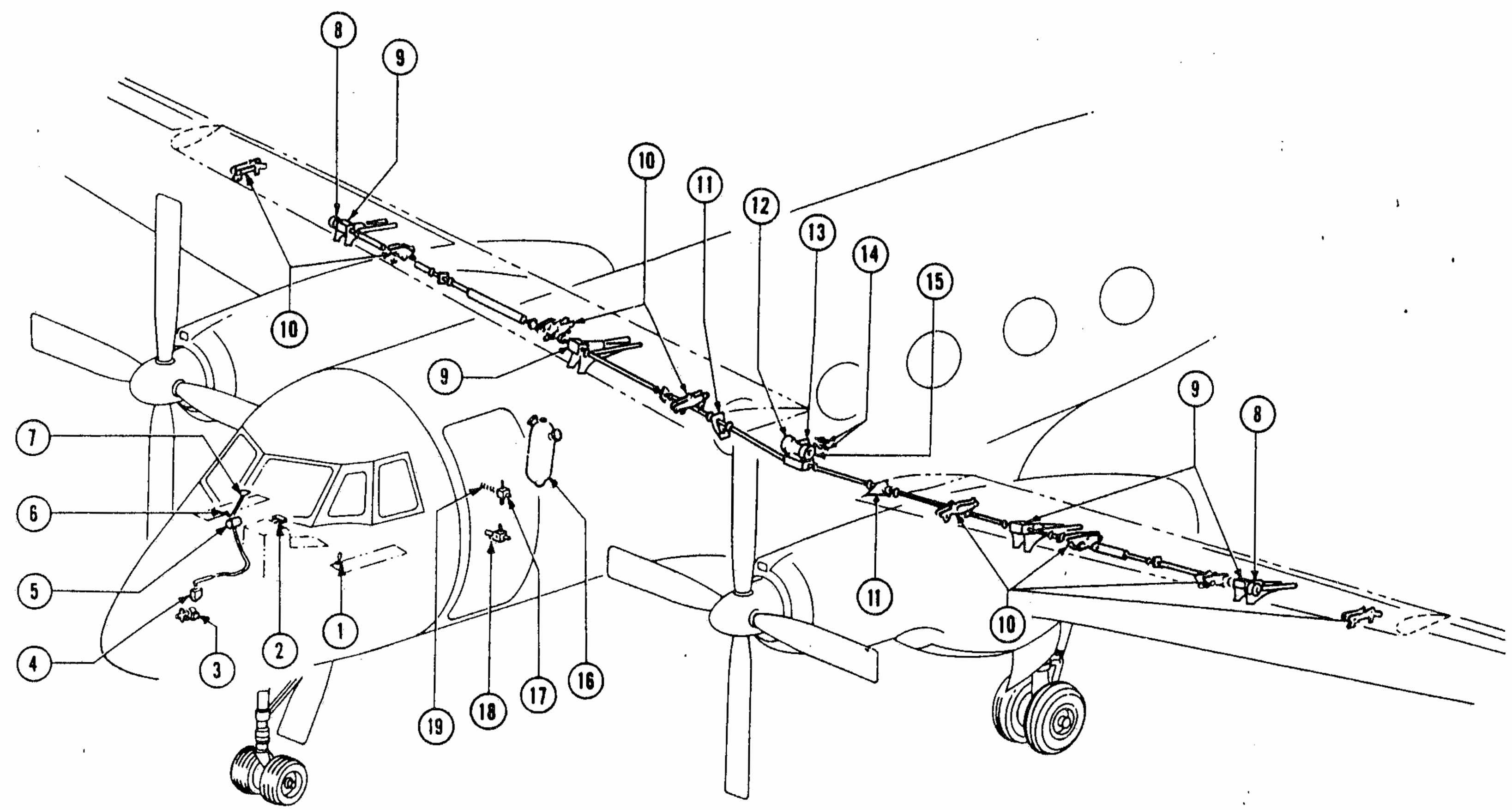
NOTE: An asymmetrical flap condition prevents flap operation by any control system.

WARNING: WHEN MOVEMENT IS STOPPED BY AN ASYMMETRICAL CONDITION, DO NOT, UNDER ANY CIRCUMSTANCES, (IN FLIGHT) PULL THE FLAP CONTROL CIRCUIT BREAKER ON THE PILOTS CIRCUIT BREAKER PANEL. IF THIS IS DONE, ALL ASYMMETRY PROTECTION IS REMOVED FROM THE EMERGENCY FLAP CONTROL SYSTEM.

Flap position is sensed by a transmitter mounted on the central gearbox under floorboard No. 15. This unit has a continuous ring winding, tapped at three points. Two sliding brushes introduce dc power and ground. Three wires connect the transmitter with an indicator on the cockpit center instrument panel. In the indicator, three coils produce fields which vary as the transmitter brushes move, thus causing the pointer to move. (See Figure 8). Since the flap system is of the fowler type, the largest amount of motion is between the UP to TAKEOFF positions. Continuing to subsequent positions, rather small amounts of motion are needed. This condition makes the flap positions on the indicator somewhat condensed at one end of the scale. This type of indication has the advantage of reporting rotations of the flap drive shaft, and therefore, the entire mechanism.

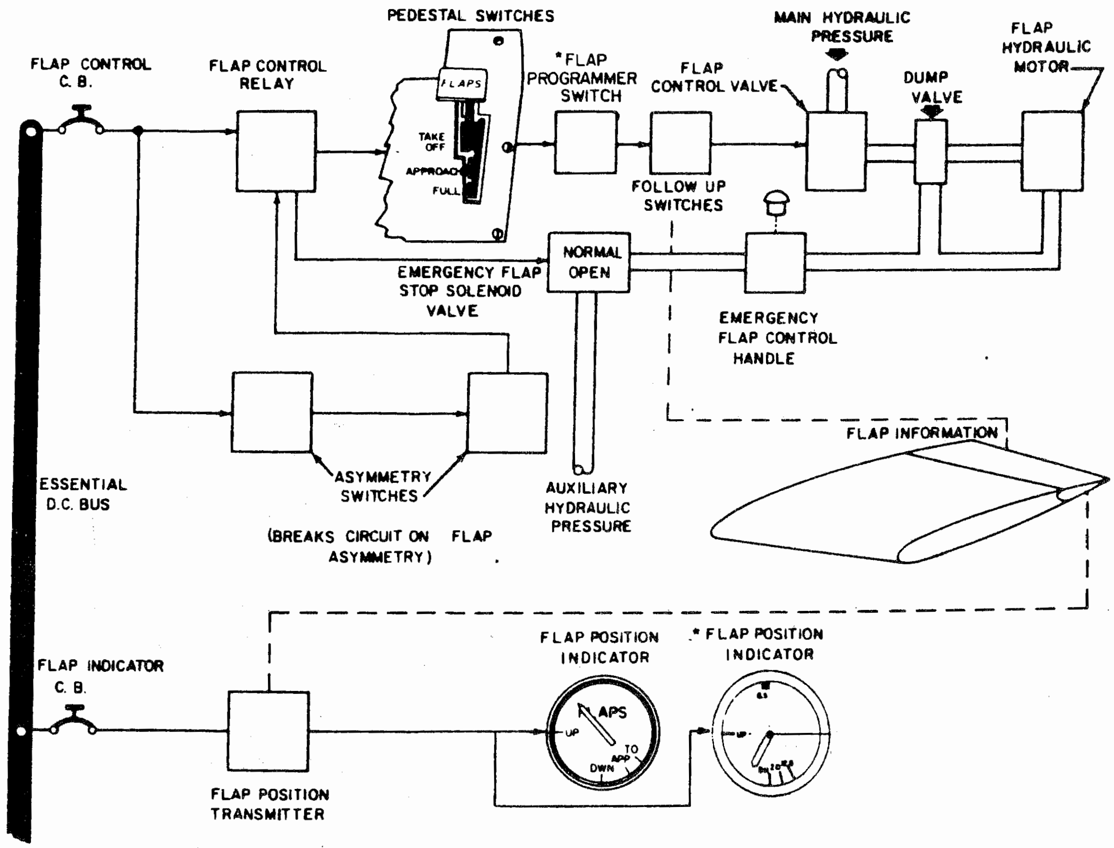
21. Emergency DC Operation

- Flap normal control system is operative in emergency.
- Asymmetry switch protection is fully operative in emergency.
- Flap indication is operative in emergency.
- Emergency flaps extension is not available in emergency due to the auxiliary hydraulic pump being inoperative. (Main DC Bus)



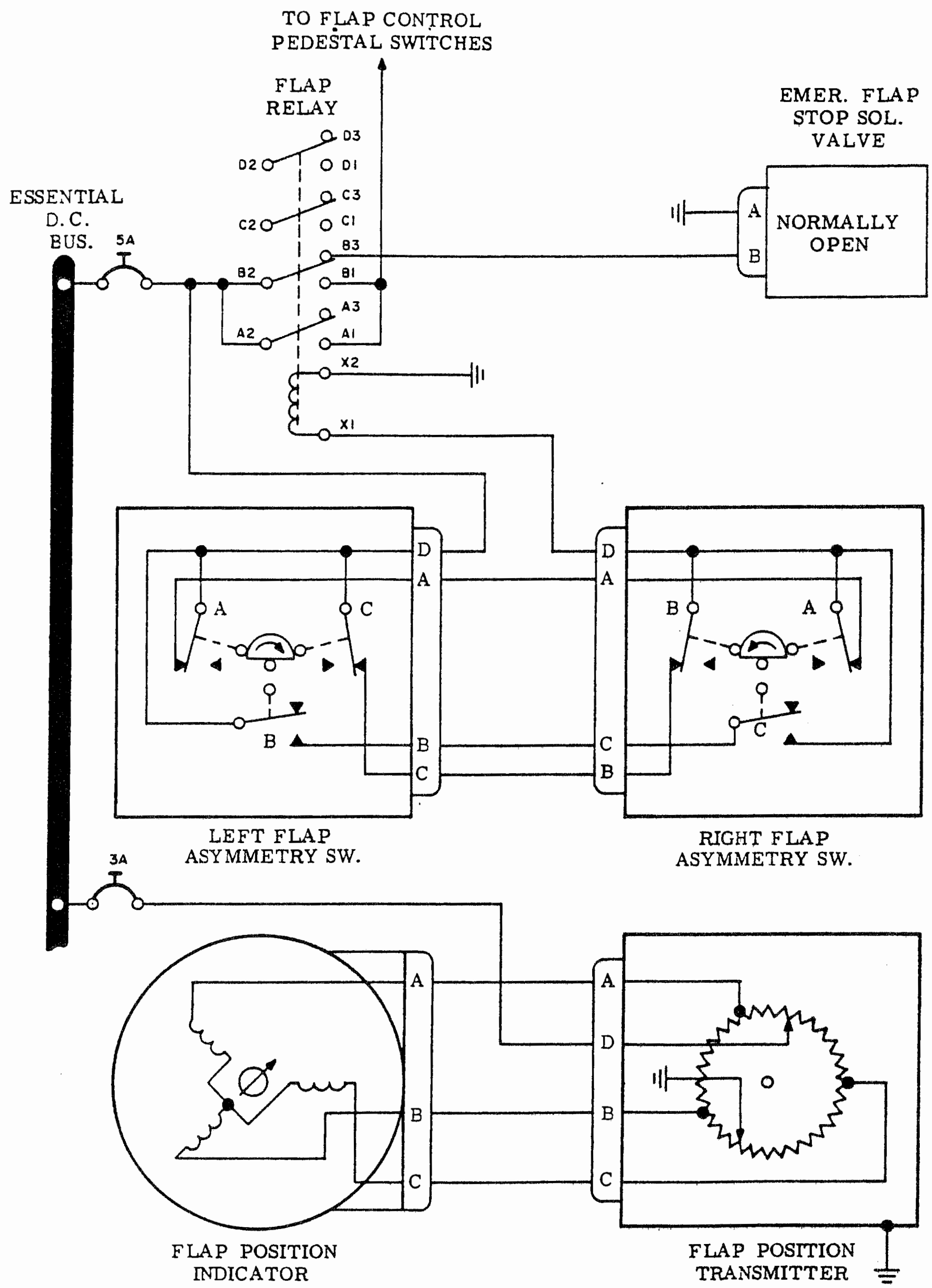
1. Pilot's Auxiliary Hydraulic Pump Switch	10. Carriage
2. Flap Control Lever	11. Drive Shaft Seal
3. Shut-off Valve	12. Follow-up Switch
4. Emergency Flap Selector Valve	13. Hydraulic Motor
5. Flap Position Indicator	14. Shuttle Valves
6. Copilot's Auxiliary Hydraulic Pump Switch	15. Spring Trim Tab Lockout Sheave
7. Emergency Flap Control	16. Hydraulic Reservoir
8. Assymetry Switch (2)	17. Solenoid Control Valve
9. Actuator	18. Dump Valve
	19. Restrictor

Flap System
Figure 1.

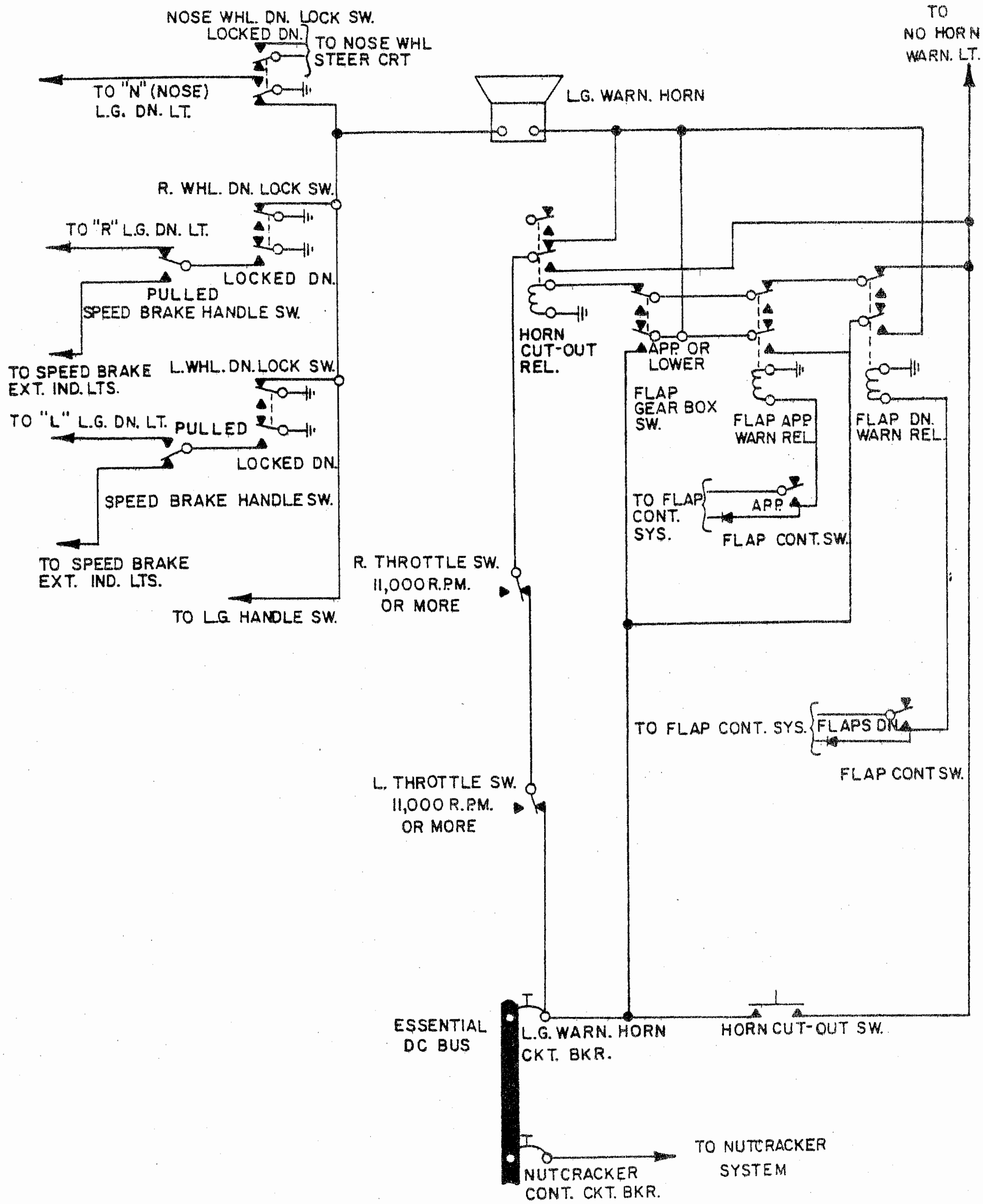


*Aircraft 1 through 200 including 322 and 323 having ASC 252 incorporated.

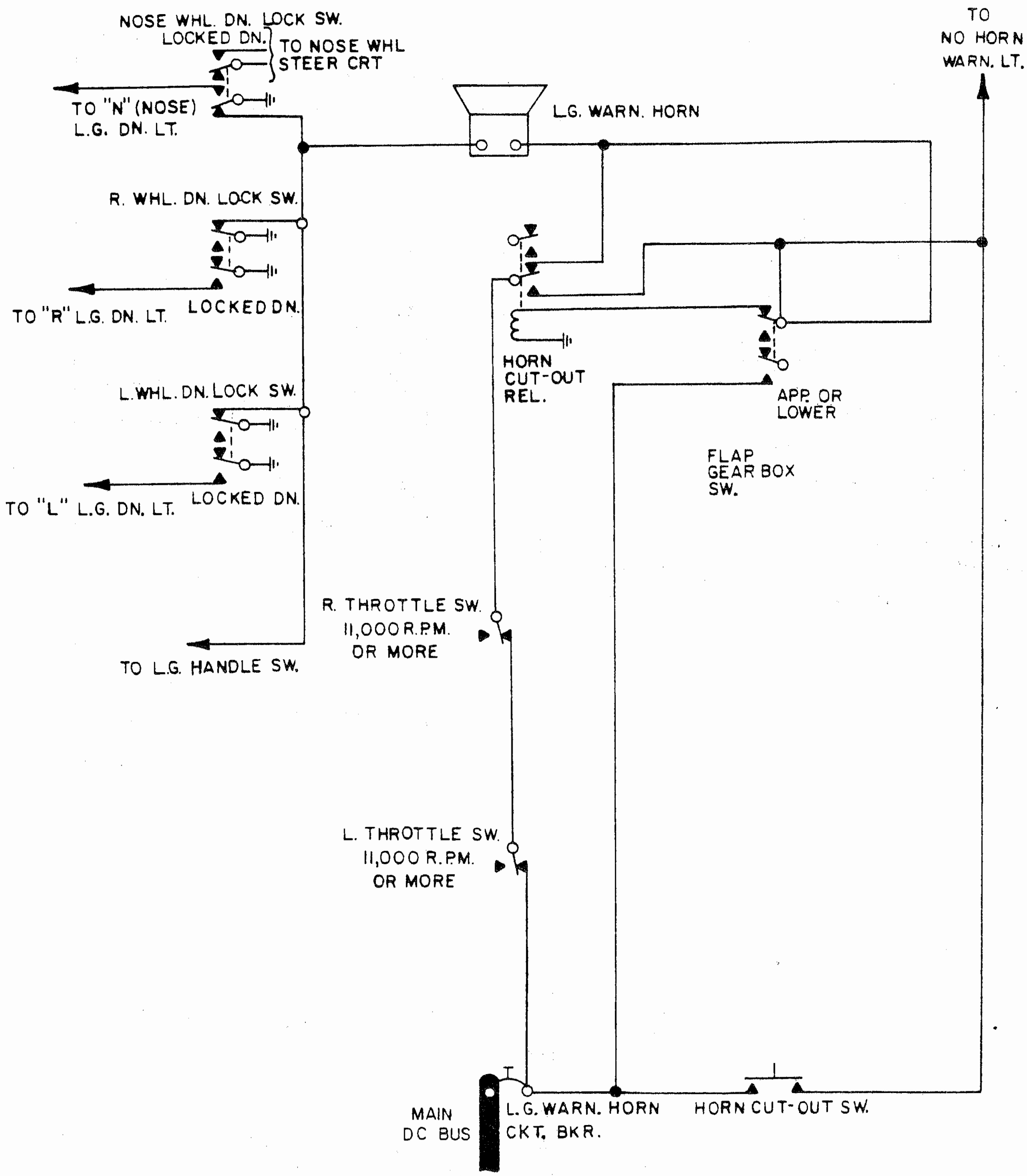
Flap Control System Circuit — Functional Block Diagram
Aircraft having ASC 252
Figure 2.



Flap Asymmetry Switch and Indication Circuit — Schematic
Figure 3.



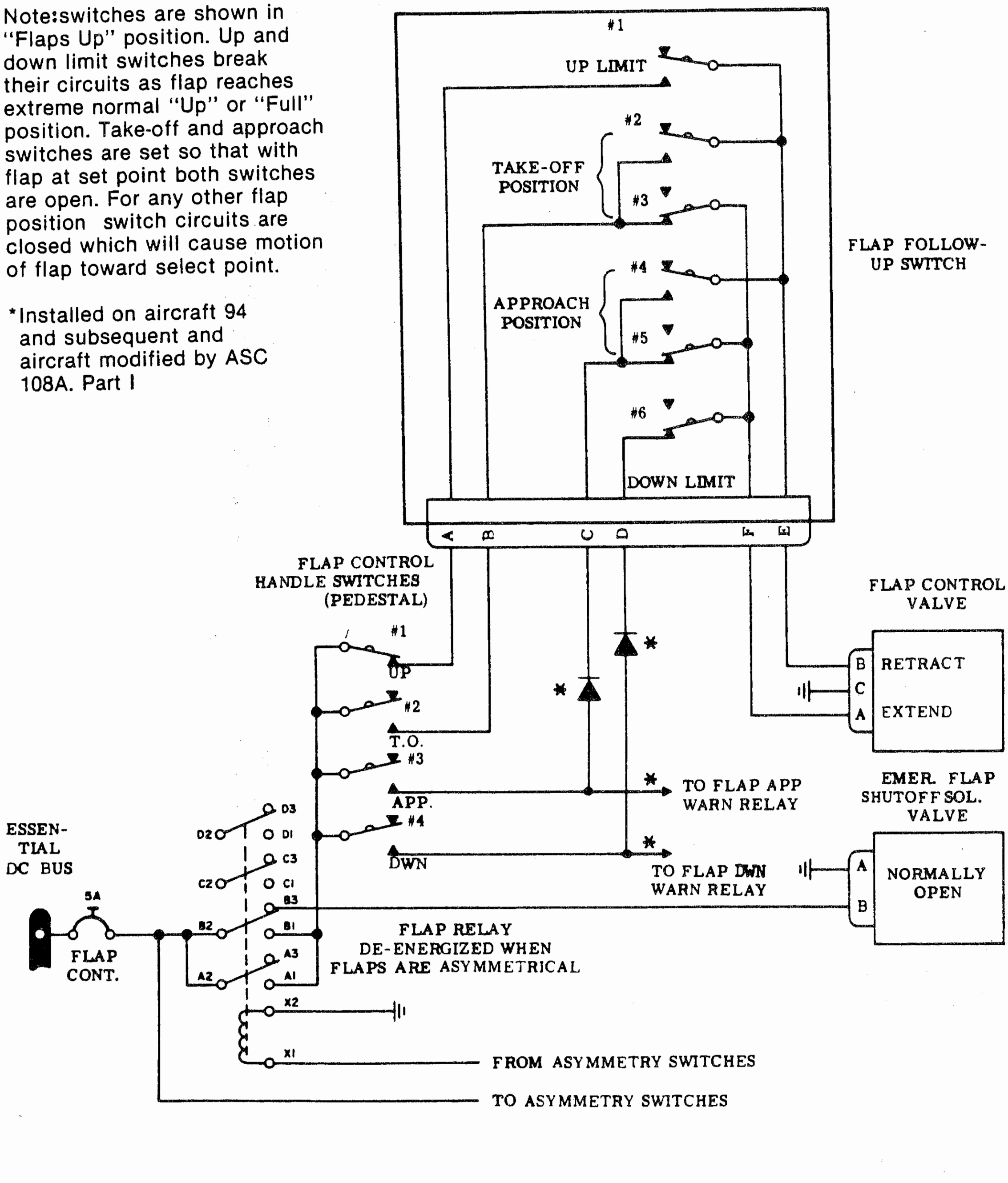
Flap System Alighting Gear Warning
Aircraft 2 - 93 and 114 having ASC 108A, Part I (only) and Aircraft 94 - 200, 322 and 323
Figure 4.



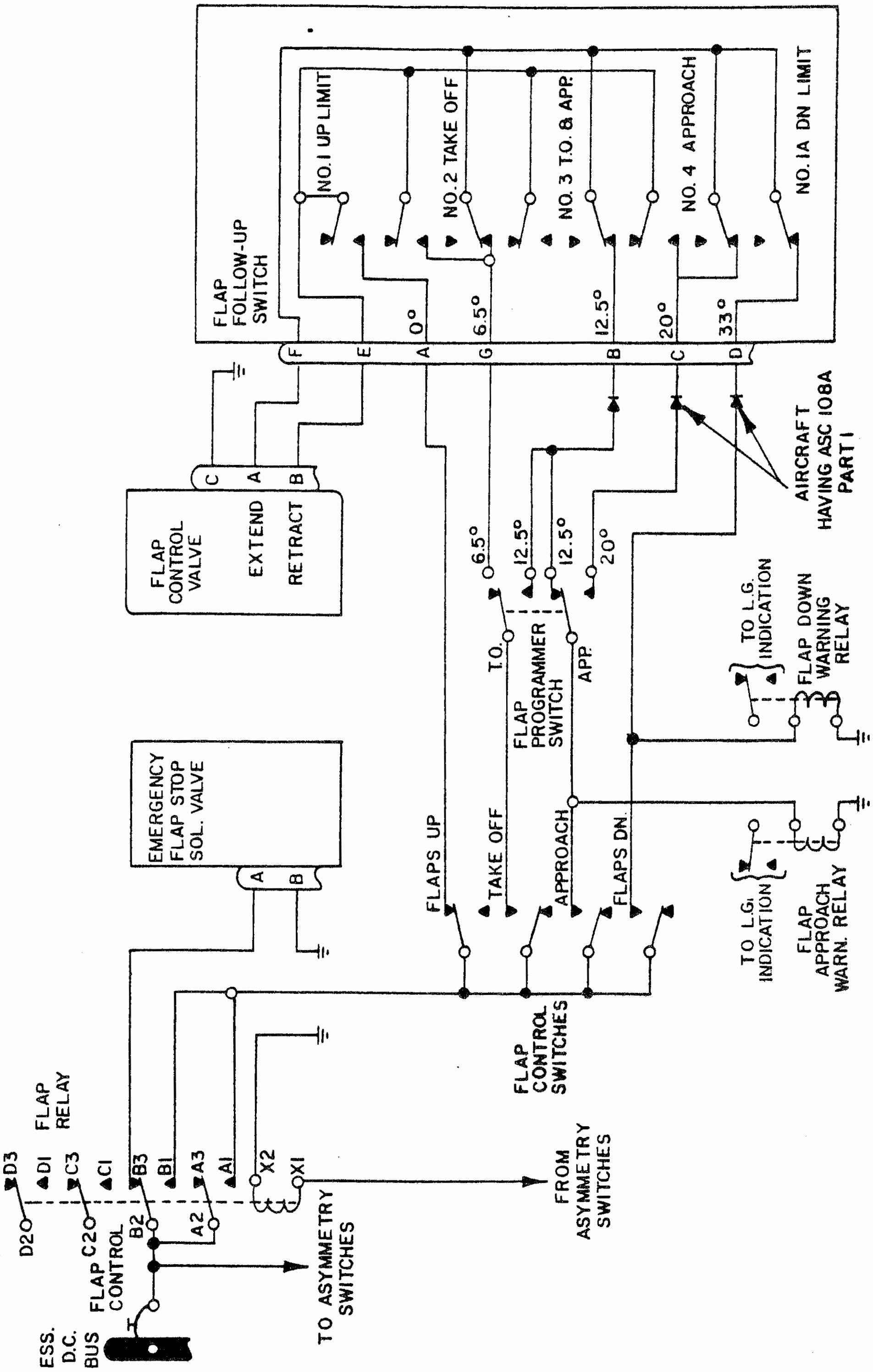
Flap System Alighting Gear Warning
Aircraft 2 - 93 and 114 having ASC 108A, Part II (only)
Figure 5.

Note:switches are shown in "Flaps Up" position. Up and down limit switches break their circuits as flap reaches extreme normal "Up" or "Full" position. Take-off and approach switches are set so that with flap at set point both switches are open. For any other flap position switch circuits are closed which will cause motion of flap toward select point.

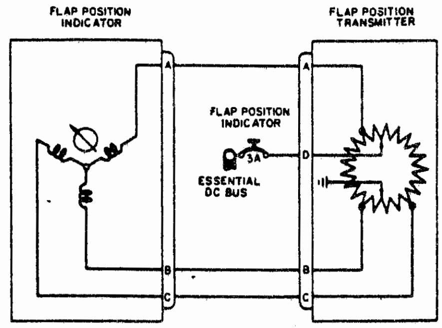
*Installed on aircraft 94 and subsequent and aircraft modified by ASC 108A. Part I



Flap Control System Circuit —Schematic
Figure 6.



Flap Control System Circuit —Schematic
Aircraft having ASC 252
Figure 7.



Flap Position Indicating Circuit —Schematic
Figure 8.

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WING FLAP SYSTEM — MAINTENANCE PRACTICES

1. **Flap — Removal / Installation**

WARNING: ENSURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR OF FLAPS BEFORE OPERATION.

A. Removal

(1) Apply electric power to aircraft.

(2) Ensure FLAP CONTROL and FLAP INDICATOR circuit breakers are engaged.

(3) Lower flaps to full down position.

(4) Depress trailing edge of flap and remove rollers from two middle carriages. (See Figure 204).

(5) Disconnect actuators by removing bolts securing adjustable link to flap.

(6) Unlock latches at inner carriages by pressing downlatch.

CAUTION: POSITION FOUR MEN TO SUPPORT FLAP WHEN UNLOCKING LATCHES TO FACILITATE REMOVAL OF FLAP.

(7) Unlock latches at outer carriages by pressing screwdriver under latch from rear.

CAUTION: SUPPORT TWO CENTER CARRIAGES DURING REMOVAL OF FLAP.

(8) Remove flap.

B. Installation

(1) Position flap on tracks and engage inner and outer carriage latches.

(2) Connect adjustable link of actuators to flaps with bolts.

(3) Install rollers on two center carriages. See Figure 201.

(4) Perform Flap System — Operational Test, this Section.

2. **Flap Drive Hydraulic Motor — Removal / Installation**

A. Removal

(1) Remove hydraulic and electrical power from aircraft.

(2) Remove FLR 15.

(3) Disconnect hydraulic lines, remove bolts securing hydraulic motor to central gearbox and remove motor.

B. Installation

(1) Apply light coating of grease on splines and install hydraulic motor on central gearbox with four bolts.

(2) Connect hydraulic lines to motor.

(3) Connect hydraulic and electrical power to aircraft.

(4) Operate motor, fully lower and raise flaps.

(5) Check for hydraulic fluid leaks.

(6) Inspect area for presents of foreign objects, security of all attachments and Install FLR 15.

3. Flap Central Gearbox — Removal / Installation

A. Removal

NOTE: Flap central gearbox assembly includes hydraulic motor, follow-up switch and flap position transmitter.

(1) Using EMERGENCY FLAP CONTROL HANDLE, lower flaps to full EMERGENCY DOWN position.

(2) Remove access cover on right side of fuselage just under leading edge of stabilizer, remove FLR 15.

(3) Install 1/4 inch rig pin in lockout cable crank in tail.

(4) Loosen lockout cable turnbuckles aft of Fuselage Station 640 in tail; separate cables from rudder lockout tab sheave on gearbox.

(5) Remove hydraulic and electrical power from aircraft.

NOTE: Before removing shafts, mark tubes to ensure original mating of bolt holes during installation.

(6) Remove longer half of telescopic drive shafts on both sides of gearbox, leaving short shafts attached.

(7) Remove electrical connectors from follow-up switch and flap position transmitter; remove hydraulic lines from hydraulic motor.

(8) Remove bolts and remove flap central gearbox assembly.

B. Installation

(1) If a new or replacement gearbox is to be installed, transfer follow-up switch and hydraulic motor to new gearbox; remove flap position transmitter, but do not install at this time.

(2) Position rudder lockout tab sheave so that drilled hole in sheave is located 9/16 inch from edge of guard. (See Figure 202)

(3) Reinstall short drive shafts.

(4) Install gearbox with bolts.

(5) Connect telescoping rods (right and left) being sure to keep flaps on emergency stops and rudder lockout tab sheave on 9/16 inch setting.

(6) Connect hydraulic lines and electrical connectors to gearbox.

(7) Install lockout cables on rudder lockout tab sheave, connect cables in tail compartment aft of Fuselage Station 640 and tension cables to 45 $\pm$ 5 pounds; lockwire turnbuckles and remove rig pin from rudder lockout tab crank in tail compartment.

NOTE: For each 10° above 70°F add 1.5 pounds to rigging load. For each 10° below 70°F subtract 1.5 lbs. from rigging load.

CAUTION: TO AVOID DAMAGE TO THE CONTROL VALVE, MOVE FLAPS, UTILIZING THE EMERGENCY SYSTEM, INTO NORMAL OPERATING RANGE PRIOR TO RESETTING DUMP VALVE.

(8) Manually reset emergency flap dump valve.

(9) Connect electrical and hydraulic power to aircraft.

(10) Place FLAP CONTROL handle to UP position.

(11) Remove covers on trailing edge of wing over left and right inboard screwjacks; check traveling nut is one full turn from forward stops.

NOTE: To obtain one full turn from forward stops, disconnect drive rod inboard of inboard actuators and make necessary adjustment. Both flaps must be as close as possible to the one full turn adjustment.

(12) Install flap position transmitter.

(13) Inspect area for presents of foreign objects, security of all attachments.

(14) Close wing and fuselage access cover and install FLR 15.

4. Flap follow-up Switch — Removal / Installation

A. Removal

(1) Remove FLR 15.

(2) Pull FLAP CONTROL and FLAP INDICATOR circuit breakers.

(3) Remove electrical connector from follow-up switch.

(4) Remove switch.

B. Installation

(1) Install follow-up switch with electrical receptacle below switch centerline.

(2) Install electrical connector and safety wire.

(3) Inspect area for presents of foreign objects, security of all attachments and Install FLR 15.

(4) Push in FLAP CONTROL and FLAP INDICATOR circuit breakers.

(5) Perform Flap System — Operational Test, this section.

5. Inboard Flap Actuator — Removal / Installation

A. Removal

(1) Connect electrical power to aircraft.

(2) With auxiliary hydraulic pump in operation, push EMERGENCY FLAP CONTROL handle down so that flaps extend to full EMERGENCY DOWN position. Shut off auxiliary hydraulic pump.

(3) Remove access cover above actuator on wing.

(4) Remove torque shafts on both sides of actuator.

(5) Remove T-fitting on aft end of actuator track.

(6) Disconnect flap drive link.

(7) Remove bolts from actuator head, remove lugs.

(8) Remove aft end of actuator (traveling link) from track. Screw traveling link to actuator head and remove actuator through access hole.

B. Installation

(1) Insert actuator through access hole and engage traveling link rollers in track.

(2) Install actuator with lugs, four bolts and safety.

(3) Install T-fitting with four bolts and connect flap drive link.

(4) Install torque shafts on both sides of actuators being careful that the down emergency stops on all the actuators are hitting. Align index holes on shafts to ensure proper alignment of bolt holes.

(5) Using auxiliary hydraulic pump, pull EMERGENCY FLAP control handle up to raise flaps. Then position lever in neutral position.

(6) Manually reset dump valve at Fuselage Station 169 bulkhead.

(7) Perform Flap System — Operational Test, this section.

6. Flap Outboard Actuator — Removal / Installation

A. Removal

(1) Apply electrical power to aircraft.

(2) With auxiliary hydraulic pump operating, push EMERGENCY FLAP CONTROL handle down to extend flaps to full EMERGENCY DOWN position, shut off auxiliary hydraulic pump.

(3) Remove torque shaft inboard of actuator.

(4) Remove electrical power from aircraft, then remove electrical connector from actuator.

(5) Disconnect flap drive link.

(6) Remove four bolts from actuator head, then remove lugs.

(7) Remove outboard side of actuator head mounting bracket, then remove actuator.

B. Installation

(1) Using either test light or asymmetry test box, attach leads to asymmetry switch terminals A and D.

(2) Place actuator on bench with head and traveling nut in same relation to each other as when installed on aircraft, then rotate spline on actuator head in extend direction until traveling nut engages bottom stop; light on test box should come on about 1/8 inch before contacting stop, and stay on after stop is engaged.

(3) If light fails to come on, rotate traveling nut one full turn from stop; repeat procedure as many times as necessary to obtain proper setting.

(4) Install actuator with lugs, bolts and safety.

(5) Install outboard side of mounting bracket and attach flap drive link.

CAUTION: WHEN INSTALLING TUBES, ENSURE ALL ACTUATOR DOWN STOPS ARE MAKING CONTACT.

(6) Install inboard torque shaft, being careful to index holes in torque tube with index holes in telescopic tube; this will ensure proper alignment of bolt holes.

(7) Using auxiliary hydraulic pump, pull EMERGENCY FLAP control handle up to raise flaps, then place lever in NEUTRAL position.

(8) Manually reset dump valve at Fuselage Station 169 bulkhead.

(9) Perform Flap System — Operational Test this Section.

7. Flap Actuator Yoke — Inspection

NOTE: This inspection applies to hollow yokes only - not required if aircraft has solid yokes.

A. Inspect flap actuator yoke assembly as follows:

NOTE: Ensure only one actuator yoke assembly is disconnected from a flap at one time. If following procedure is carefully followed, no rigging of flaps will be required.

WARNING: ENSURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR PRIOR TO OPERATING FLAPS.

(1) Apply electrical and hydraulic power to aircraft.

(2) Using normal flap control place the flaps full down.

(3) Remove electrical and hydraulic power from aircraft.

(4) Remove flap door springs from yoke assembly flap attachment bolt. (See Figure 203)

(5) Remove cotter pin, nut, bushings, washer, and bolt attaching yoke assembly to flap.

NOTE: Prior to removing yoke assembly rod end, mark or count number of threads showing on rod end so rod can be reinstalled to same position.

(6) Remove rod end lockwire, loosen jam nut and remove locking key.

NOTE: Locking key can be broken when being removed. It is recommended a new key or substitute key, P/N NAS559-4, be on hand before removal is attempted.

(7) Remove rod end from yoke assembly.

(8) Inspect yoke assembly inner tube for corrosion, using a 1/8 inch boroscope.

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP SOLVENT AWAY FROM OPEN FLAME.

(9) If no corrosion is found, clean inner part of tube with cleaning solvent. Apply a corrosion preventive compound (MIL-C-16173 Grade-1-Paralketone) to inner part of tube.

(10) If corrosion is present that is more than just surface corrosion, (i.e. that which cannot be removed with 3 or 4 passes with aluminum oxide paper), parts should be replaced within 150 hours or 3 months. If part is severely corroded, (i.e. a rough pitted surface as seen by a boroscope), and extends more than half way around the circumference then that part should be replaced within 50 hours or 1 month.

NOTE: It is possible a part may be corroded badly enough to be on point of failure. The inspector will have to make a judgement, based on visual appearance of part, as to whether corrosion is severe enough to warrant replacement before further flight. Gulfstream Aerospace Engineering Department will be able to give advice if they can be supplied with measurements of thickness of material remaining and the area of damage. However, the only way to get such measurements is by sophisticated ultrasonic techniques, using trained personnel, a high grade ultrasonic unit and special probes to fit tube.

(11) Reassemble flap actuator yoke assembly. Install new Safety wire and cotter pin.

NOTE: Ensure yoke rod end is adjusted to original position.

(12) Replace flap door springs.

WARNING: ENSURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR PRIOR TO OPERATING FLAPS.

(13) Apply electrical and hydraulic power.

(14) Perform Flap System — Operational Test this Section.

(15) Remove electrical and hydraulic power.

8. Flap Position Transmitter — Removal / Installation

A. Removal

(1) Gain access to flap position transmitter, located under FLR 15 Transmitter is mounted on flap central gearbox.

(2) Remove electrical connector from gearbox.

(3) Remove screws that secure transmitter to gearbox.

(4) Withdraw transmitter and keyed beveled gear from gearbox.

B. Installation

NOTE: If replacement transmitter is installed transfer beveled gear from old transmitter to replacement unit.

(1) Connect electrical connector to transmitter.

(2) Ground case of transmitter to structure using jumper.

(3) Energize main and essential dc buses.

(4) Using internal or external hydraulic power move flaps to normal up position.

(5) Slowly rotate transmitter shaft with beveled gear until flap position transmitter indicates UP.

(6) Without moving transmitter shaft, carefully mate transmitter to gearbox and secure with screws; remove jumper.

NOTE: Before performing Step (7) below on Aircraft having ASC 252, ensure the FLAP PROGRAMMER switch is in the 12.5° T/O 20° APP position.

(7) Operate flaps through T/O, APP and DOWN using normal flap control and check indications on flap position indicator.

NOTE: Step (8) below is to be performed on Aircraft having ASC 252.

(8) Select FLAP PROGRAMMER switch to 6.5° T/O, 12.5° APP, move FLAP CONTROL handle to TAKEOFF postion, flap indicator should indicate 6.5°. Move FLAP CONTROL handle to APPROACH position, flap indicator should indicate 12.5°. Move FLAP CONTROL handle to FULL position, flap indicator should indicate FULL.

9. Flap Control Valve — Removal / Installation

A. Removal

NOTE: Valves are on aft side of Fuselage Station 169 bulkhead in area of hydraulic reservoir.

(1) Remove electrical and hydraulic power.

(2) Remove electrical connector from control valve.

(3) Remove hydraulic lines and mounting bolts and remove valve.

B. Installation

(1) Position valve on structure with mounting bolts.

(2) Install hydraulic lines and electrical connector on valve.

(3) Energize main and essential dc buses.

(4) Check operation of flaps through each of the following positions:

NOTE: Step (a) below covers Aircraft not having ASC 252. Step (b) below covers Aircraft having ASC 252.

(a)

Flap Position	Indication
UP	UP
TAKEOFF	12.5°
APPROACH	20°
FULL	DN

(b)

Flap Position	Programmer Position	Indication
UP	N/A	UP
TAKEOFF	T/O 6.5°	6.5°
APPROACH	APP 12.5°	12.5°
TAKEOFF	T/O 12.5°	12.5°
APPROACH	APP 20°	20°
FULL	N/A	DN

(5) Check for hydraulic fluid leakage.

10. Flap Emergency Shutoff Valve — Removal / Installation

A. Removal

NOTE: Flap emergency shutoff valve is located in right side of nose wheel well.

(1) Ensure electric and hydraulic power is removed from aircraft.

(2) Disconnect electrical connector and hydraulic lines from valve.

(3) Remove valve.

B. Installation

(1) Install valve and connect hydraulic lines and electrical connector.

(2) Energize essential dc bus.

(3) Apply hydraulic power to aircraft.

(4) Place FLAPS handle to FULL position.

(5) When pointer on flap position indicator indicates DN pull (open) FLAP CONTROL circuit breaker, pilots circuit breaker panel.

(6) Disconnect asymmetry switch connectors at each outboard flap actuator (screwjack).

(7) Push in (close) FLAP CONTROL circuit breaker.

(8) Move FLAPS handle through following positions: APPROACH, TAKE OFF and UP. Flaps should not move.

(9) Move FLAPS handle back to FULL position.

(10) Pull (open) FLAP CONTROL circuit breaker.

(11) Connect asymmetry switch connectors.

(12) Push in (close) FLAP CONTROL circuit breaker.

(13) Place FLAPS handle in following detents. APPROACH, TAKEOFF and UP position. Flaps should operate normally.

(14) De-energize essential dc bus and remove hydraulic power.

11. Up and Down Flap Shuttle Valve — Removal / Installation

A. Removal

(1) Ensure electrical and hydraulic power is off.

(2) Remove FLR 15.

NOTE: Shuttle valves are located aft of the flap central gearbox.

(3) Remove shuttle valve.

B. Installation

(1) Install shuttle valve.

(2) Apply electrical and hydraulic power to aircraft.

(3) Check operation of shuttle valve using normal flap system.

CAUTION: WHEN USING THE EMERGENCY FLAP HANDLE, FLAPS CAN GO BEYOND THE NORMAL "UP" AND "FULL" POSITIONS. BEFORE RETURNING THE SYSTEM TO NORMAL OPERATIONAL, ENSURE FLAPS HAVE NOT BEEN STOPPED ABOVE THE NORMAL "UP" OR BELOW THE NORMAL "FULL".

(4) Operate emergency flap system. After system is checked set emergency flap handle to neutral position.

(5) Manually reset flap dump valve on Fuselage Station 169 bulkhead.

(6) Check for hydraulic leakage.

(7) Inspect area for presents of foreign objects, security of all attachments.

(8) Install FLR 15.

12. Drive Shaft — Removal / Installation

See Figure 203

A. Removal

(1) All sections of the drive shaft telescope on one end with the removal of two through bolts. When shortened, the ends may then be withdrawn from the splines of the adjacent shaft.

B. Installation

(1) Run actuators to forward stops.

(2) Install shafts between inboard and outboard actuators, pinning the drilled universal joints to drilled splines with AN525 screws.

(3) Turn shafts until inboard actuators are exactly one turn from forward stop.

(4) Install remainder of shaft.

13. Seals — Removal / Installation

A. Removal

(1) Remove the first section of drive shaft.

(2) Unbolt bracket containing seal from fuselage and remove. Inner stud of shaft will withdraw from shaft inside fuselage.

B. Installation

(1) Replace seal and install inner stud into shaft inside fuselage.

(2) Attach bracket to fuselage and connect drive shaft.

14. Sealing Nuts on Flap Track Rollers — Installation

A. Prepare aircraft for safe ground maintenance.

B. Remove and retain flaps.

NOTE: Description of installation will be detailed for one typical flap track roller nut replacement. Remainder will be presented in tabular form.

C. Locate carriage assemblies on flaps (See Figure 204) and replace existing nuts on flap track rollers as follows:

(1) Remove and discard castellated nut and cotter pin securing roller to carriage assembly.

(2) Remove any paint that may be on threaded portion of track roller stud. (See Figure 206, view A.)

(3) Clean track roller stud threads and cotter pin hole thoroughly of all grease with cleaning solvent.

(4) Install new washers on track roller stud. (See Figure 206, view B.)

(5) Mark end of roller stud to show position of cotter pin hole in stud. (See Figure 206, view C.)

(6) Apply epoxy primer to threads of track roller stud from cotter pin hole to face of washer.

CAUTION: DO NOT GET ANY PRIMER IN COTTER PIN HOLE OF ROLLER STUD.

(7) Apply epoxy primer to threads of new nut.

(8) Install new nut, while epoxy primer is still wet, on roller stud.

(9) Use penciled mark as guide to locate cotter pin hole in roller stud.

(10) Torque nut to standard torque.

(11) Using an awl or sharp needle-like object, (approximately 1/16 inch in diameter) carefully clean out cotter pin hole in stud.

NOTE: Hole should be centered and be perpendicular to its face.

(12) Press fit, by hand, appropriate size cotter pin. (See Figure 206, view D.)

(13) Grease each flap track roller. Use low temperature grease MIL-G-23827.

NOTE: Allow 1/2 hour setting time for epoxy primer prior to application of any grease.

(14) Reinstall flaps and check out installation in accordance with Paragraph 1, this section.

(15) Perform Flap System -- Operational Test this Section.

15. Flap System — Operational Test

WARNING: ENSURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR OF FLAPS BEFORE OPERATION.

A. Normal System Operational Test:

(1) Connect hydraulic and electrical power to aircraft.

(2) Close FLAP CONTROL and FLAP INDICATOR circuit breakers.

(3) Move flap control lever from UP position to TAKEOFF, APPROACH and DOWN; ensure flaps stop at the positions shown on indicator.

NOTE: Select each position in its proper detent and allow flaps to stop at the selected position before going to the next detent.

(4) Check flap drive rods and screwjacks for proper operation.

(5) Repeat flap operation from DOWN to UP through each detent; ensure flaps stop at positions shown on indicator.

B. Flap Emergency Shutoff Valve Check:

(1) Operate flaps using normal control to DOWN.

(2) Pull FLAP CONTROL circuit breaker (pilots circuit breaker panel).

(3) Disconnect asymmetry switch plug at either outboard flap screwjack.

(4) Push in FLAP CONTROL circuit breaker.

(5) Attempt to move flaps up using normal flap control; flaps should not move.

(6) Pull FLAP CONTROL circuit breaker.

(7) Reconnect asymmetry switch plug disconnected previously.

(8) Push in FLAP CONTROL circuit breaker.

(9) Operate flaps to UP using normal control; flaps should operate normally.

C. Emergency System Operational Check:

CAUTION: WHEN USING THE EMERGENCY FLAP HANDLE, FLAPS CAN GO BEYOND THE NORMAL UP AND FULL POSITIONS. BEFORE RETURNING THE SYSTEM TO NORMAL OPERATION, ENSURE THAT THE FLAPS HAVE NOT BEEN STOPPED ABOVE THE NORMAL UP OR BELOW THE NORMAL FULL POSITION.

(1) With flaps in UP position, pull FLAP CONTROL circuit breaker.

(2) Push EMERGENCY FLAP CONTROL lever to FLAPS DOWN and operate auxiliary hydraulic pump; flaps should extend to full EMERGENCY DOWN, contacting emergency stops.

(3) Pull EMERGENCY FLAP CONTROL lever to FLAPS UP position and operate auxiliary hydraulic pump, flaps should retract to UP position as shown on indicator.

(4) Place EMERGENCY FLAP lever in NEUTRAL position.

(5) Reset flap dump valve located on bulkhead 160.

NOTE: Aircraft 2 - 6 have two dump valves, each with reset button.

(6) Close FLAP CONTROL circuit breaker.

(7) Operate normal flap system through one complete cycle, and recheck for proper operation.

(8) Remove hydraulic and electrical power.

16. Flap Deflection Crank — Inspection

A. Apply electrical power to aircraft.

B. Ensure FLAP CONTROL and FLAP INDICATOR circuit breakers are engaged.

C. Lower flaps to full down. Remove electrical power.

D. Depress trailing edge of flap and remove rollers and attaching hardware from the two middle carriages (one at a time) at Wing Station 133 and 205. (See Figure 205, view B.)

E. Inspect roller and flap deflection cranks for following:

(1) Check the rollers for ease of operation and flat spots.

(2) Clean lugs on the fitting that holds the roller and check for cracks and/or corrosion pits with dye penetrant or equivalent.

(3) Check for wear on inside face of lugs. Maximum allowable wear is 0.025 inch on inboard fitting and 0.04 inch on outboard fitting. Wear should be made up with shims so inner race of roller is properly gripped between lugs when bolt is tightened without bending lugs in together.

NOTE: Replace fittings if they are worn beyond the above mentioned tolerances or if they are

F. Before assembly refinish lugs with No. 76 (Alodine) and 2012 (Epoxy Primer).

NOTE: Before installing bolt, coat it with wet 2012 (Epoxy Primer).

G. Install roller and attaching hardware. Torque nut to 95-110 inch pounds.

H. Perform Flap System — Operational Test, this Section.

17. Flap System — Rigging

(See Figure 207)

A. With inboard actuator flap drive link adjusted to 9.825 inches (using Gulfstream Aerospace 159GT1050-3), install inboard flap actuator.

B. With outboard actuator flap drive link adjusted to 8.1875 inches (using Gulfstream Aerospace No. 159GT1050-5), install outboard actuator. Do not disturb synchronization of the flap drive link with the asymmetry switch box.

(1) The asymmetry switchbox will be synchronized with outboard actuator flap drive link when following conditions are fulfilled:

(a) Actuator traveling nut is in contact with aft stop.

(b) Actuator traveling nut guide rollers center line is parallel to actuator forward mounting trunnion centerline.

(c) Actuator flap drive link is in position it will assume after installation. (Underside of actuator screw shaft).

(d) Test lamp installed across asymmetry switchbox terminals A and D indicates circuit is just closing.

(2) A direct method (to be performed on bench) of achieving conditions listed in B.(1), above, is as follows:

(a) With asymmetry switchbox attached to actuator, drive actuator gearbox (in a direction to extend traveling nut) until test lamp installed across asymmetry switch terminals A and D indicates circuit is just closing.

(b) Spin traveling nut, independently, until it contacts rear stop; note whether conditions listed in B.(1) above, are fulfilled.

(c) If conditions listed in B.(1) above are not fulfilled, spin traveling nut, independently, away from rear stop and repeat B.(2) as many times as necessary.

C. Install pillow block and drive shafting between inboard and outboard actuators while traveling nuts are in contact with forward stops.

D. Rotate drive shafting until inboard actuator traveling nut is driven exactly one turn away from forward stop. (Outboard actuator traveling nut will be less than one turn away from forward stop.)

E. Install central gearbox assembly. (This assembly consists of gearbox, flap follow-up switch, flap position transmitter and hydraulic motor.

   (1) Flap follow-up switchbox should be installed with electrical receptacle below switch centerline.

   (2) Flap position transmitter is indexed as described in Step H., below.

   (3) Hydraulic motor should be installed with name plate down.

F. Connect hydraulic motor and flap follow-up switch to cockpit controls. Connect asymmetry switches.

G. Operate central gearbox by means of cockpit pedestal FLAP position selector to flaps UP position. While actuators are positioned as described in Step D., above, install remaining pillow blocks and shafting.

H. With flap drive system positioned as described in Step G., above, synchronize flap position transmitter to the cockpit flap position indicator as follows:

   (1) Make electrical connection between transmitter and indicator.

   (2) Remove transmitter and bevel gear housing as a unit.

   (3) Rotate bevel gear until indicator needle is on UP mark and re-engage bevel gears.

   (4) If needle consistently falls to either side of mark, remove transmitter and bevel gear housing as a unit. As a unit, rotate transmitter and bevel gear housing 1/4 turn and repeat Step H. (3) above.

I. Perform intermediate check-out as follows:

   (1) Using auxiliary hydraulic pump and emergency flap control handle, extend actuators and check for simultaneous contact of mechanical stops.

   (2) Turn auxiliary hydraulic pump OFF and return emergency control handle to the neutral position with flap actuators against the aft stops. Connect test lamps across terminals A and D of left and right wing asymmetry switchboxes. Both test lamps should be ON. Rotate drive shafting by hand and check that test lamps go out within 1/4 turn of drive shafting of each other.

   (3) Using normal operation, retract actuators and check that inboard actuator traveling nut is one turn away from mechanical stops.

J. Using normal operation, extend actuators and install left hand flap assembly with supporting tracks and connect to actuators.

K. Retract flap until inboard actuator traveling nut is one turn away from mechanical stop and with flap upper surface in contact with wing slot lip, adjust length of flap mid span drag links to bring cam followers in contact with cams.

L. Using normal operation select flaps to full extend position and check cam followers for equal contact with cams. If one cam follower is taking all the hinge movement exerted by bungees, lengthen drag link connected to that cam follower and check again for equal contact.

M. Install right hand flap assembly with supporting tracks and connect to actuators. Repeat Steps K. and L above.

N. With flaps rigged according to above instructions, check that actuators make contact with mechanical stops.

O. Perform final checkout as follows:

   NOTE: To perform the following steps the main hydraulic system must be operative.

   (1) Move FLAP CONTROL handle to up detent, flaps should move forward until inboard actuators are one turn away from mechanical up stops. Flaps should then be in full up position.

   (2) Move FLAP CONTROL handle to TAKEOFF detent. Flaps should move to 12.5° ± 1°.

   (3) Move FLAP CONTROL handle to APPROACH detent. Flaps should move to 20° ± 1°.

   (4) Move FLAP CONTROL handle to FULL detent, flaps should move to 33° ± 1°.

NOTE: Steps (5) thru (7) below cover Aircraft having ASC 252.

(5) Move FLAP CONTROL handle to up detent. With flaps in full up position move FLAP PROGRAMMER switch to 6.5° T/O 12.5° APP, then move FLAP CONTROL handle to TAKEOFF detent. Flaps should move to 6.5° $\pm$ 1°, when 6.5° $\pm$ 1° is verified move FLAP CONTROL handle to APPROACH detent, flaps should move to 12.5° $\pm$ 1°.

(6) Move FLAP CONTROL handle to up detent and FLAP PROGRAMMER switch to center (OFF) position.

(7) With flaps in full up position, move FLAP PROGRAMMER switch to 12.5° T/O 20° APP. Perform flap check out as per Steps (2) and (3) above.

(8) In the up, TAKEOFF, APPROACH AND FULL positions, the maximum difference between left and right hand flaps shall be 1°.

P. If flap defections exceed tolerances specified in Step O., above, and a recheck indicates that Steps through M. were performed accurately, remove flap follow-up switchbox from central gearbox and check to see that switch box shaft rotates clockwise through the following angles from the UP position (up position is established when shaft index hole is opposite scribe mark on switch base plate) to the extended positions:

(1) UP to TAKEOFF - 79°(6.5°) - Aircraft having ASC 252. UP position - switchbox receptacle pins A and E circuit just opening. TAKEOFF - switchbox receptacle pins G and F circuit just opening.

(2) UP to TAKEOFF - 231.2 $\pm$ 1°(12.5°) - UP position - switchbox receptacle pins A and E circuit just opening. TAKEOFF position - switchbox receptacle pins B and F circuit just opening.

(3) UP to APPROACH - 245.8 $\pm$ 1°(20.°) - APPROACH position - switchbox receptacle pins C and F circuit just opening.

(4) UP to FULL - 266.7 $\pm$ 1° (33°) - FULL position - switchbox receptacle pins D and F circuit just opening.

Q. Minimum clearance between flap and aileron through full operating range of each shall be 5/16-inch.

18. Flap Carriage Assembly — Removal / Installation

A. Removal

(1) Remove flap (See Flap — Removal / Installation, this Section)

(2) Disconnect springs at carriage and remove hardware securing carriage to flap.

(3) Remove carriage.

WARNING: WITH BUNGEE ASSEMBLY REMOVED FROM FLAP CARRIAGE, ENSURE THAT NO PERSONNEL ARE IN FRONT OF BUNGEE DURING REMOVAL OF NAS 1202 BOLT. IF RIVETS (AN426-ADS) ARE SHEARED, THE SPRING WILL LAUNCH ITSELF WITH TREMENDOUS FORCE, INJURY TO PERSONNEL COULD RESULT. IF PROPER PRECAUTIONS ARE NOT TAKEN.

(4) Inspect carriage attachment fittings for elongated holes, distortion, cracks and security of mounting.

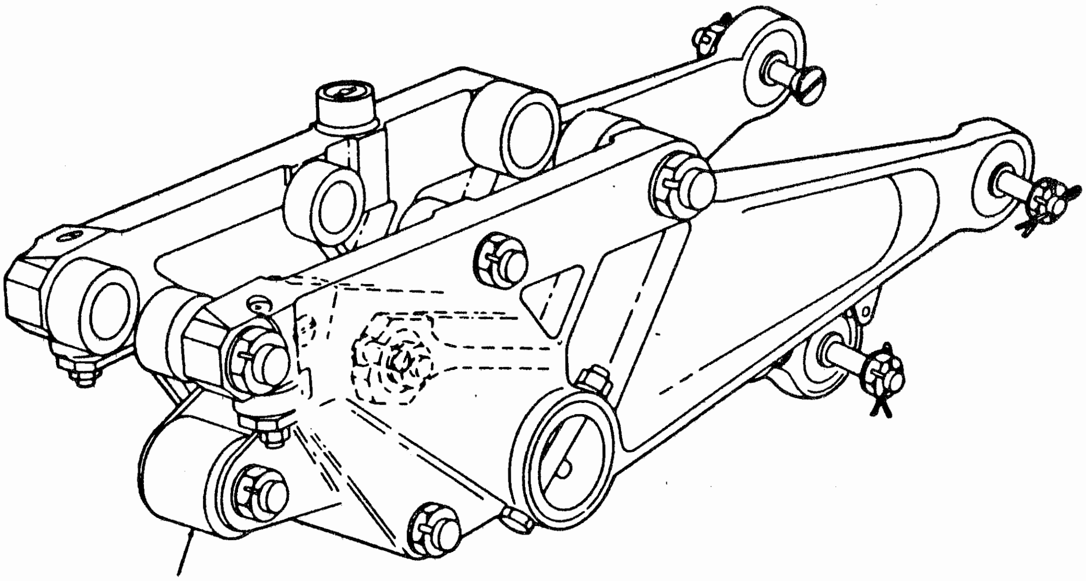
B. Installation

(1) Install carriage and secure with attaching hardware.

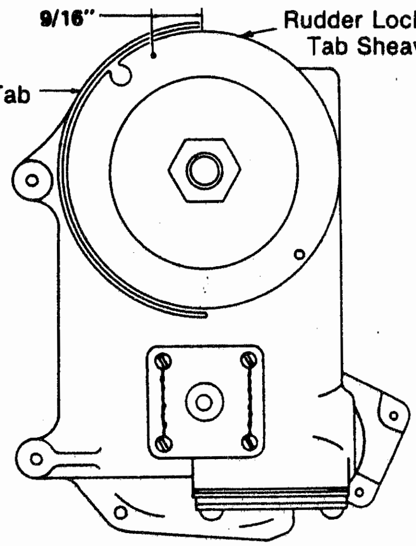
(2) Install springs at carriage.

(3) Install flap (See Flap — Removal / Installation, this Section)

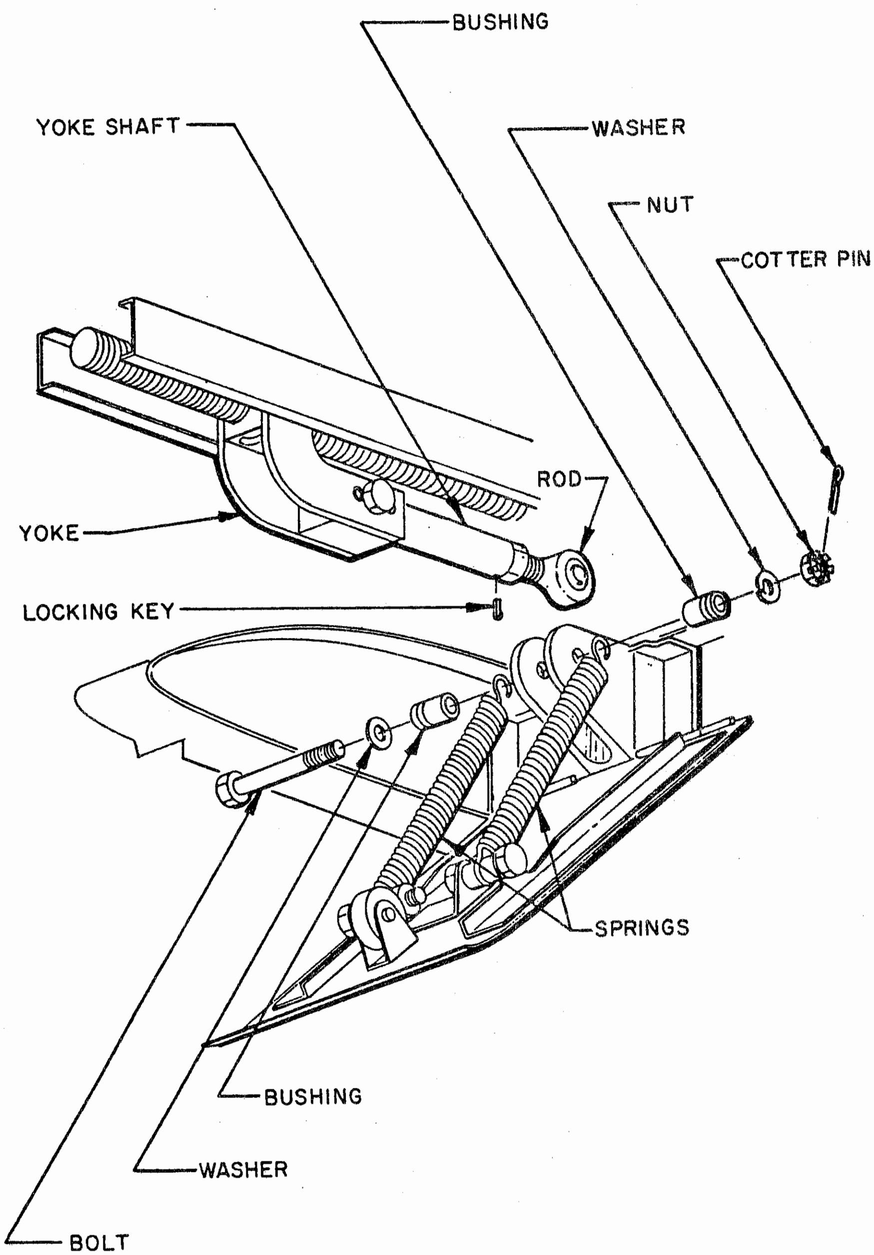
(4) Perform Flap System — Operational Test, this Section.



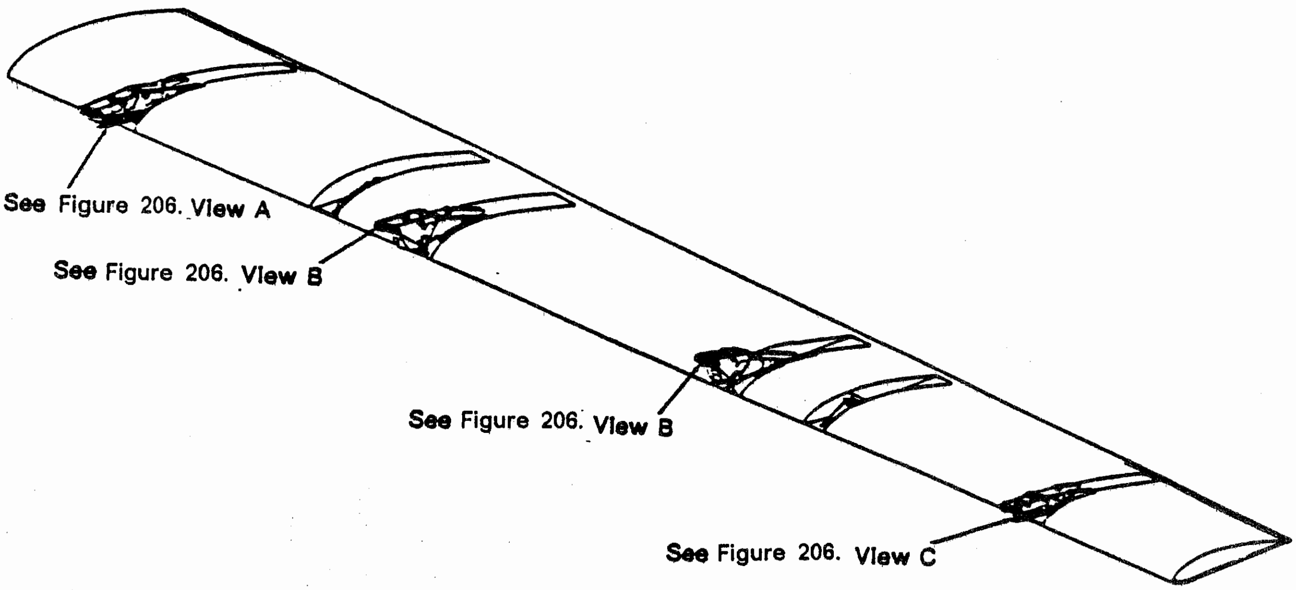
Roller Removal
Figure 201.



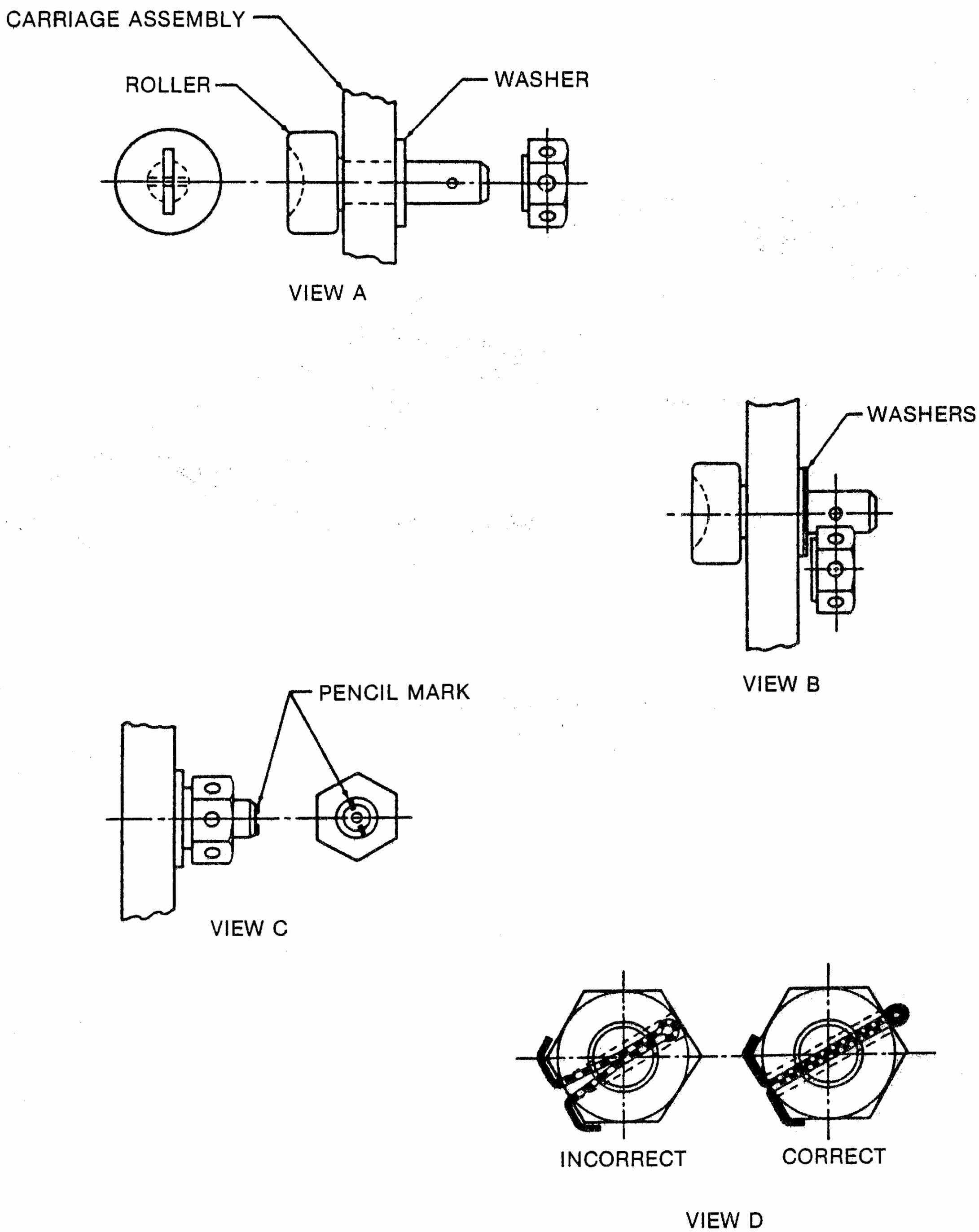
Central Gearbox (Side View)
Figure 202.



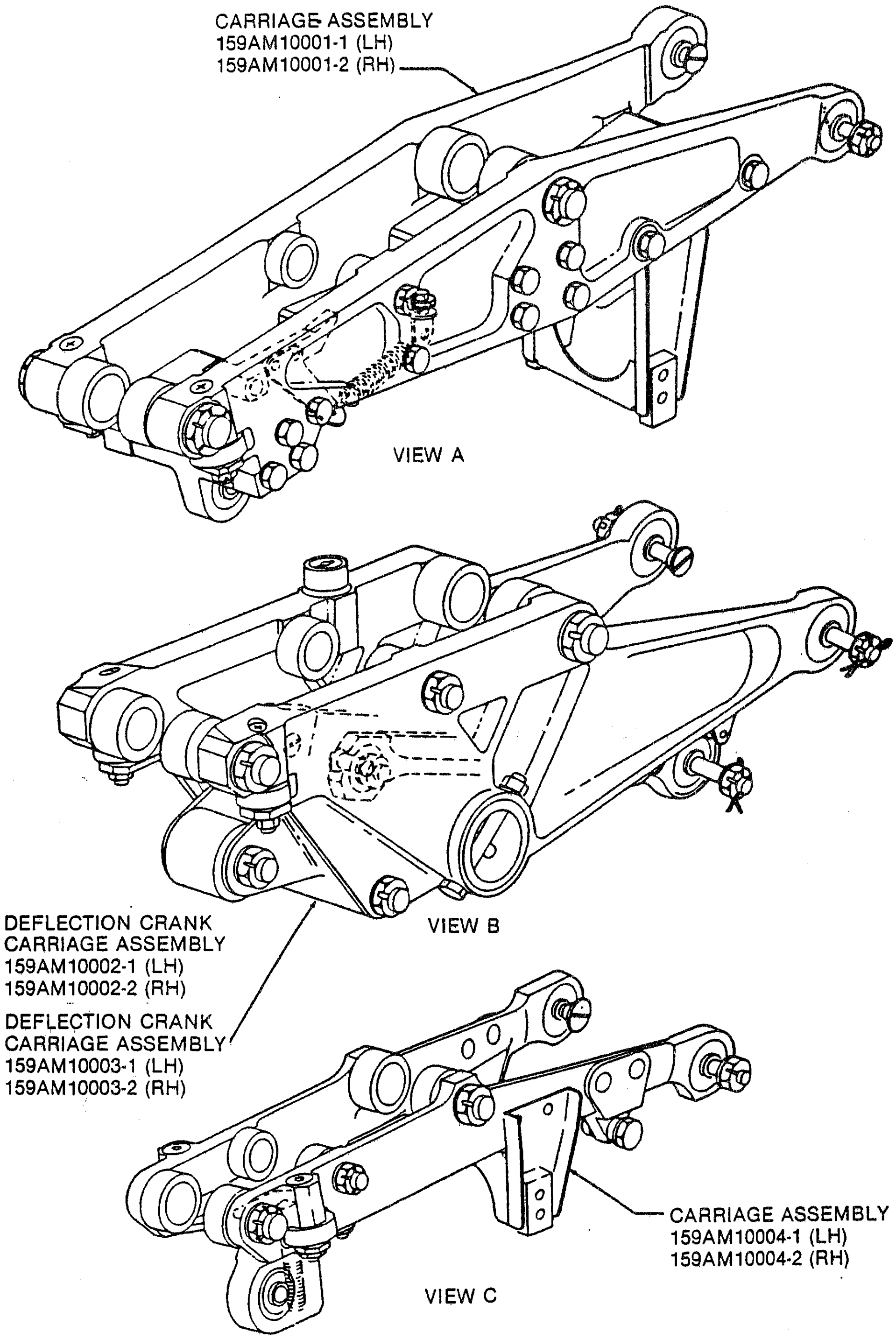
Flap Actuator Yoke Assembly
Figure 203.



Flaps and Carriage Assemblies — Location
Figure 204.



Sealing Nuts on Flap Track Roller — Detail — Installation
Figure 205.



Flaps Track Roller Nuts — Replacement
Figure 206.

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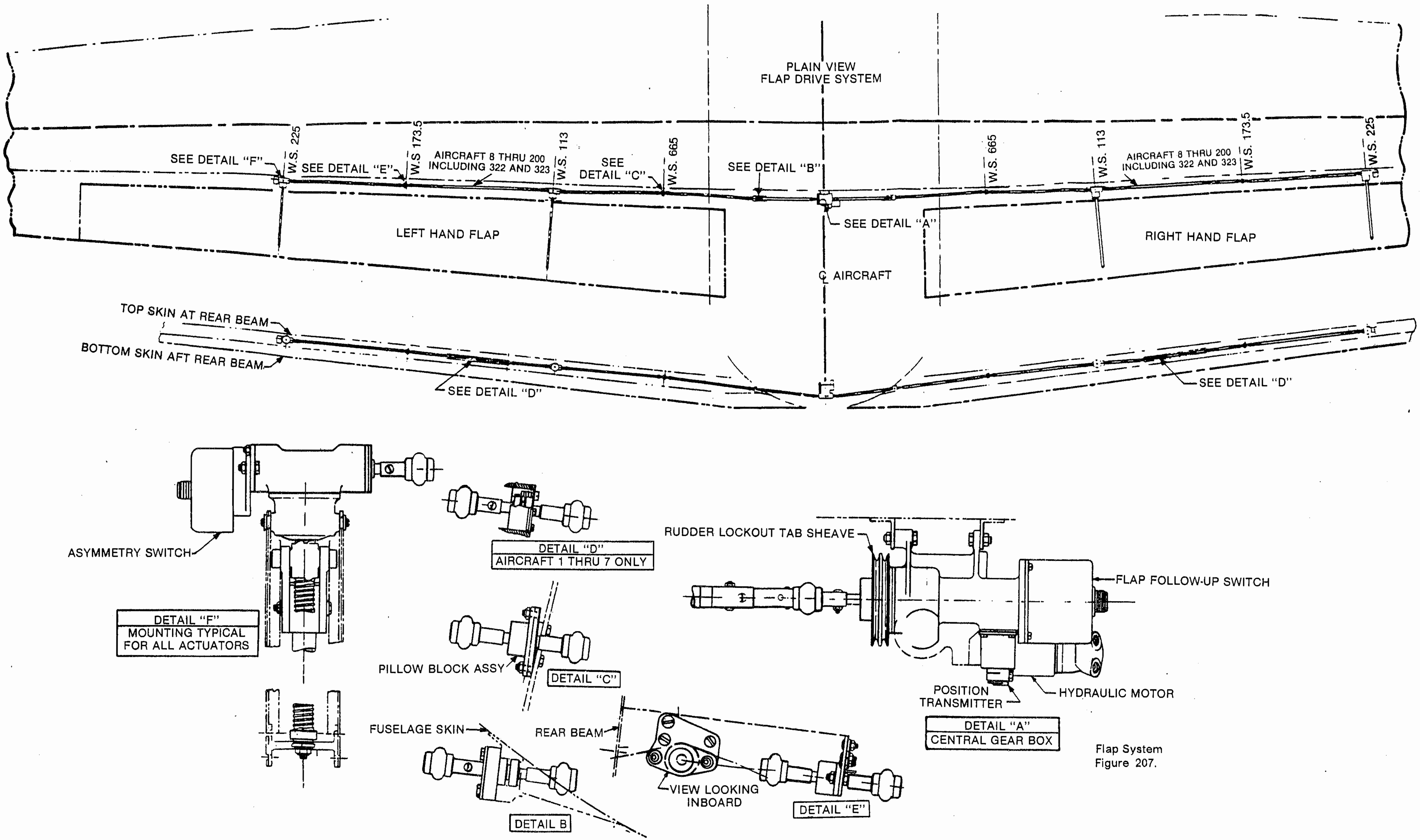


Flap System
Figure 207.

STALL WARNING SYSTEM — DESCRIPTION / OPERATION

1. Description

The electronic stall warning system gives warning of an impending stall by means of a control shaker mounted on the copilots control column. The system is designed to give accurate prestall warning regardless of flap settings, power settings, accelerations, and configurations by automatically compensating for these conditions.

The system consists of the following components:

Unit	No. Per A/C	Location
Lift Transducer	1	Leading edge, left wing
Flap Position Potentiometer	1	Tail Compartment
Lift Computer Box	1	Left Side of Fuselage Station 170
Control Shaker	1	Mounted Right Control Column Under Cockpit Floor
Test Switch	1	Copilots Console
Pitot Heat Switch	1	Upper Overhead Panel
STALL CONT - Circuit Breaker	1	Pilots Circuit Breaker Panel
STALL HTR - Circuit Breaker	1	Pilots Circuit Breaker Panel

2. Operation

(See Figure 1.)

The computer, after having received the information from the lift transducer and the flap potentiometer, decides whether a stall is forthcoming. If a stall is imminent, a signal is sent to the control column shaker. This shaker produces a low frequency, high amplitude buffet signal to the pilot.

The lift transducer is electrically heated to prevent icing when the PITOT HEAT switch on the Upper Overheated Panel is set to ON. To check the stall warning circuit, the STALL WARN TEST momentary switch, located on the copilots console, must be held to ON. The control shaker should then vibrate the control column. This will occur in flight or on the ground.

Power to the control column shaker is removed by nutcracker relay No. 2 when the airplane is on the ground. Use of the STALL WARN TEST switch overrides nutcracker relay operation.

3. Emergency DC Operation

- Stall warning system is operative in Emergency.
- Stall warning transducer heater is inoperative in Emergency.

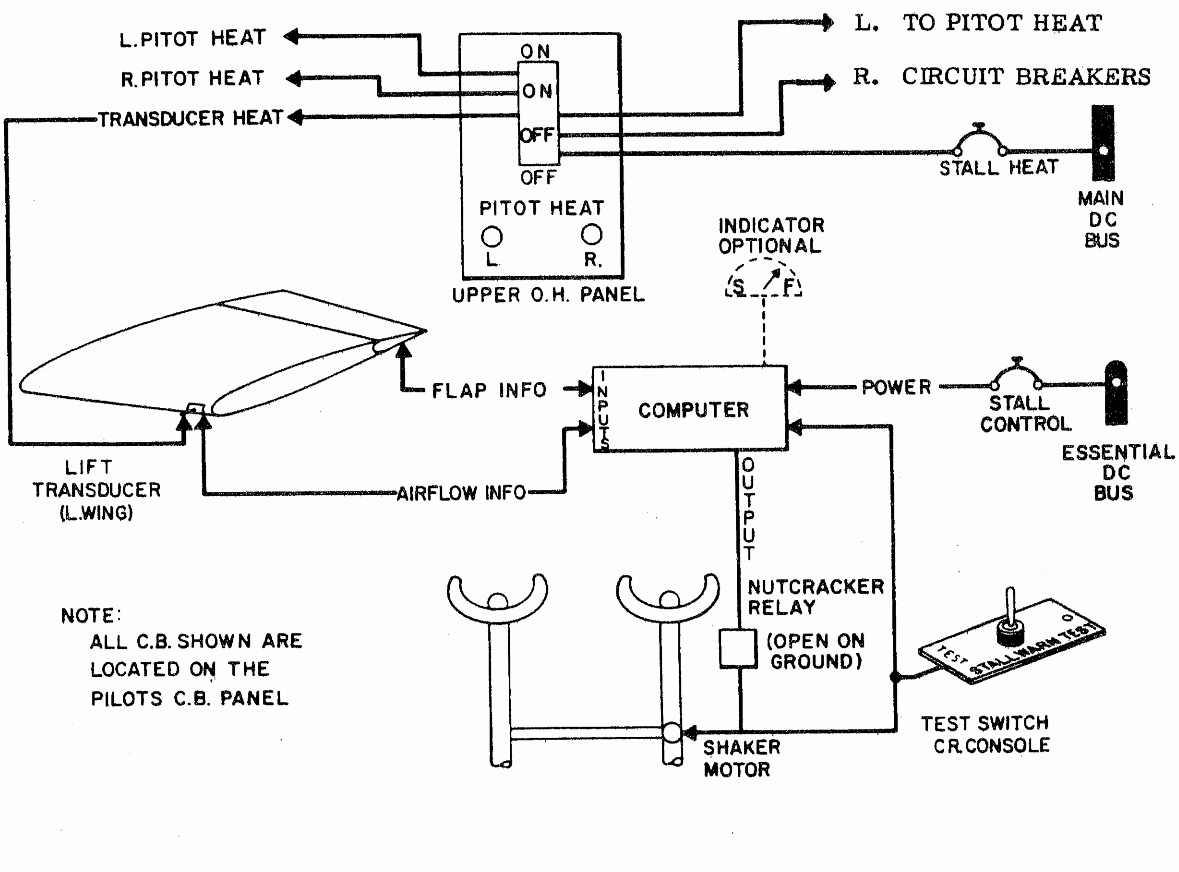
CAUTION: SINCE THE NUTCRACKER RELAYS REMAIN IN AIRBORNE CONFIGURATION WHEN IN EMERGENCY DC OPERATION, REGARDLESS OF NUTCRACKER SWITCH POSITION, THE STALL WARNING CONTROL COLUMN SHAKER MOTOR WILL NOT CUT OUT ON THE GROUND WITH POWER ON THE AIRCRAFT UNLESS THE AIRCRAFT HAS ASC 108A, PART I. THE NUTCRACKER RELAYS ARE OPERATIVE IN EMERGENCY DC CONDITION ON AIRCRAFT HAVING ASC 108A, PART I.

4. **Indicator Provisions**

The computer is capable of driving an indicator (not supplied by Gulfstream), however the four wires necessary to operate this indicator are wired into the computer disconnect plug and dead-ended at the stall warning terminal panel, located at the lower part of the Fuselage Sta. 193, aft side, below the radio and radar terminal panels, and inverter fuse panel. If it is desired to install this instrument in the cockpit, it is necessary to pick up the four terminals for the indicator provisions on the stall warning terminal panel and complete the wiring to the cockpit to a mating plug, and install the indicator. (See GULFSTREAM I Wiring Diagram Manual for terminal numbers.)

5. **Flap Position Potentiometer**

The flap position potentiometer is located at fuselage sta 697.5, adjacent to the rudder spring tab lock sector wheel. It is attached to a bracket with three AN500AC8-8 screws, and connected to the rudder spring tab lock sector wheel through mechanical linkage. An adjusting bolt is located on the lower end of the connector arm.



Stall Warning System — Functional Block Diagram
Figure 1.

STALL WARNING SYSTEM — MAINTENANCE PRACTICES

1. **Stall Warning Lift Transducer — Removal / Installation**

(See Figure 201)

WARNING: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. PULL (OPEN) STALL CONT AND STALL HTR CIRCUIT BREAKERS.

A. Removal

(1) Gain access to stall warning transducer, located in gap band area of left wing leading edge, by removing leading edge strap holding transducer.

(2) Disconnect electrical connection and remove transducer with strap.

NOTE: If difficulty is encountered in disconnecting the electrical connection it may be necessary to remove the mid leading edge section.

(3) If transducer is to be replaced, remove screws securing transducer to strap.

B. Installation

NOTE: If transducer is a replacement, install with gasket, on strap and secure with screws.

(1) Connect electrical connection at wing leading edge.

(2) Perform Stall Warning Lift Transducer — Operational Test, this Section.

(3) Inspect area for presents of foreign objects, security of all attachments.

(4) Install leading edge strap with transducer to wing leading edge.

2. **Stall Warning Lift Transducer — Operational Test**

A. Depress (close) the following circuit breakers: (Pilots circuit breaker panel.)

- STALL CONT
- STALL HTR

B. Pull (open) the following circuit breakers: (Pilots circuit breaker panel.)

- L PITOT HTR
- R PITOT HTR
- NUTCRACKER CONT (Aircraft 1 - 93 and 114 not having ASC 108A, part I, pull the LG DOWN LIGHTS and LG WARN LIGHTS circuit breakers.)

C. Energize main and essential dc buses.

D. Move vane on stall warning lift transducer upward slowly and verify that at some point the control column shaker operates.

E. Check stall warning heat system as follows:

CAUTION: DO NOT ALLOW HEAT TO BE APPLIED IN EXCESS OF 30 SECONDS.

(1) Place PITOT HEAT switch to ON.

NOTE: Transducer vane and plate should be warm to the touch within 30 seconds.

(2) Place PITOT HEAT switch to OFF.

F. Depress (close) circuit breakers that were pulled (opened) in Step 2.B. above.

G. De-energize main and essential dc buses and disconnect electrical power.

3. Stall Warning Shaker Motor — Removal / Installation

(See Figure 202)

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. PULL (OPEN) STALL CONT CIRCUIT BREAKER.

A. Removal

(1) Gain access to shaker by removing copilots seat.

(2) Remove boot and boot ring at base of copilots seat.

(3) Remove floor access plate directly aft of copilots control column.

(4) Disconnect two electrical leads from motor at terminal strip 60. (Leads are on terminals 7 and 8.)

(5) Remove clamp that secures electrical wire run to structure.

(6) Remove lockwire and remove motor-to-column attachment screws.

(7) Remove shaker motor.

B. Installation

(1) Install unit on control column and secure with screws. Lock wire screws.

(2) Route wires in same manner as removed and secure wire run with clamp.

(3) Connect wires from shaker motor to terminals 7 and 8 of terminal strip 60.

(4) Tighten terminal strip nuts and install terminal cover.

(5) Perform Stall Warning Shaker — Operational Test, this Section.

(6) Inspect area for presents of foreign objects, security of all attachments.

(7) Replace control column boot, boot ring and access cover previously removed.

(8) Install copilot seat.

4. Stall Warning Shaker — Operational Test

A. Depress (close) STALL CONT circuit breaker on pilots circuit breaker panel.

B. Energize main and essential dc buses.

C. Place STALL WARN TEST switch on copilots panel to TEST position and hold. Shaker should operate.

D. Release test switch. Shaker should stop.

E. De-energize main and essential dc buses.

5. Stall Warning Potentiometer — Removal / Installation

(See Figure 203)

WARNING: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING EQUIPMENT. PULL (OPEN) STALL CONT. CIRCUIT BREAKER.

A. Removal

(1) Ensure flaps are in the full up position.

(2) Gain access to stall warning potentiometer, located in aft section of tail compartment under rudder spring tab lockout mechanism.

(3) Disconnect electrical connector.

(4) Remove and retain nut, bolt and washers, from vertical arm connecting potentiometer to horizontal arm of sector wheel.

(5) Remove screws securing potentiometer to mount.

(6) Remove potentiometer.

NOTE: If replacement potentiometer is to be installed, remove vertical arm from potentiometer shaft by loosing screw at arm to shaft connection.

B. Installation

NOTE: If potentiometer is a replacement, install vertical arm loosely on shaft before installing in aircraft.

(1) Ensure flaps are in full up position.

(2) Install potentiometer and connect sector arm to vertical arm using hardware retain in Step 5.D. above.

(3) With shaft arm jam clamp loose, rotate shaft of potentiometer until indicator mark on potentiometer aligns with letter "A" on case.

(4) Carefully tighten shaft arm jam clamp screw. Ensure that indicator mark aligns with letter "A".

6. Stall Warning Potentiometer — Functional Test

A. Disconnect electrical connector from stall warning computer in entrance compartment above hydraulic reservoir.

B. Connect ohmmeter between pins K and L of electrical connector. Ohmmeter should indicate 0 to 30 ohms.

C. Provide a hydraulic power source (internal or external) to enable normal flap operation.

D. Energize main and essential dc buses and apply hydraulic power.

E. Lower flaps to full down position (33°) and observe ohmmeter. Indication on ohmmeter should increase to 2700 to 3000 ohms and hold until flaps are down.

F. Raise flaps using normal flap control handle. Indication on ohmmeter shall hold at 2700 to 3000 ohms until approximately takeoff position then gradually decrease to 0 to 3 ohms when flaps are up.

G. Remove ohmmeter.

H. Remove electrical and hydraulic power.

I. Connect electrical connector.

7. Stall Warning Lift Computer — Removal /Installation

A. Removal

(1) Gain access to computer in hydraulic compartment, on shelf directly above the hydraulic reservoir.

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING THE COMPUTER.

(2) Disconnect electrical plug from unit.

(3) Remove screws holding unit to shelf.

(4) Remove stall warning lift computer.

B. Installation

C. Install stall warning computer and mounting screws.

D. Install electrical plug.

E. Perform Stall Warning Lift Computer — Adjustment / Check, this Section.

8. Stall Warning Lift Computer — Adjustment / Check

A. Prior to flight, remove cover from computer by removing screws from bottom of unit corners. Holes are drilled in the shelf for screw access, and unit does not have to be removed from shelf.

B. Energize main and essential dc buses.

C. Check test circuit by holding STALL WARNING TEST switch (copilots console) to TEST position; shaker motor should operate.

D. Release TEST switch; shaker should stop.

E. Operate aircraft in level flight between 8000 and 13000 feet.

F. With aircraft in clean configuration (gear up - flaps up) and 100 pounds of torque on each engine, have crew slowly approach a stall, noting carefully the following:

(1) Indicated airspeed when stall warning actuates (V_w)

(2) Indicated airspeed at actual stall (V_s).

G. Recover from stall; if stall warning was 5 to 11 knots above actual stall speed, proceed to step L.; if not, adjustment of computer null potentiometer is required (proceed with steps H. through K.)

H. Have crew hold airspeed at 9 to 11 knots above actual stall speed with clean configuration and 100 pounds of torque on each engine.

I. Adjust computer null potentiometer by loosening 5/16 inch jam nut and slowly turn adjustment screw as follows:

(1) For early warning (above 11 knots) - clockwise.

(2) For late warning (below 5 knots) - counterclockwise.

J. When crew indicates that shaker is operating, lock jam nut.

K. Repeat steps F. through J. until warning occurs 5 to 11 knots above actual stall speed.

L. Check and record stall IAS and warning IAS in the following configurations:

(1) Gear up - flaps up - engines 100 pound torque each.

(2) Gear down - flaps down - engines 100 pound torque each.

NOTE: If adjusted properly, warning must come 5 to 11 knots before stall in both configurations.

M. Recheck STALL WARNING TEST switch for shaker operation in normal flight.

NOTE: Computers with serial numbers from 1 thru 400 will not operate in test mode above 140 KIAS. If computer is in this serial number block, slow aircraft to 140 KIAS or below and check TEST switch operation.

N. With aircraft on ground and parked, de-energize main and essential dc buses, replace cover on computer and secure with screws.

9. Stall Warning System — Functional Test

A. Disconnect cannon plug on the lift computer located on shelf in the hydraulic compartment.

NOTE: All checks will be made at the lift computer cannon plug on the aircraft wiring. No checks are made on the lift computer.

B. Close the STALL WARN circuit breaker.

C. Checkout power supply by checking for 28 volts dc across pin S and pin T.

D. Checkout shaker motor by connecting a jumper across pin S and pin P and moving the STALL WARN TEST switch to the TEST position. The shaker motor should vibrate the control columns.

E. Open the STALL WARN circuit breaker.

F. Checkout the flap potentiometer by operating the flaps to the UP position and checking resistance across pin Land pin K. Resistance should be from 0-3 ohms. Lower the flaps to T/O (12.50) position and check that the resistance increases to 2700-3000 ohms. Lower the flaps to FULL (330) position and check that the resistance does not change (2700-3000 ohms).

G. Checkout the lift transducer by checking resistance:

(1) Across pin A and pin T (excitation winding) 8-10 ohms.

(2) Across pin C and pin D (fast winding) - 300 $\pm$ 15 ohms.

(3) Across pin D and pin E (slow winding) - 300 $\pm$ 15 ohms.

GUST LOCK SYSTEM — MAINTENANCE PRACTICES

1. Gust Lock — Rigging

A. Special Tools Required

ITEM	DESCRIPTION
Tensiometer	Pacific Scientific 30 pounds to 300 pounds
Rigging Pins	1/4 inch diameter bolts, AN4 or equivalent

B. Rigging

(See Figure 201)

NOTE: Part numbers are used in this manual only as a means of identification when nomenclature alone is inadequate. This manual must not be used for identifying spare parts by number. Consult Gulfstream I Illustrated Parts Catalog for this information.

(1) Starting in right wing, install latch assembly 159CM10501 on aft side of rear wing beam.

(2) Attach cable assemblies 159C10522 to bellcrank assembly 159C10502, install and pin in locked position.

(3) Install springs G102-464.

(4) Install and adjust bungee assembly 159C10510 until latch assembly 159CM10501 bottoms on roller mounted in leading edge of aileron.

(5) Install crossover cable 159C10522.

(6) In left wing, install latch assembly 159CM10501 on aft side of rear wing beam.

(7) Attach cable assembly 159C10522 to top of crank assembly 159C10502, install and pin in locked position.

(8) Attach crossover cable assembly 159C10522 to lower end of crank assembly 159C10502.

(9) Install and adjust bungee assembly 159C10510 until latch assembly 159CM10501 bottoms on roller mounted in leading edge of aileron.

(10) Install sector assembly 159C10511 at Fuselage Station 278 and pin in locked position.

(11) Tension cables to the following tensions:

- 159C10522-5 (right wing - upper) 30 $\pm$ 5 pounds
- 159C10522-7 (left wing - upper) 88 pounds $\pm$ 10%
- 159C10522-9 (Crossover) 60 pounds $\pm$ 10%

NOTE: For each 10° above 70°F add 1.5 pounds to rigging load, for each 10° below 70°F subtract 1.5 pounds from rigging load.

(12) In fuselage, install latch assembly 159CM10521 at Fuselage Station 691.14.

(13) Install crank assembly 159C10518 at Fuselage Station 674.45 and pin in locked position.

(14) Install springs G102-49.

(15) Install and adjust bungee assembly 159C10517 until latch assembly 159CM10521 bottoms on roller mounted on elevator control bellcrank assembly.

(16) Install latch assembly 159CM10522 at Fuselage Station 678.75.

(17) Install crank assembly 159C10507 at Fuselage Station 672 and pin in locked position.

(18) Install and adjust bungee assembly 159C10506 until latch assembly bottoms on roller mounted on elevator control bellcrank assembly.

(19) Install and adjust pushrod assembly 159C10523 between crank assemblies 159C10507-1 and 159C10518.

(20) Install cable assemblies 159C10522 between crank assembly 159C10507 and sector assembly 159C10511 at Fuselage Station 278.

(21) Tension cables to the following tensions:

- 159C10522-1 (lock cable) 68 pounds ± 10%.
- 159C10522-3 (unlock cable) 40 pounds ± 5%

NOTE: For each 10° above 70°F add 1.5 pounds to rigging load, for each 10° below 70°F subtract 1.5 pounds from rigging load.

(22) Install cable assemblies 159C10508 between sector assembly 159C10511-1 at Fuselage Station 276.21 and cables from pedestal in cockpit. Gust lock handle on pedestal must be in aft or locked position.

(23) Tension cables to the following tensions:

- 159C10508-1 (unlock cable) 40 pounds ± 5%
- 159C10508-3 (lock cable) 68 pounds ± 10%

NOTE: For each 10° above 70°F add 1.5 pounds to rigging load, for each 10° below 70°F subtract 1.5 pounds from rigging load.

(24) Make any re-adjustments in system until all rigging pins are easily removed. Safety all turnbuckles and cable adjusters.

(25) Remove all rigging pins and check system for freedom of motion. Double check for locking action of cranks versus gust lock handle motion (handle lock - latches locked). While rigging system, install GS13F-3 cable pressure seals.

2. Gust Lock System — Operational Test

WARNING: ENSURE PERSONNEL AND EQUIPMENT ARE CLEAR OF ALL CONTROL SURFACES BEFORE PERFORMING FOLLOWING TEST.

A. Ensure flight fine pitch lock selector handle is in GROUND (aft) position; move power levers aft.

B. Operate gust lock control aft to CONTROL LOCKED position and neutralize controls.

C. Attempt to operate aileron, rudder and elevator controls; they should be locked and must not move.

D. Attempt to move both power levers forward: they should not move beyond 30°

E. Alternately push one power lever full ahead: remaining power lever should move below the 30° position.

F. Place gust lock control lever forward to CONTROLS UNLOCKED position.

G. Operate aileron, rudder and elevator controls: they should move freely through their full ranges.

H. Place flight fine pitch lock selector handle in FLIGHT position.

I. Attempt to move gust lock lever aft to CONTROLS LOCKED position, lever must not move.

J. Position all controls and levers to on ground normal configuration.

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FUELS

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CHAPTER 28

FUELS

FUEL SYSTEMS — DESCRIPTION / OPERATION

1. General Information

The fuel system of the Gulfstream I aircraft contains and supplies all fuel requirements for the two turboprop engines and for the auxiliary power unit (APU), located in the aft fuselage.

The sections will be separated into two parts; the Standard System and the Long Range System, (Aircraft having ASC 125).

Both are low pressure systems to the engine fuel pumps. Operation of the systems are basically identical, but for reasons of clarification, they are covered separately.

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FUEL SYSTEM — STANDARD SYSTEM — DESCRIPTION / OPERATION

1. Description

The fuel system contains and supplies all fuel requirements for the two turboprop engines and for the auxiliary power unit (APU) in the aft fuselage. The fuel system comprises the following components, as shown in Figure 1.

Unit	No. Per A/C	Location
Fuel Filler Cap	2	One at outboard end of each tank
Fuel Tank Vent Valve	2	One at outboard closure rib of each tank
Integral Fuel Tank Water Drain Valve	4	Two in each fuel tank (under wing) cavity bay
Low Pressure Fuel Pump	4	Two in each fuel tank cavity bay
Fuel Flow Check Valve	4	One attached to each low pressure pump
Engine Main Fuel Shutoff Valve	2	One attached to rear wing beam outside of each integral tank
Defueling Drain Valve	2	One in each main wheel well area
Fuel Flow Transmitter	2	One in aft side of each main wheel well
Low Pressure Warning Switch	2	One mounted at each engine interstage compressor casing
Differential Pressure Switch	2	One mounted at each engine interstage compressor casing
Crossfeed Shutoff Valve	1	Inboard of left rear wing beam
Crossfeed Line Water Drain Valve	2	One each side of fuselage
Fuel Quantity Probe	8	Four in each integral tank
Fuel Ejector	2	One in each fuel cavity bay

Fuel is contained in and supplied from an integral fuel tank in each wing. The tanks are vented to atmosphere at Wing Station 307. A float valve minimizes fuel spillage overboard during aircraft maneuvers. The tanks are gravity filled from the top of each wing, at a fuel filler cap.

The fuel tanks comprise a large portion of each wing. The inboard end of the integral tank is at Wing Station 45. This is approximately the junction point of the wing and the fuselage skin. The outboard end of this tank is at Wing Station 307. The integral fuel tanks are constructed with a shallow channel groove manufactured between the overlapping metal skin and adjacent structural members. Non-curing "Thiocol" rubber is injected under pressure into this channel seal groove to seal the fuel tanks. This method of fuel cell sealing is quick and efficient. One of the great advantages of this method is that any fuel leak can be sealed from the outside of the tank without removing any tank access covers or disturbing any component inside the tank.

Normally, the right wing tank supplies the right engine and the left wing tank supplies the left engine and the auxiliary power unit (APU). A fuel crossfeed line, containing an electrically operated fuel shutoff valve, interconnects the fuel supply line to the left and right engine, and permits the left and right wing tanks to supply one or both engines. The fuel crossfeed system is used to maintain a balanced fuel condition. A direct fuel transfer between the left wing tank and the right wing tank cannot be accomplished.

NOTE: Abnormal system operation may move fuel from one tank to the other.

Flapper check valves are located on the bottom of the first rib and are installed inside the inboard bay. These flapper valves permit fuel to flow to the inboard bay but prevent the loss of fuel from the bay as would occur if the aircraft performed an uncoordinated turn maneuver.

NOTE: The inboard bay will not positively trap fuel within the cavity because fuel may still pass through rib lightening holes located above the flapper valves. The flapper valves, however, will slow down the rate of flow from the bay.

A fuel cavity is formed in the inboard bay of each integral tank by the front and rear beams and the closure rib and first rib. The wing dihedral of 6 1/2 degrees causes this area to be the lowest point of the tanks, therefore, two electrically operated centrifugal low pressure submersible boost pumps are installed in this cavity.

The boost pumps normally supply fuel (at 20 to 22 psi at sea level) to their respective engine through a common engine fuel feed line. When necessary, one boost pump can supply both engines. A check valve on the discharge part of each low pressure fuel pump prevents reverse fuel flow to the integral tank when the pump is inoperative. The boost pumps are submerged in fuel until the tanks are practically dry.

The forward boost pumps are know as the auxiliary pumps and the aft boost pumps as the normal pumps. Switches to operate these pumps are located on the right side of the lower overhead panel in the cockpit and placarded in accordance with the pump they control. The main dc bus supplies electrical power to the auxiliary pumps (forward) and the essential dc bus supplies power to the normal pumps (aft).

The left normal boost pump is also utilized to supply fuel to the APU (Aircraft having ASC 146 have the capability of suppling fuel to the APU from the right normal boost pump). A separate boss and line in the tank carry fuel to the rear beam. This line is then picked up by a teflon lined, wire braided, flexible line, which carries APU fuel into the pressurized area at the left fillet, under the cabin floor. Through the aft pressure dome to an APU external shutoff valve outside the APU container in the tail. This line passes through special pressure bulkhead fittings and is unbroken from the left wing beam to the APU shutoff valve aft of the pressure dome in the tail.

Fuel quantity capacitance probes (tank units) in each tank system electrically measure and transmits continuously the fuel quantity to its FUEL QUANTITY indicator on the right side of the cockpit, center instrument panel. Each indicator is independent of the other and calibration is from 0-5600 pounds. One indicator is supplied for each tank system. Access plates in the top of each wing permit access to probes. By depressing the FUEL GAGE TEST button on the pilots side of the cockpit eyebrow panel, the fuel quantity gaging system in the left and right wing tanks is simultaneously tested and both tanks fuel quantity indicator is driven downscale at maximum speed to check system operation. A low fuel warning system is incorporated in the fuel quantity gaging system. When the fuel quantity in either wing's tank(s) is reduced to approximately 720 pounds, the L FUEL QUAN or R FUEL QUAN light(s) on the master warning lights panel come on.

The usable fuel and tank capacities for the Gulfstream are as follows: (based on temperature of 59°F and a fuel weight of 6.75 lbs/gal).

Left Tank:	5231.2 pounds	—	775.0 gallons
Right Tank:	5231.2 pounds	—	775.0 gallons
Total:	10,462.4 pounds	—	1550.0 gallons

The low pressure fuel filter, in the flow control unit (FCU) on the lower left side of each engine, removes solids from fuel in the supply line. Excessive fuel pressure drop (1.5 psi to 1.9 psi) through the filter is sensed by the fuel pressure differential switch, mounted on the left side of each engine. Should ice crystals in the fuel clog the filter and cause this pressure drop, a fuel deicing system is used to maintain normal fuel pressure in the engine supply line.

A fuel low pressure warning switch, mounted on the left side of each engine and connected to each engine fuel supply line, by a pressure sensing line, senses fuel pressure to the engine-driven high pressure fuel pump. When fuel pressure from boost pumps falls to 9 $\pm$ 1/4 psi, the warning switch electrically lights the L FUEL PRESS or R FUEL PRESS light on the master warning system. Warning lights automatically go out when fuel pressure in the engine fuel supply line is above 9 $\pm$ 1/4 psi but less than 11 psi.

Nonelectrical ejectors are used to pick up fuel from a remote pocket of each tank and move it up to the pump area. The installation of the fuel ejector system in the aircraft increases the available supply of fuel to the fuel boost pumps at low fuel state conditions with the aircraft in an aborted landing configuration. The ejector utilizes fuel pressure tapped from the boost pump crossfeed line to power a venturi in the ejector. The low pressure section of the venturi draws fuel from the aft section of the fuel boost pump bay and delivers it to the

forward side of the transverse baffle into the fuel boost pump intake area. The ejector operates whenever either or both fuel boost pumps are operating. However, the function of the ejector is only of value at low fuel state in a climb attitude.

A main fuel shutoff valve is attached to the rear beam on the outside of the fuel tank, in the flap bay area. It is an electrically operated sliding gate valve, receiving its power from the essential dc bus. The valves are normally open at all times. Pulling the T-handle closes the valve while returning the handle to the normal position causes the shutoff valves to open. Markings (OPEN-CLOSE) on the valve case and a flag lever attached to the sliding gate valve are provided to indicated the position of the shutoff valve.

A manually operated, normally closed, fuel drain valve is located in the feed line running to each engine, permitting drainage of the fuel tank. The valves are located in the main gear wheel well, just aft of the fuel flow transmitter.

NOTE: The fuel drain valve on Aircraft 1 - 164, 322 and 323 having ASC 178 and Aircraft 165 - 200 is located forward of the fuel flow transmitter. (See Figure 1)

A fuel flow transmitter is installed in the fuel feed line running to each engine. The transmitters are spring loaded, vane, auto-sync type units, and will provide 100 - 1500 pounds-per-hour indication. The unit incorporates a bypass valve to permit passage of fuel to the engine if the unit fails.

The transmitters are shock mounted and located in each main gear wheel well. Two fuel flow indicates, one for each engine, are installed on the center instrument panel.

Eight water drain valves, one on each side of the fuselage in the crossfeed line, two on the inboard bottom on each integral tank and on the bottom of each low pressure filter bowl (FCU), permit maintenance drainage of accumulated water settled from the fuel system. These valves are operated manually. To preclude the admittance of air into the system, the crossfeed and filter drains should be operated only when the boost pumps are operating.

2. Operation

(See Figure 2).

Placing the fuel boost pump switches to the ON position starts operation of the boost pumps. Fuel is picked up by the pumps, flows out the pump discharge ports and manifolds into a single engine feed line. This is accomplished in the following manner: The rear pump incorporates a wiggins T-Connection which connects to the aircraft's crossfeed line. A metal line interconnects the forward leg of this fitting with a second wiggings T-Connection located between the two pumps. A metal line from the forward boost pump connects to the forward leg. The engine feed line attaches to the third leg of the forward T-Connection, is directed outboard and then routed aft, terminating at the main fuel shutoff valve, located on the wing rear beam just outboard of the engine nacelles. From the main fuel shutoff valve the feed line is routed into the main landing gear wheel well each side. A manually operated, normally closed, fuel drain valve is located in this line and permits drainage of the fuel tank. A fuel flow transmitter is also installed in the feed line running to each engine. The transmitters are shock mounted and located inside each main gear wheel well. The transmitter incorporates a bypass valve to permit the passage of fuel to the engine if the unit fails.

Leaving the fuel flow transmitter and drain valve, the engine feed line run continues forward towards the engine. The line run in the left and right wheel wells are slightly different from each other. Since the filters and the engine-driven fuel pumps are located on the left side of the engines, the fuel lines are routed correspondingly. In the left wheel well, the fuel flow transmitter is on the aft outboard side. The fuel feed line continues straight forward towards the engine. In the right wheel well, the fuel flow transmitter is also located on the aft outboard side, but this is the right hand side of the wheel well. Consequently, the fuel line in this nacelle is directed towards the left side of the wheel well and then continues forward in the direction of the engine. These lines from the flow transmitter to the firewall are flexible wire braided hose assemblies. On the forward side of the firewall the engine feed line continues to the heat exchange unit. The filter and the engine-driven fuel pump are a part of the engine fuel system, and are furnished with the engine by Rolls-Royce, Ltd.

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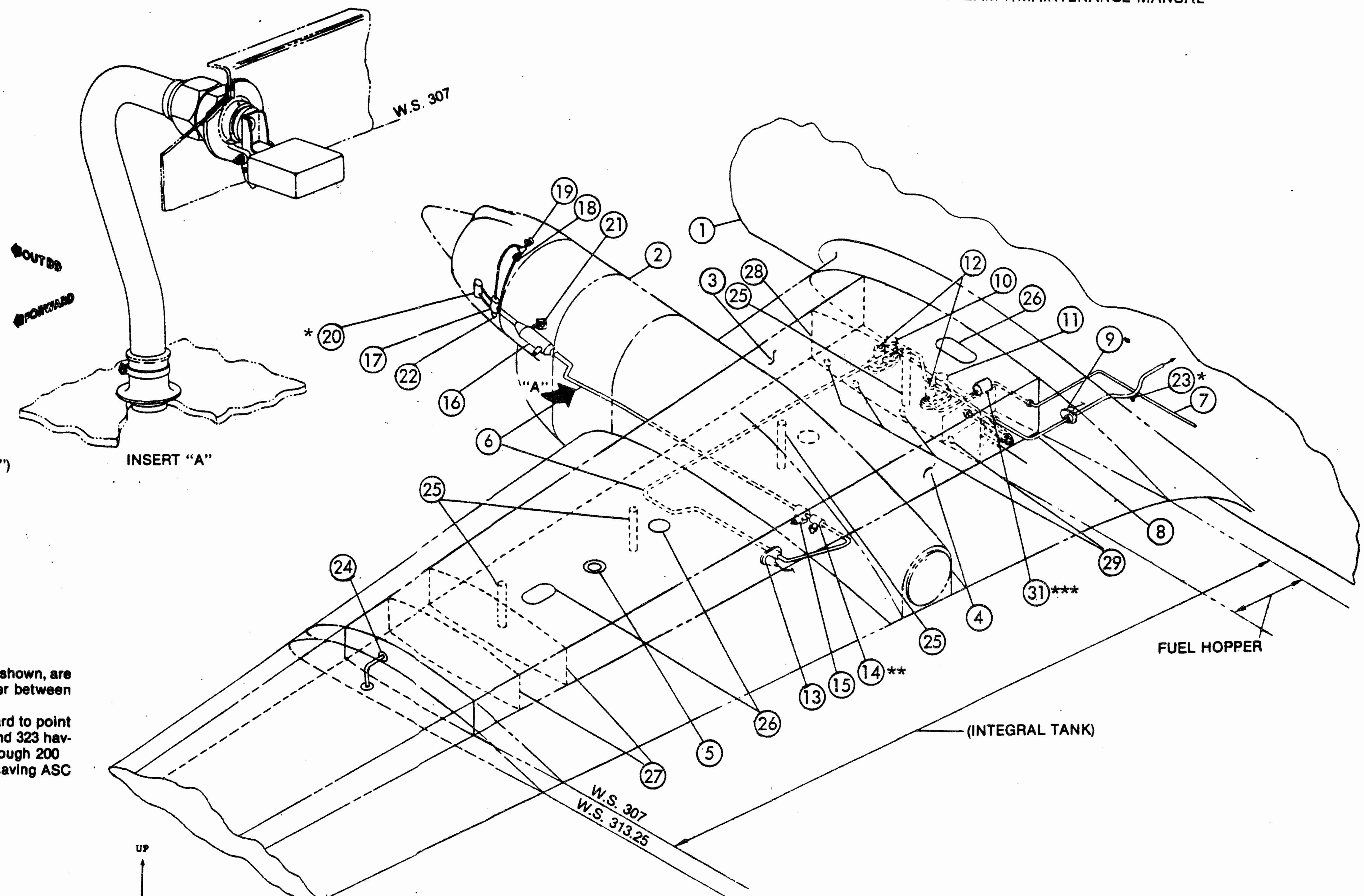
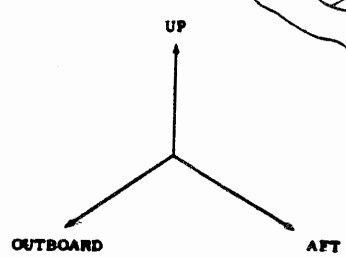
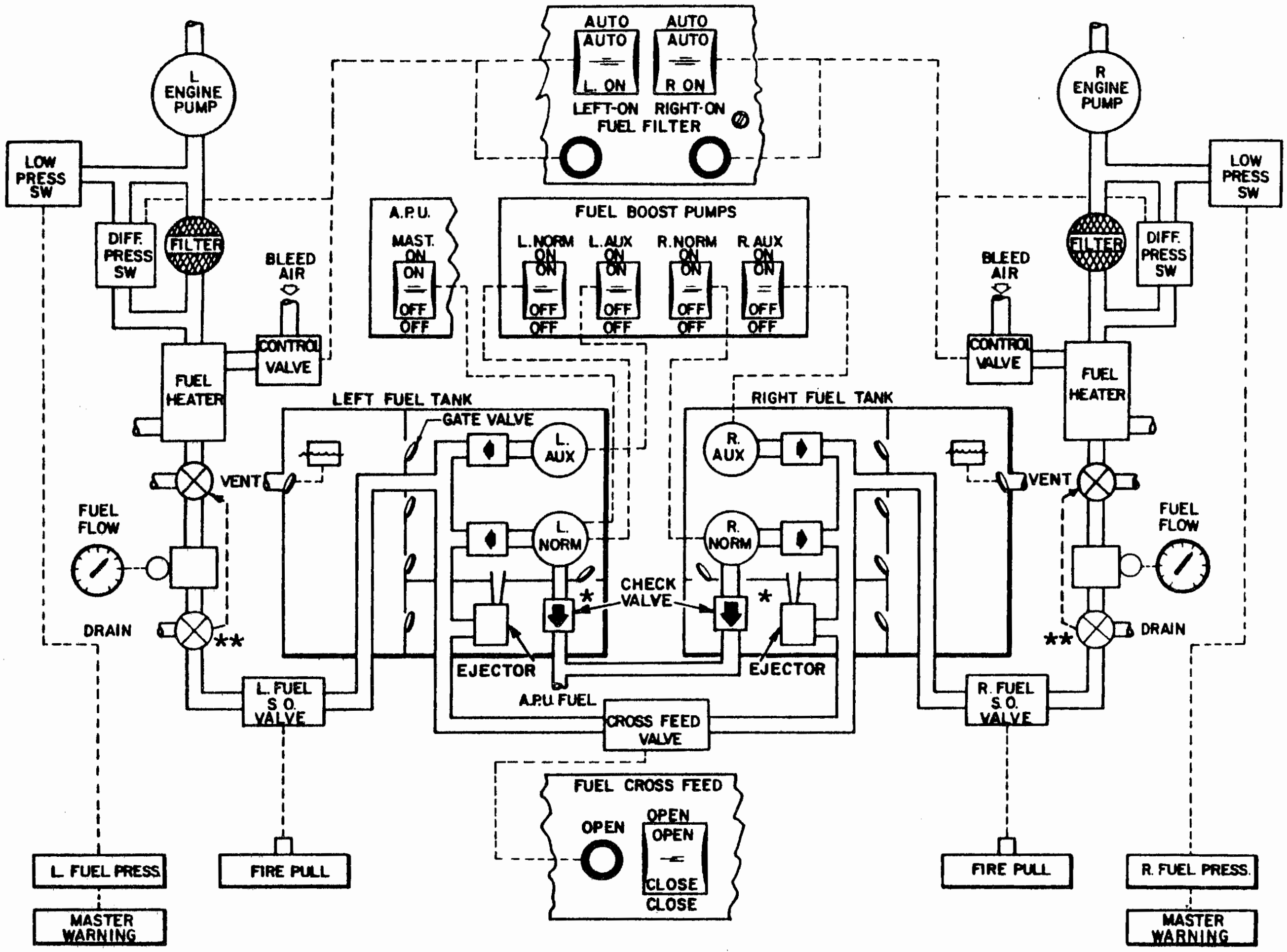
1. FUSELAGE
2. LEFT NACELLE
3. WING FRONT BEAM
4. WING REAR BEAM
5. FILLER CAP
6. ENGINE FUEL FEED LINE
7. FUEL SUPPLY LINE TO APU
8. FUEL CROSSFEED LINE
9. FUEL CROSSFEED SHUTOFF VALVE
10. FUEL BOOST PUM (AUX.)
11. FUEL BOOST PUMP (NORM.)
12. BOOST PUMP CHECK VALVE
13. FUEL SHUTOFF VALVE
14. DEFUELING DRAIN VALVE
15. FUEL FLOWMETER TRANSMITTER
16. FUEL HEATER (EXCHANGER)
17. FUEL FILTER
18. FUEL DIFFERENTIAL PRESSURE SWITCH
19. FUEL LOW PRESSURE WARNING SWITCH
20. FUEL HIGH PRESSURE PUMP
21. HOT AIR GATE VALVE
22. LOW PRESSURE FUEL FILTER DRAIN
23. WATER DRAIN VALVE (3)
24. FLOAT VALVE AND VENT (SEE INSERT "A")
25. FUEL QUANTITY PROBE (TANK UNIT)
26. ACCESS PLATE
27. BAFFLE PLATE
28. WING FIRST RIB
29. FLAPPER VALVE
30. FUEL EJECTOR
31. CHECK VALVE



* Two additional water drain valves (Item 23) not shown, are located in integral tank bottom of fuel hopper between boost pumps (Item 10 and 11)
** Defueling drain valve (Item 14) is moved forward to point "A" on Aircraft 1 through 164 including 322 and 323 having ASC 178 incorporated and Aircraft 179 through 200
*** Aircraft 1 through 200 including 322 and 323 having ASC 146 incorporated



Fuel System Components Location — Left Wing
(Right Wing Similar) Standard
Figure 1.



* AIRCRAFT 1 THROUGH 200, INCLUDING 322 AND 323, HAVING ASC 146 INCORPORATED

** AIRCRAFT 1 THROUGH 164, INCLUDING 322 AND 323 HAVING ASC 178 INCORPORATED AND AIRCRAFT 165 THROUGH 200, THE "DRAIN" VALVE IS MOVED FORWARD OF THE FUEL FLOW TRANSMITTER, AS INDICATED.

Fuel System — Functional Schematic
Figure 2.

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FUEL SYSTEM — STANDARD SYSTEM — MAINTENANCE PRACTICES

1. **Fuel System — Bleed Procedure**

WARNING: TO PREVENT POSSIBILITY OF ENGINE SURGE OR COMPLETE FAILURE OF AN ENGINE (OR ENGINES) IN FLIGHT, CAUSED BY ENTRAPPED AIR IN THE FUEL SYSTEM, IT IS MANDATORY THAT THE FUEL SYSTEM BE BLED WHEN MAINTENANCE IS PERFORMED AS OUTLINED IN THE FOLLOWING NOTES.

NOTE: Entire fuel system must be bled if: (perform Paragraph A. and B. below)

- Any fuel system component or fuel line in fuel tank has been changed or disconnected.
- Fuel system has been drained and refilled.
- Any items in Notes listed below were performed with fuel shutoff valve open.

Fuel system downstream of main fuel shutoff valve must be bled if: (perform Paragraph 2. below)

- Any unit has been changed downstream of shutoff valve, not including low pressure warning switch or differential pressure switch.
- Any fuel line downstream of shutoff valve is disconnected.
- Low-pressure filter is changed.
- Engine is shutdown by closing shutoff valve before closing HP cock.
- Fuel filters were drained without fuel boost pumps operating.
- Filter de-icing lines have been disconnected from fuel control unit.
- An engine is changed.

Fuel filter pressure sensing lines must be bled if: (perform Paragraph C. below)

- Low-pressure warning switch is removed.
- Differential pressure switch is removed.
- Any fuel pressure sensing line disconnected at any point other than at fuel control unit.

A. Bleed Fuel System

(1) Attach 1 1/4-inch inside diameter hose of adequate length to fuel drain in lower nacelle on side being bled.

NOTE: Hose must maintain constant inside diameter for entire length; a slow acting shutoff valve should be installed in drain line.

(2) Close crossfeed shutoff valve.

(3) Energize AUX and NORM pumps on side being bled.

(4) Open fuel drain valve and allow fuel to flow for 1 minute.

NOTE: Fuel can be returned to fuel cell, fuel pit, or other suitable container.

(5) Energize opposite side AUX and NORM fuel pumps.

(6) Open crossfeed shutoff valve.

(7) De-energize fuel pumps on side being bled.

(8) Permit fuel to flow from opposite tank to tank being bled and out of drain for 1 minute.

(9) Energize AUX and NORM pumps on side being bled.

(10) Close crossfeed shutoff valve.

(11) De-energize fuel pumps on opposite side.

NOTE: Do not under any circumstances deenergize fuel pumps on side being bled until bleeding has been completed.

(12) Close slow acting shutoff valve, then close drain valve in lower nacelle and remove drain hose.

(13) Perform Paragraph B. below.

B. Bleed Downstream of Main Fuel Shutoff Valve:

(1) Ensure fuel crossfeed shutoff valve is closed.

(2) Ensure main fuel shutoff valve is closed for side on which work was performed.

(3) Ensure AUX and NORM pumps are energized on side affected.

(4) Open main fuel shutoff valve.

(5) Open engine fuel filter drain valve and attach Rolls-Royce bleed hoses to both bleed valves across engine-driven pump.

(6) Permit approximately one gallon of fuel to flow out of filter drain, then close drain valve.

(7) Alternately open and close both engine-driven pump bleed valves several times; leave both bleed hoses attached.

(8) Loosen B-nut on low pressure warning switch, bleed until solid fuel (no air) comes from line; tighten B-nut.

(9) Loosen swivel nut of flex line on HP side of differential pressure switch at 45° elbow; bleed until solid flow of fuel comes from line, tighten swivel nut.

(10) Repeat Steps (8) and (9) above to ensure release of trapped air.

(11) Remove all hoses and cap all fittings; check that all connections and caps are secure.

(12) De-energize AUX and NORM fuel pumps.

(13) Perform engine run as detailed in Paragraph D.

C. Bleed Pressure Sensing Lines:

(1) Close fuel crossfeed shutoff valve.

(2) Check that main fuel shutoff valve is closed for side which was bled.

(3) Energize AUX and NORM fuel pumps on side affected.

(4) Open main fuel shutoff valve.

(5) Loosen B-nut on low pressure warning switch and bleed until solid fuel (no air) comes from line; tighten B-nut.

(6) Loosen swivel nut of flex line on HP side of differential pressure switch at 45° elbow; bleed until solid fuel comes from line and tighten swivel nut.

(7) Repeat Steps (5) and (6) above to ensure release of trapped air.

(8) Check that all connections are secure.

(9) De-energize fuel pumps.

(10) Perform engine run as detailed in Paragraph D. below.

D. Engine Ground Run After Bleeding:

(1) Apply electrical power to essential dc bus.

(2) Ensure water methanol switch is off.

(3) Pull left and right water methanol circuit breakers.

(4) Start engine(s) using normal start procedures.

(5) With engine running, advance one engine to 13,000 rpm and hold for 1 minute.

(6) Reduce power to 8,000 rpm and de-energize fuel boost pumps.

(7) Advance engine to 15,000 rpm operate fuel crossfeed valve to open and hold for 1 minute.

(8) Hold engine to 15,000 rpm and select fuel crossfeed valve to close.

(9) Reduce power to 8,000 rpm.

(10) Repeat Steps (5) - (9) above for other engine.

(11) Shutdown engine(s).

(12) Depress water methanol circuit breakers.

NOTE: If rpm fluctuation is experienced, fuel system must be rebled due to trapped air still present in fuel system. If engines continue to operate normally, bleeding operation has been completed satisfactorily.

2. Fuel System — Integrity Test

NOTE: This is a suction and pressure check to ascertain tightness of inner tank lines and fittings. After fuel system bleeding has been accomplished, conduct a maximum power (15,000 rpm) engine ground run as follows:

A. Apply power to main and essential dc bus.

B. Fill each integral tank with approximately 50 gallons of fuel.

C. Close fuel crossfeed and water methanol switches.

D. Pull Water methanol circuit breakers.

E. Place NORM and AUX switches to ON for the side being tested.

F. Start engine using normal start procedures (Refer to Airplane Flight Manual).

G. After engine has started, place NORM and AUX pump switches to OFF.

H. Run engine for approximately 3 minutes at varying rpm from ground idle to takeoff.

I. Place NORM and AUX pump switches to ON and vary engine rpm at all ranges for approximately 3 minutes.

J. Place NORM and AUX pump switches to OFF and vary engine rpm at all ranges for approximately 3 minutes.

K. Turn NORM and AUX pump switches ON.

L. Shutdown engine using normal procedures.

M. Remove Electrical Power.

N. Depress water methanol circuit breakers.

NOTE: With boost pumps off, engine pumps draw fuel from tanks. Operation of engine depends on suction of engine pump. If there is a leak in a connection such as a Wiggins fitting, the pump sucks air and flames out engine. This only holds true when there is a low fuel quantity in tanks, thus exposing all fittings within fuel tanks.

3. Defuel Drain Valve — Removal / Installation

A. Removal

(1) Energize essential dc bus.

(2) Pull FIRE PULL T-handle to close main fuel shutoff valve.

(3) De-energize essential dc bus.

(4) Remove drain valve.

B. Installation

(1) Install new O-ring on drain valve.

(2) Install drain valve. Tighten jam nut.

(3) Perform Fuel System — Bleed Procedure, this Section.

4. Defueling

WARNING: WHEN DRAINING FUEL, HAVE FIRE FIGHTING EQUIPMENT AVAILABLE TO COMBAT PETROLEUM FIRES. DO NOT PERMIT SMOKING, OR USE OF ANY ELECTRICAL EQUIPMENT WHICH DOES NOT EMPLOY EXPLOSION PROOF FEATURES. ELECTRICALLY GROUND ALL EQUIPMENT AND AIRCRAFT. ENSURE THAT THERE ARE NO DISCONNECTED ELECTRICAL LEADS WITHIN THE TANKS.

CAUTION: PRIOR TO DEFUELING, THE FUEL TANK VENT SHOULD BE CHECKED TO ENSURE THAT IT IS OPEN AND THAT NO OBSTRUCTIONS EXIST IN THE OVERBOARD VENT LINE. A CLOSED OR OBSTRUCTED VENT COULD RESULT IN COLLAPSING THE WING.

ON AIRCRAFT 1 - 163, 322 AND 323, HAVING ASC 178 AND AIRCRAFT 164 - 200 THE DRAIN VALVES ARE RELOCATED DOWNSTREAM OF THE FLOWMETER TRANSMITTER. WHEN DEFUELING OR DRAINING FUEL, THE FUEL FLOW MAY EXCEED THE FLOW LIMITATIONS OF THE FLOWMETER INDICATOR RESULTING IN DAMAGE TO THE INSTRUMENT. IN ORDER TO PREVENT THIS, THE AC INSTRUMENT BUS MUST BE DEACTIVATED BY ENSURING THAT THE A, B AND E INVERTERS ARE IN THE OFF POSITION.

NOTE: These procedures are written for the right side only. It must be repeated on the left side to completely defuel the aircraft fuel tank. It is recommended that a slow acting shutoff valve be installed at the drain valve outlet during defueling operations. The slow acting shutoff valve is closed first when draining under pressure in order to prevent shocking of the altitude capsule in the flow control unit.

A. Pumping through the drain valve in the main landing gear wheel well.

(1) Attach one end of drain hose to right hand drain valve located in right wheel well.

(2) Ensure that the following controls are in position indicated.

- High Pressure Fuel Cock - FUEL OFF
- FUEL CROSSFEED switch - CLOSE
- FIRE PULL T-handle - Pushed in (Open)

(3) Place BATT switch to NORM.

(4) Place R NORM or R AUX boost pump to ON.

(5) Open drain and slow acting valves.

(6) When defueling is complete, place R NORM or R AUX boost pump to OFF.

(7) Place BATT switch to OFF.

(8) Close slow acting valve, then drain valve.

(9) Remove drain hose and remove slow acting valve.

(10) Fuel remaining in tanks at this point may be removed through two tank drains, located on lower inboard wing planks.

B. Suction through drain valves in main landing gear wheel well.

(1) Attach one end of drain hose to right hand drain valve located in right wheel well.

(2) Ensure that the following controls are in position indicated.

- High Pressure Fuel Cock - FUEL OFF
- FUEL CROSSFEED switch - CLOSE
- FIRE PULL T-handle - Pushed in (Open)

(3) Place BATT switch to NORM.

(4) Place R NORM or R AUX boost pump to ON.

(5) Open drain and slow acting valves.

(6) Under truck suction, start defueling.

(7) Place R NORM or R AUX boost pump to OFF.

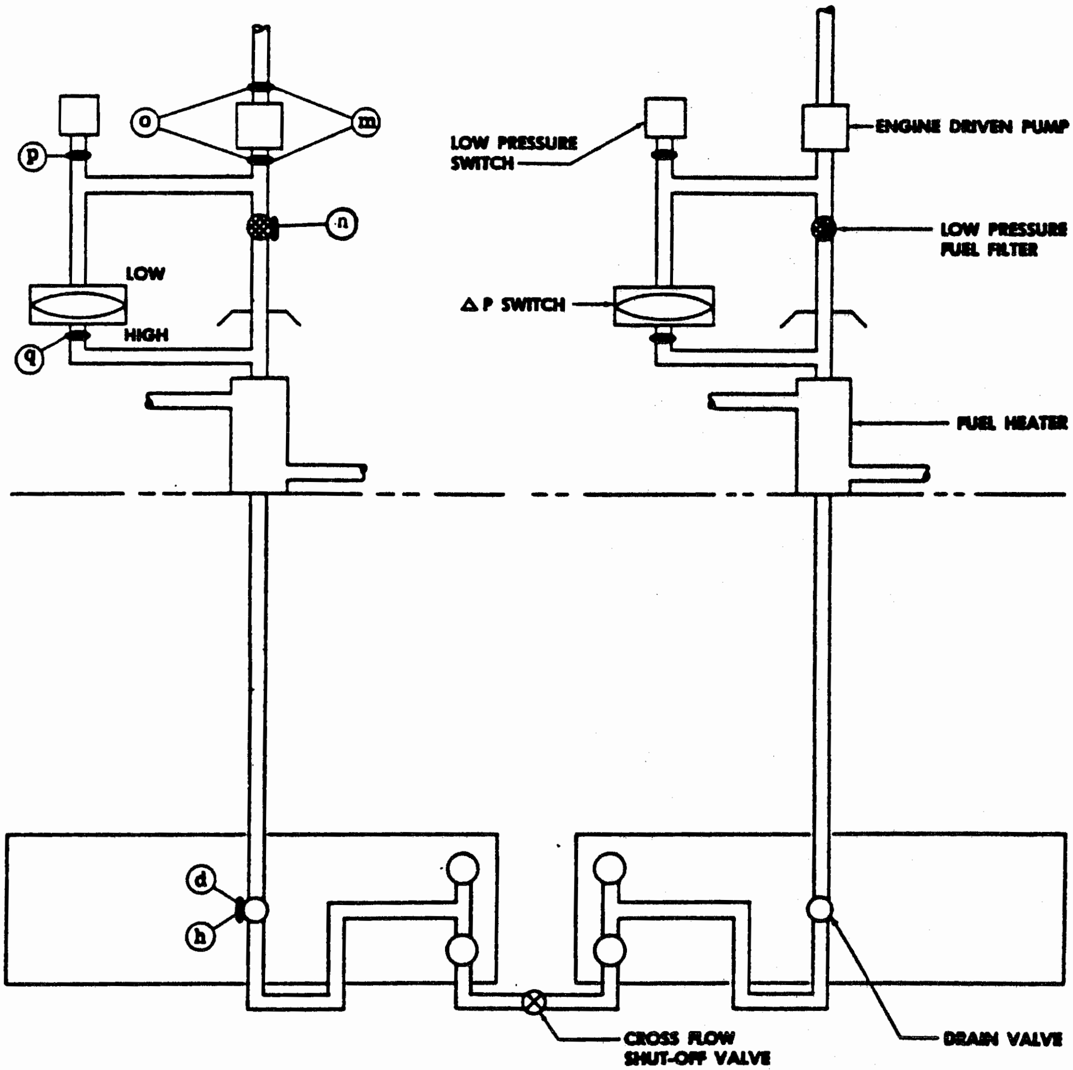
(8) When defueling is completed close slow acting valve and drain valve.

(9) Remove defueling hose.

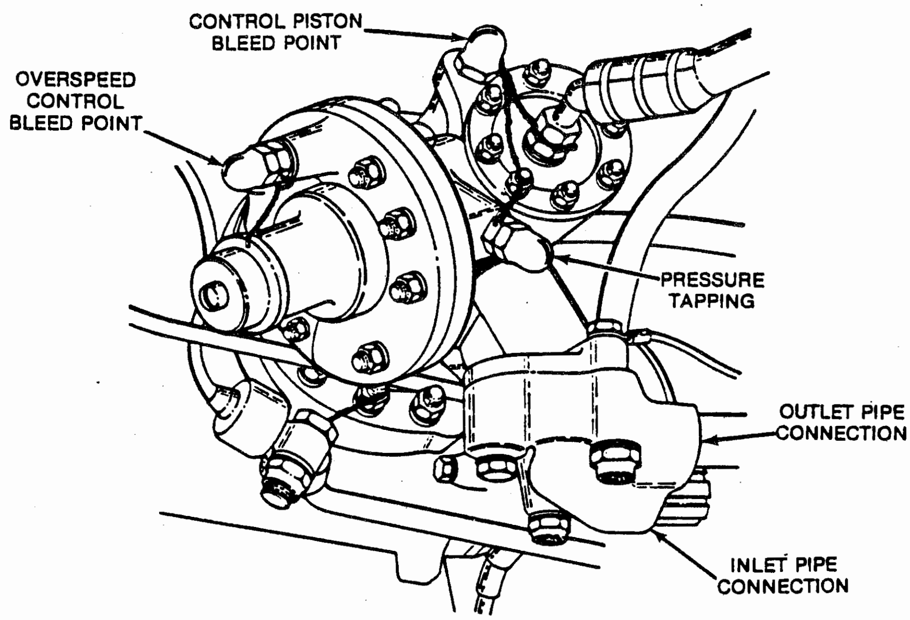
(10) Remove slow acting valve and drain remaining fuel in tanks, remove 2 tank drains located on lower inboard wing planks.

Note

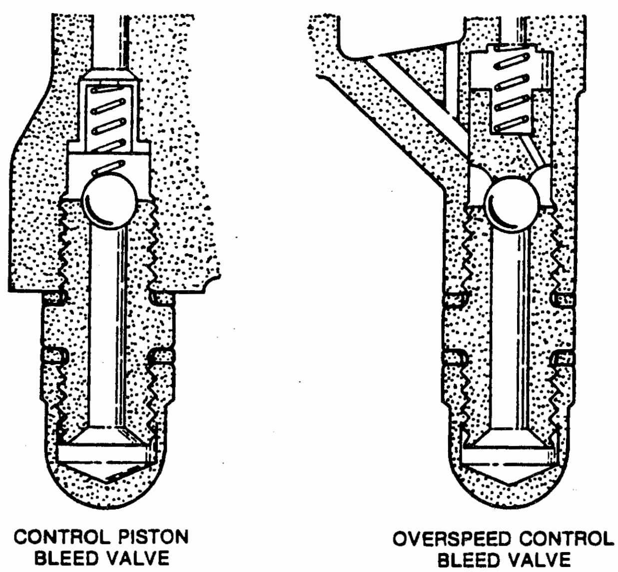
This diagram applies to bleed procedures part 1 only.



Fuel System Bleed Points
Figure 201.



HP Fuel Pump Bleed Points
Figure 202.



Bleed Valve Assembly
Figure 203.

FUEL SYSTEM (AIRCRAFT HAVING ASC 125) — DESCRIPTION / OPERATION

1. Description

The fuel system contains and supplies all fuel requirements for the two turboprop engines and for the auxiliary power unit (APU) in the aft fuselage. The fuel system comprises the following components, as shown in Figure 1.

Unit	No. Per A/C	Location
Fuel Filler Cap	2	One in No. 5 fuel cell, Wing Station 386
Fuel Tank Vent Valve	2	One at each bypass valve in integral tank (Note overboard vent has no valve)
Integral Fuel Tank Water Drain Valve	4	Two in each fuel tank (under wing) cavity bay
Low-Pressure Fuel Pump	4	Two in each fuel tank cavity bay
Fuel Flow Check Valve	4	One attached to each low-pressure pump
Engine Main Fuel Shutoff Valve	2	One attached to rear wing beam outside of each integral tank
Defueling Drain Valve	2	One in each main wheel well area
Fuel Flow Transmitter	2	One in aft side of each main wheel well
Low-Pressure Warning Switch	2	One mounted at each engine interstage compressor casing
Differential Pressure Switch	2	One mounted at each engine interstage compressor casing
Crossfeed Shutoff Valve	1	Inboard on left rear wing beam
Crossfeed Line Water Drain Valve	2	One each side of fuselage
Fuel Quantity Probe	12	Six in each system
Fuel Ejector	2	One in each fuel cavity bay

Fuel is contained in, and supplied from eight fuel tanks (No. 1 to No. 8, See Figure 1 in each wing. The inboard, integral tank (No. 1), containing most of the fuel in the wing, is connected at Wing Station 307 to seven, inter-connected, bladder-type fuel cells (No. 2 to No. 8 proceeding outboard) installed between the outer wing panel ribs from Wing Station 313 to 456. All wing fuel tanks (No. 1 to No. 8) are gravity filled from the top of each wing, at a fuel filler cap connected to cell No. 5 at Wing Station 386. A bypass vent line vents the integral No. 1 tank of each wing to the outboard, bladder-type cell No. 5. A common vent line, connecting all the bladder-type cells (No. 2 to No. 8) of each wing, vents all wing tanks (including integral No. 1 tank) to the outboard bladder cell No. 8, which is vented overboard under this cell. A float valve, in the integral tank bypass vent line, prevents fuel surge outboard to bladder cells, during aircraft maneuvers.

NOTE: A differential pressure vent (spring check) valve is located in extreme outboard (cell No. 8) and only comes into action, if the regular vent (overboard lower side of wing) should freeze, the resultant being a lower pressure in the cell, whereby higher ambient pressure would overcome spring tension and allow the spring check valve to become the vent until icing condition is alleviated.

The fuel tanks comprise the entire wing. The outboard end of the integral tank is at Wing Station 45. This is approximately the junction point of the wing and the fuselage skin. The outboard end of this tank is at Wing Station 307. The seven bladder type tanks extend from Wing Station 313 to 456. The integral fuel tanks are constructed with a shallow channel groove manufactured between the overlapping metal skin and adjacent structural members. Non-curing Thiocol rubber is injected under pressure into this channel seal groove to seal the fuel tanks. This method of fuel cell sealing is quick and efficient. One of the great advantages of this method is that any fuel leak can be sealed from the outside of the tank without removing any tank access covers or disturbing any component inside the tank.

Normally, the right wing tank supplies the right engine and the left wing tank supplies the left engine and the Auxiliary Power Unit (APU). A fuel crossfeed line, containing an electrically operated fuel shutoff valve, interconnects the fuel supply line to the left and right engine and permits the left and right wing tanks to supply one or both engines. The fuel crossfeed system is used to maintain a balanced fuel condition. A direct fuel transfer between the left wing tank and the right wing tank cannot be accomplished.

NOTE: Abnormal system operation may move fuel from one tank to the other.

Flapper check valves are located on the bottom of the first rib and are installed inside the inboard bay. These flapper valves permit fuel to flow to the inboard bay but prevent the loss of fuel from the bay as would occur if the aircraft performed an uncoordinated turn maneuver.

NOTE: The inboard bay will not positively trap fuel within the cavity because fuel may still pass through rib lightening holes located above the flapper valves. The flapper valves, however, will slow down the rate of flow from the bay.

A fuel cavity is formed in the inboard bay of each integral tank by the front and rear beams and the closure rib and first rib. The wing dihedral of 6 1/2 degrees causes this area to be the lowest point of the tanks, therefore, two electrically operated centrifugal type low-pressure submersible boost pumps are installed in this cavity.

The boost pumps normally supply fuel (at 20 to 22 psi at sea level) to their respective engine through a common engine fuel feed line. When necessary one boost pump can supply both engines. A check valve on the discharge part of each low-pressure fuel pump prevents reverse fuel flow to the integral tank when the pump is inoperative. The boost pumps are submerged in fuel until the tanks are practically dry.

The forward boost pumps are known as the auxiliary pumps and the aft boost pumps as the normal pumps switches to operate these pumps are located on the right side of the lower overhead panel in the cockpit and placarded in accordance with the pump they control. The main dc bus supplies electrical power to the auxiliary pumps (forward) and the essential dc bus supplies power to the normal pumps (aft).

The left normal boost pump is also utilized to supply fuel to the APU (Aircraft 322 and 323 having ASC 146 have the capability of suppling fuel to the APU from the right normal boost pump). A separate boss and line in the tank carry fuel to the rear beam. This line is then picked up by a teflon lined, wire braided, flexible line, which carries APU fuel into the pressurized area at the left fillet, under the cabin floor through the aft pressure dome to an APU external shutoff valve outside the APU container in the tail. This line passes through special pressure bulkhead fittings and is unbroken from the left rear wing beam to the APU shutoff valve aft of the pressure dome in the tail.

Fuel quantity capacitance probes (tank units) in each tank system electrically measure and transmit continuously the fuel quantity to its FUEL QUANTITY indicator on the right side of the cockpit center instrument panel. Each indicator is independent of the other and calibration is from 0 - 6300 pounds. One indicator is supplied for each tank system. Access plates in the top of each wing permit access to probes. By depressing the FUEL GAGE TEST button on the pilots side of the cockpit eyebrow panel, the fuel quantity indicating system in the left and right wing tanks is simultaneously tested and both tanks fuel quantity indicator is driven downscale at maximum speed to check system operation. A low fuel warning system is incorporated in the fuel quantity indicating system. When the fuel quantity in either wing tank is reduced to approximately 720 pounds, the L FUEL QUAN or R FUEL QUAN lights on the master warning system come on.

The usable fuel and tank capacities for the increased fuel capacities (ASC 125) are as follows, (based on temperature of 59°F and a fuel weight of 6.75 lbs/gal.)

Left Tank:	6033.8 pounds	—	893.9 gallons
Right Tank:	6073.0 pounds	—	899.7 gallons
Total:	12,106.8 pounds	—	1793.6 gallons

NOTE: The left wing tanks contain less fuel than the right wing tanks because of a narrower No. 6 fuel cell, which permits clearance for the aileron trim tab cables.

A fuel low-pressure warning switch, mounted on the left side of each engine and connected to each engine fuel supply line, by a pressure-sensing line, senses fuel pressure to the engine-driven, high pressure fuel pump.

When fuel pressure from boost pumps falls to 9 $\pm$ 1/4 psi, the warning switch electrically lights the L FUEL PRESS or R FUEL PRESS light on the master warning system. Warning lights automatically go out when fuel pressure in the engine fuel supply line is above 9 1/4 but less than 11 psi.

Excessive fuel pressure drop (1.5 psi to 1.9 psi) through the filter is sensed by the fuel pressure differential switch, mounted on the left side of each engine. Should ice crystals in the fuel clog the filter and cause this pressure drop, a fuel de-icing system is used to maintain normal fuel pressure in the engine supply line.

Non-electrical ejectors are used to pick up fuel from a remote pocket of each tank and move it up to the pump area. The installation of the fuel ejector system in the aircraft increases the available supply of fuel to the fuel boost pumps at low fuel state conditions with the aircraft in an aborted landing configuration. The ejector utilizes fuel pressure tapped from the boost pump crossfeed line to power a venturi in the ejector. The low-pressure section of the venturi draws fuel from the aft section of the fuel boost pump bay and delivers it to the forward side of the transverse baffle into the fuel boost pump intake area. The ejector operates whenever either or both fuel boost pumps are operating. However, the function of the ejector is only of value at low fuel state in a climb attitude

A main fuel shutoff valve is attached to the rear beam on the outside of the fuel tank, in the flap bay area. It is an electrically operated sliding gate valve, receiving its power from the essential dc bus. The valves are normally open at all times. Pulling the T-handle closes the valve while returning the handle to the normal position causes the shutoff valves to open. Markings (OPEN-CLOSE) on the valve case and a flag lever attached to the sliding gate valve are provided to indicate the position of the shutoff valve.

A manually operated, normally closed, fuel drain valve is located in the feed line running to each engine, permitting drainage of the fuel tank. The valves are located in the main gear wheel well, just aft of the fuel flow transmitter.

NOTE: The fuel drain valve on aircraft 1 - 163, 322 and 323 having ASC 178 and aircraft 164 - 200 is located forward of the fuel flow transmitter. (See Figure 1)

A fuel flow transmitter is installed in the fuel feed line running to each engine. The transmitters are spring loaded, vane, auto-sync type units, and will provide 100-1500 pounds-per-hour indication. The unit incorporates a bypass valve to permit passage of fuel to the engine if the unit fails.

The transmitters are shock mounted and located in each main gear wheel well. Two fuel flow indicators, one for each engine, are installed on the center instrument panel.

Eight water drain valves, one on each side of the fuselage in the crossfeed line, two on the inboard bottom on each integral tank and one on the bottom of each low-pressure filter bowl (FCU), permit maintenance drainage of accumulated water settled from the fuel system. These valves are operated manually. To preclude the admittance of air into the system, the crossfeed and filter drains should be operated only when the boost pumps are operating.

2. Operation

(See Standard System, Section 28-0-1)

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1. Fuselage
2. Left Nacelle
3. Wing Front Beam
4. Wing Rear Beam
5. Inter-Connect Fitting/Flapper Valve
6. Engine Fuel Feed Line
7. Fuel Supply Line to APU
8. Fuel Crossfeed Line
9. Fuel Crossfeed Shutoff Valve
10. Fuel Boost Pump (Aux.)
11. Fuel Boost Pump (Norm.)
12. Boost Pump Check Valve
13. Fuel Shutoff Valve
14. Defueling Drain Valve
15. Fuel Flowmeter Transmitter
16. Fuel Heater (Exchanger)
17. Fuel Filter (FCU)
18. Fuel Diff. Press. Switch
19. Fuel Low Pressure Warning Switch
20. Fuel High Pressure Pump (Engine Driven)
21. Hot Air Gate Valve
22. Low Pressure Fuel Filter Drain
23. Water Drain Valve (3)
24. Bypass Vent Line
25. Float Valve & Vent
26. Common Vent Line
27. Vent
28. Inter-Connect Fitting (w/o Flapper Valve)

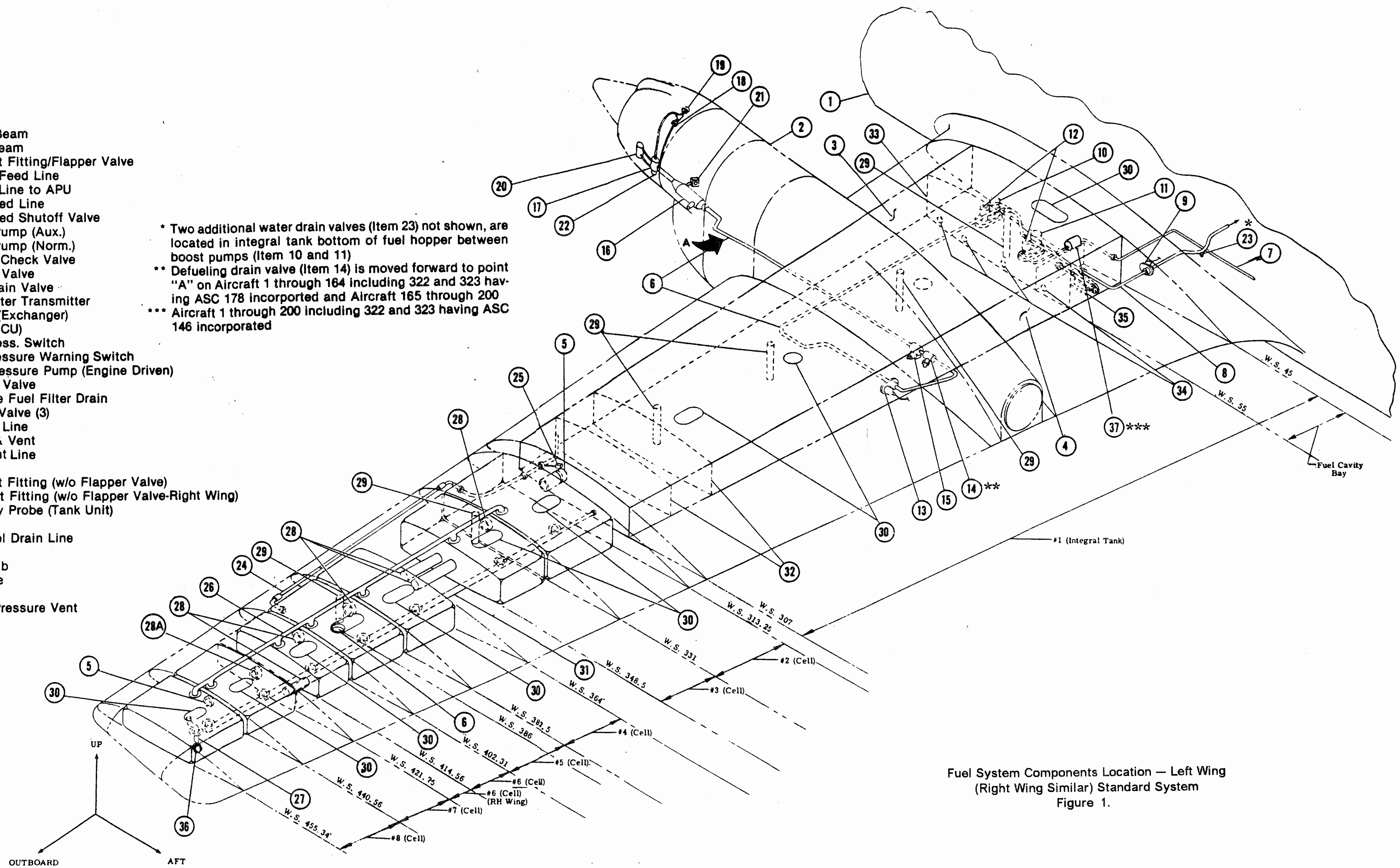
28A. Inter-Connect Fitting (w/o Flapper Valve-Right Wing)

29. Fuel Quantity Probe (Tank Unit)
30. Access Plate
31. Common Fuel Drain Line
32. Baffle Plate
33. Wing First Rib
34. Flapper Valve
35. Fuel Ejector
36. Differential Pressure Vent
37. Check Valve

* Two additional water drain valves (Item 23) not shown, are located in integral tank bottom of fuel hopper between boost pumps (Item 10 and 11)

** Defueling drain valve (Item 14) is moved forward to point "A" on Aircraft 1 through 164 including 322 and 323 having ASC 178 incorporated and Aircraft 165 through 200

*** Aircraft 1 through 200 including 322 and 323 having ASC 146 incorporated



Fuel System Components Location — Left Wing
(Right Wing Similar) Standard System
Figure 1.

FUEL SYSTEM (AIRCRAFT HAVING ASC 125) — MAINTENANCE PRACTICES

NOTE: The maintenance practices on aircraft having ASC 125 are the same as standard aircraft, see Section 28-0-1.

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FUEL SYSTEM MAINTENANCE PRACTICES
(Aircraft 1 Through 200, Including 322 and 323 Having ASC 125 Incorporated)

NOTE

The maintenance practices on aircraft having ASC 125 incorporated are the same as standard aircraft. Refer to Section 28-0-1, page 201.

FUEL TANKS — DESCRIPTION / OPERATION

1. **Standard System**

The total fuel load for the aircraft is contained in two integral fuel tanks. Each have a capacity of 775 gallons.

The left and right wing integral tanks extend from Wing Station 40.95 to 307. The structural members that form the walls of the tank are as follows: Inboard and outboard closure ribs, front and rear beams, and upper and lower covers.

NOTE: The upper and lower covers are nomenclatured as forward plank, forward intermediate plank, aft intermediate plank, and aft plank. (See Figure 1)

All internal aluminum surfaces are protected against corrosion by an anti-corrosive coating. The faying or mating surfaces of these members are sealed by injecting non-curing Thiokol sealant under pressure into the milled grooves (channels). Injection ports are spaced approximately six inches apart in these grooves. The fasteners through the tank bonding members and fittings are sealed with O-rings to provide a leak proof construction. Figure 1 illustrates methods of sealing fasteners and channels. Figure 2 shows a composite view of sealing fasteners installed on a fuel cell. The seals must contact the tank membrane in all cases. Where access to both ends of a leaking fastener is available its O-ring should be replaced. Resealing and replacing sealing fasteners cannot be accomplished without access to the inside of the tank. Figure 1 shows the methods of sealing various fasteners and the sealing methods to be used for repairing defective seals, where access to both ends of the fastener is not convenient.

Two low pressure fuel boost pumps, a fuel ejector, and a fuel quantity gaging probe are installed in the fuel cavity bay between the integral tank inboard closure rib (Wing Station 45) and the first wing rib (Wing Station 55). Four flapper check valves, on the lower inboard face of the first wing rib, reduce fuel surge outboard from the boost pumps during aircraft turns or low-wing maneuvers. The venturi-type ejector, utilizing fuel line fuel pressure, provides fuel from the aft end of the fuel cavity bay to the pumps suction area, to ensure fuel supply at the pumps during aircraft climb attitude with low fuel quantity. Four fuel quantity capacitance probes are installed span-wise in the integral tank. Access plates are provided for maintenance of pumps, ejector, and probes. Two drain valves, in the inboard bottom of each integral tank, permit drainage of accumulated water. The filler cap is located at Wing Station 268. A float-vent valve is installed in the outboard closure rib at Wing Station 307. It vents the tank overboard through 15° scarf cut vent tube to the line of flight in the lower camber of the wing. The vent valve has a 5/32 inch hole drilled through the body to prevent excessive pressures from building up in the tank in the event the float valve closes the vent.

Modified System (Aircraft having ASC 125)

All fuel is contained and supplied from eight interconnected fuel tanks installed inside each wing. An integral fuel tank (No. 1) inside the wing inner panel is connected by a flapper valve at Wing Station 307, to seven interconnected, bladder-type cells installed between the ribs of the wing outer panel. No fuel is stowed in the fuselage.

The usable fuel and tank capacities for this system are as follows based on temperature of 59° F and a fuel weight of 6.75 lbs/gal:

Left Tank:	6033.8 pounds	—	893.9 gallons
Right Tank:	6073.0 pounds	—	899.7 gallons
Total:	12,106.8 pounds	—	1793.6 gallons

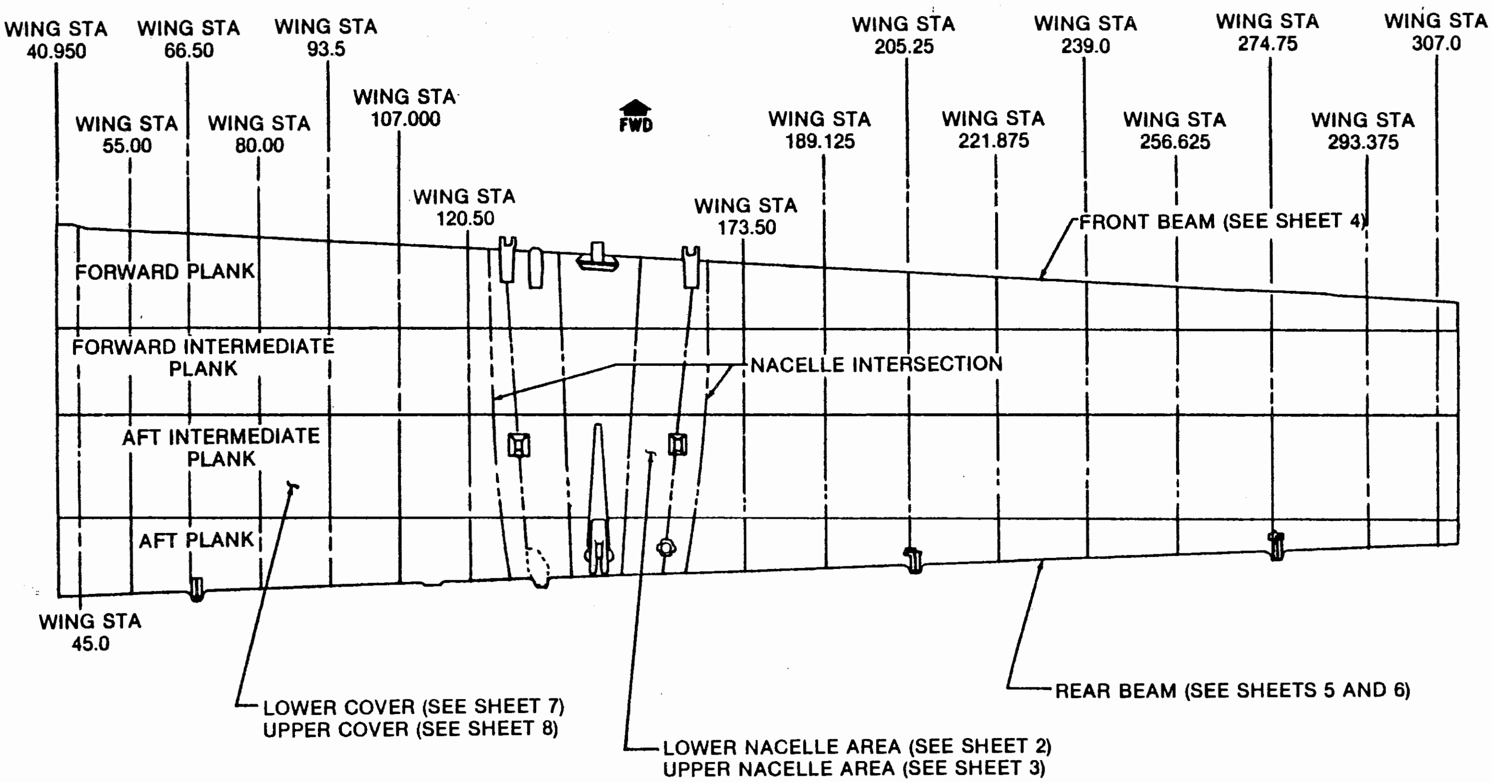
NOTE: Left wing tanks contain less fuel than right wing tanks because of a narrower No. 6 fuel cell, to permit clearance for aileron trim tab cables.

Seven (No. 2 to No. 8) bladder-type, removable fuel tank cells, installed between the ribs, front, and rear beam of each wing outer panel, are interconnected by fuel supply lines through intervening wing ribs. A flapper valve, connecting the integral (No. 1) tank with bladder-type (No. 2) cell at Wing Station 307, prevents fuel surge to the outboard cells during aircraft turning or banking. A flapper valve, on the fuel interconnect fitting

between cell No. 7 and No. 8 prevents fuel loss overboard through cell No. 8 vent. The tops of the cells are anchored to the underside of the wing top skin by cell hangers (four each for cells No. 4 to No. 8, and six each for cells No. 2 and No. 3) and also at the perimeter of the access plate hole for each cell. The sides of each cell are secured to the sides of the cell-containing wing ribs by a washer and nut on each fuel cell interconnect fitting. The forward end of cell No. 2 and cell No. 5 are secured to the wing front beam by the bypass vent line fittings. To protect the cells against abrasive damage, the internal wing surfaces (ribs, front and rear beam, top and bottom skin) are lined with glass fiber-reinforced cloth. Access plates are provided for maintenance and removal of cells and fuel quantity probes. A common fuel drain line connects the outboard bladder-type cells (No. 2 to No. 8) with the integral fuel tank. Two drain valves, in the inboard bottom of each integral tank, permit drainage of accumulated water. Fuel "dumping", or jettisoning in flight, cannot be accomplished. All tanks (No. 1 to No. 8) of each wing, are gravity-fueled on top of each wing, at a fuel filler cap at Wing Station 386.

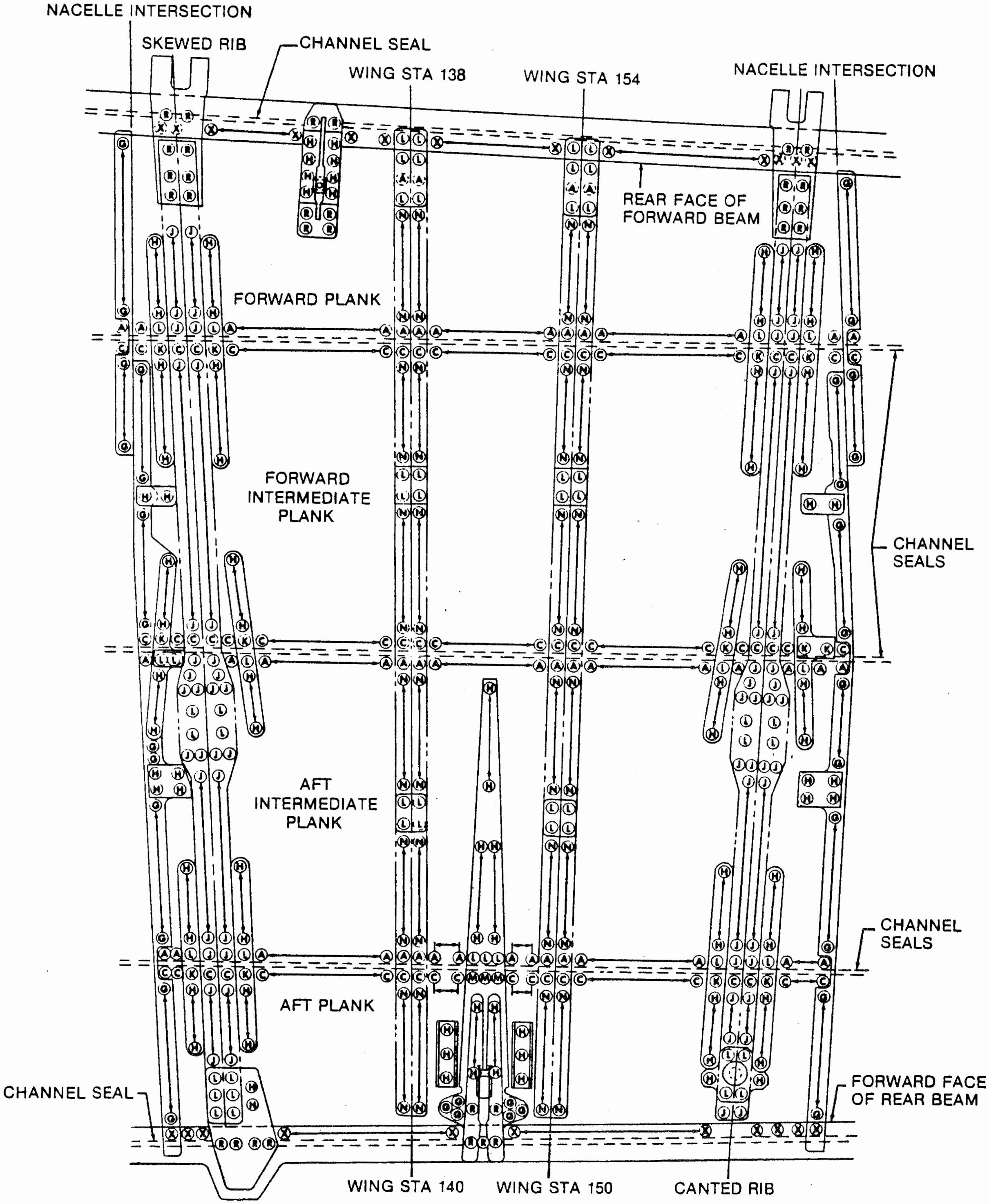
Venting is provided by a bypass vent line connected to cell No. 5. A float valve in the bypass vent line prevents fuel flow outboard from the integral tank. A common vent line, connecting the bladder-type cells of each wing, vents all wing tanks into cell No. 8, which is vented overboard at the bottom of this cell.

Two additional fuel quantity capacitance probes are added, one in tank No. 3 and the other in tank No. 5. They sense the increased fuel quantity, and are paralled with the four probes in the integral tank to give overall fuel load for each wing. The indicators are changed to a higher scale unit.

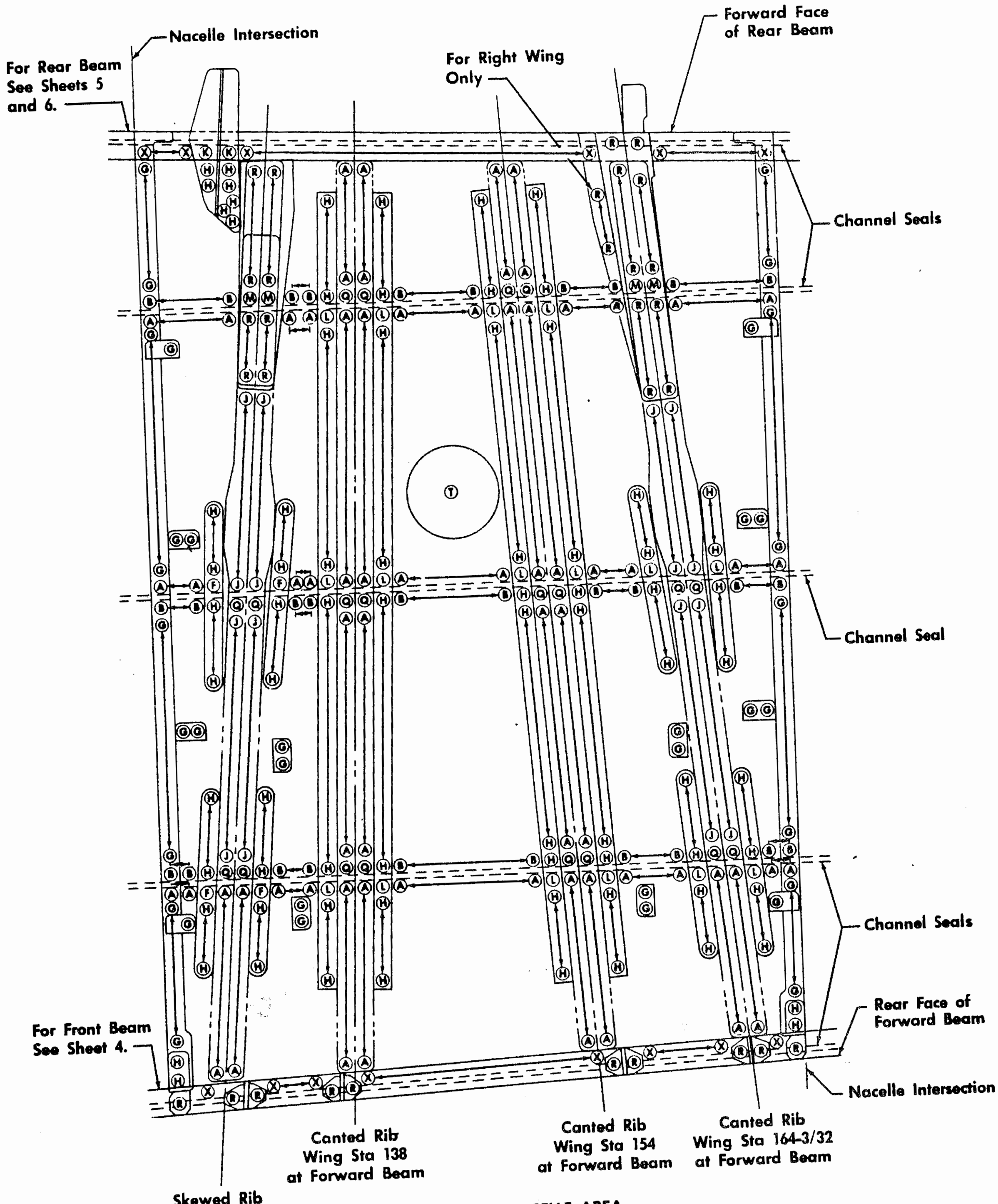


WING INNER PANEL
STA 45.0 TO 307.0
VIEW LOOKING UP

Replacement of Sealing Fasteners
Figure 1 (Sheet 1 of 13).



LOWER COVER NACELLE AREA
Replacement of Sealing Fasteners
Figure 1 (Sheet 2 of 13).



UPPER COVER NACELLE AREA

Replacement of Sealing Fasteners
Figure 1 (Sheet 3 of 13).

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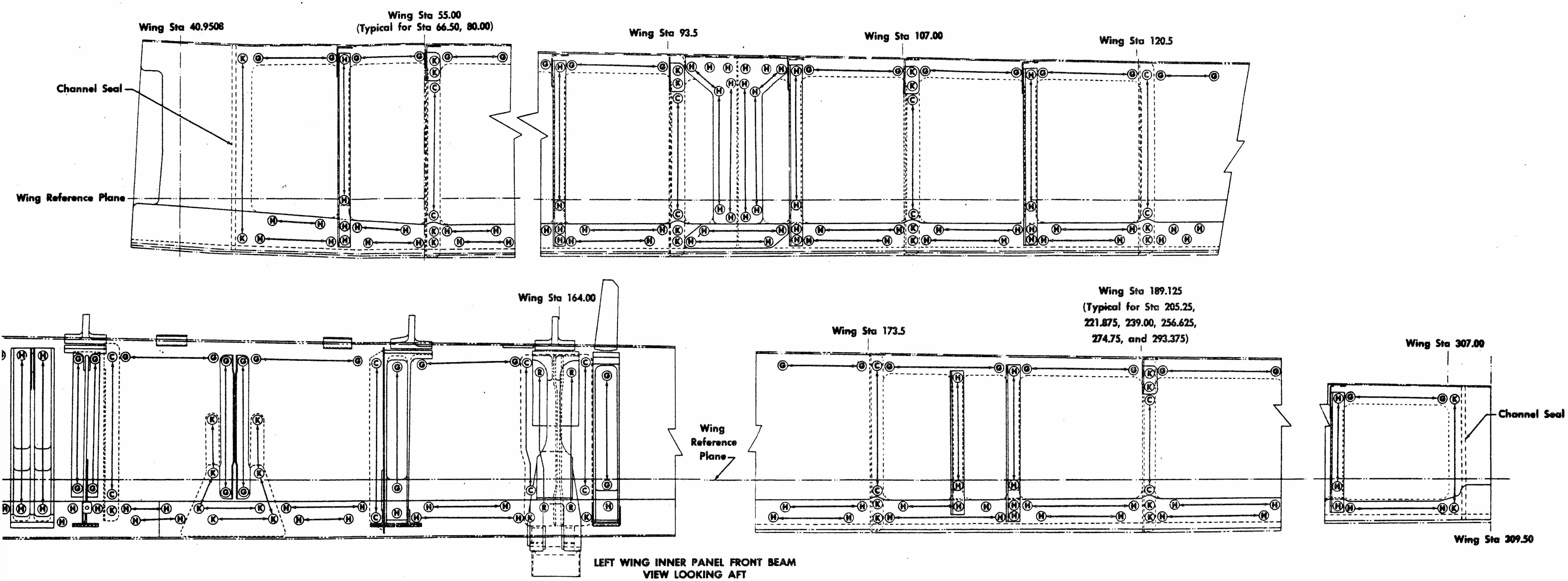
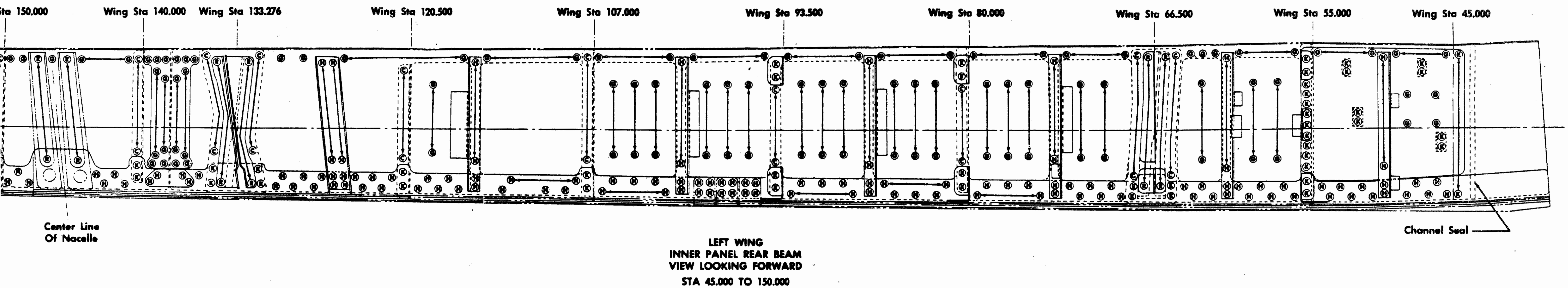
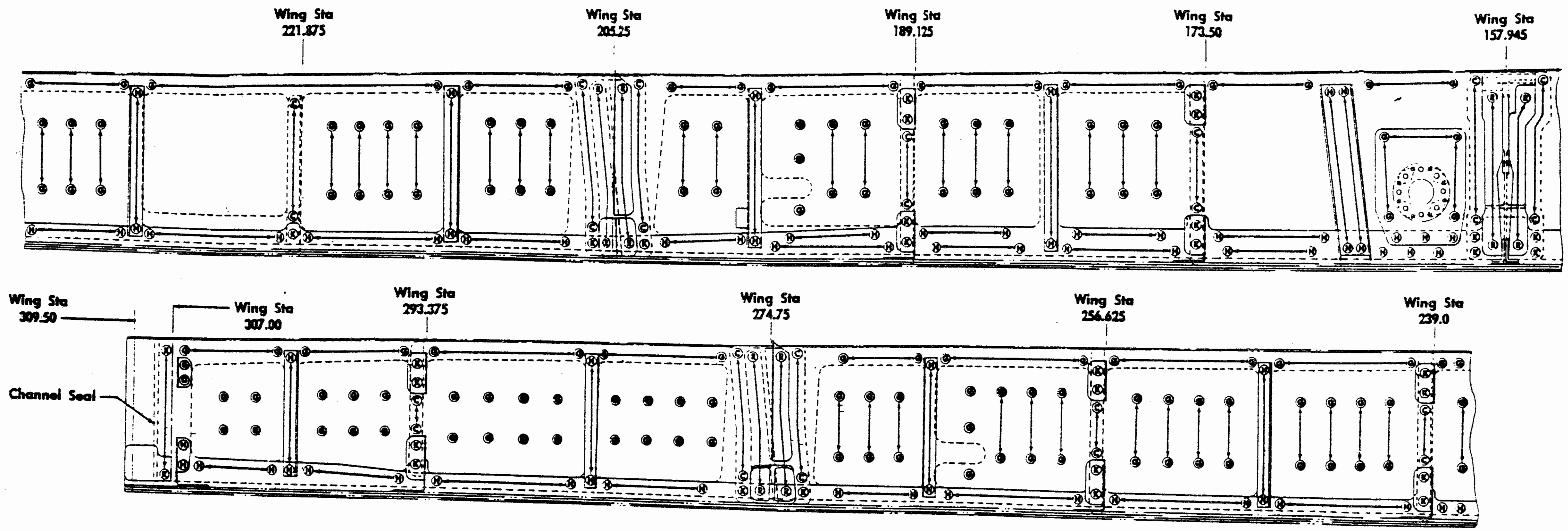


Replacement of Sealing Fasteners
Figure 1 (Sheet 4 of 13).



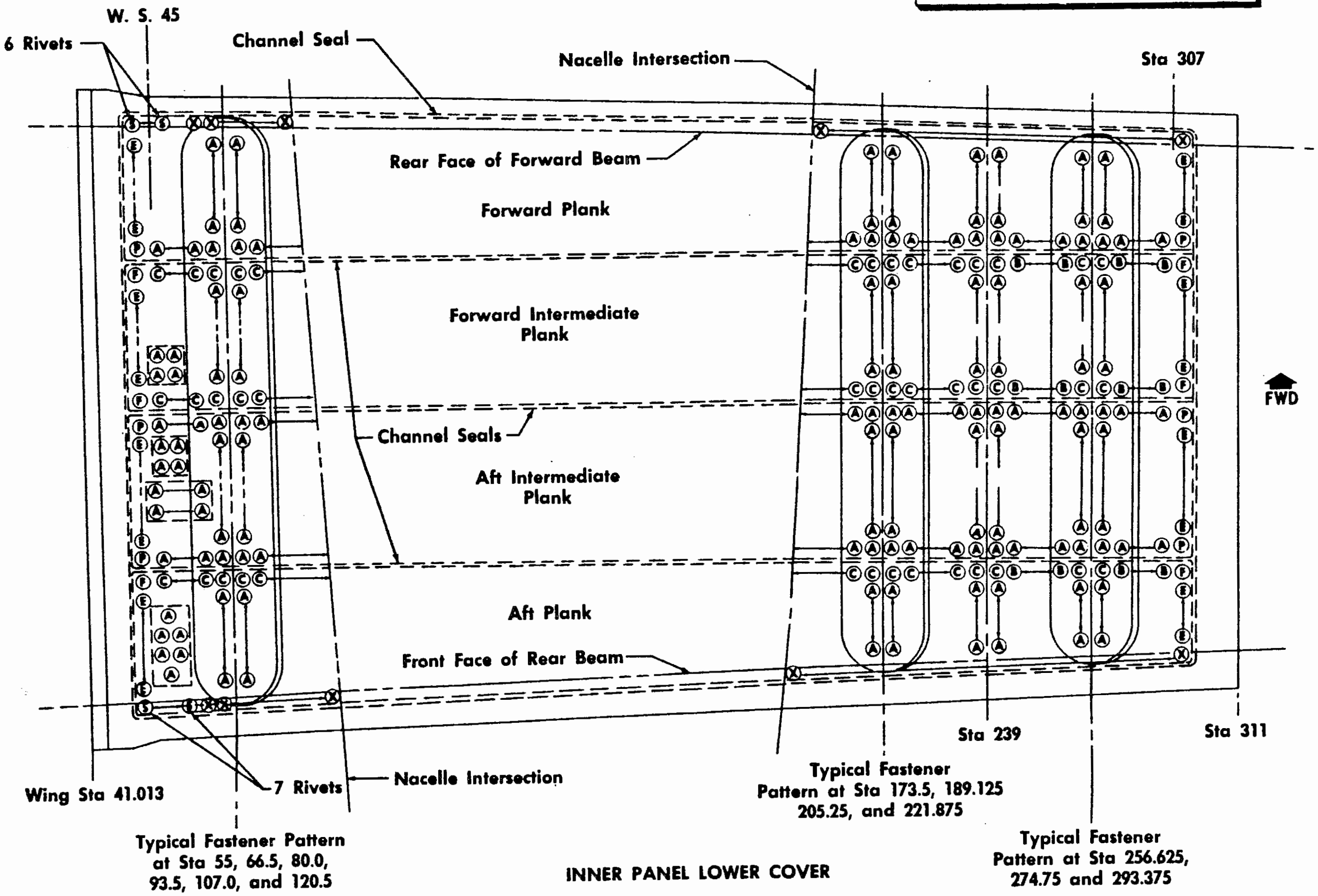
Replacement of Sealing Fasteners
Figure 1 (Sheet 5 of 13).



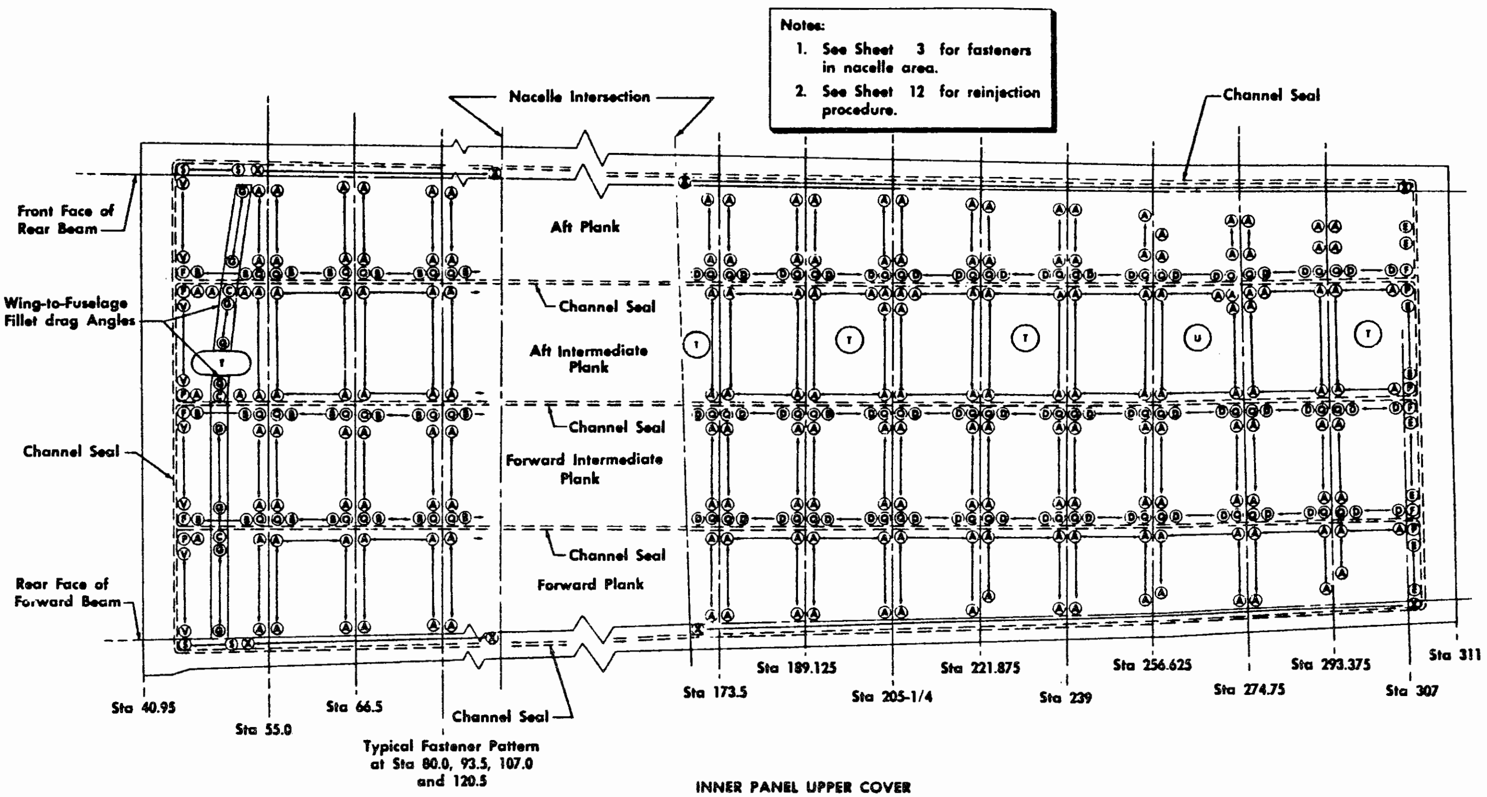
LEFT WING
INNER PANEL REAR BEAM
VIEW LOOKING FORWARD
STA 150 TO 307

Replacement of Sealing Fasteners
Figure 1 (Sheet 6 of 13).

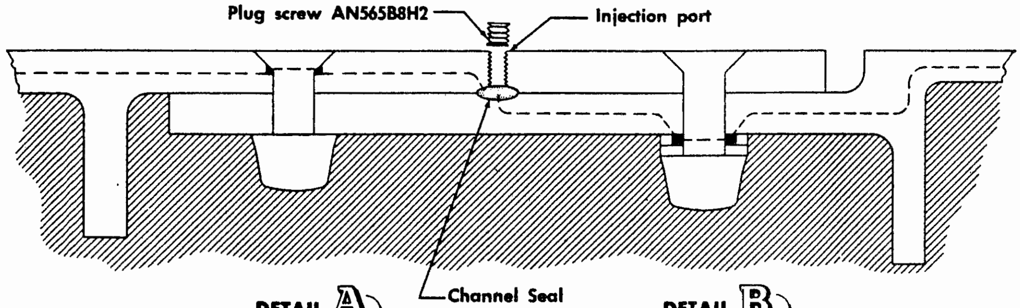
Notes:
1. See sheet 2 for fasteners in nacelle area.
2. See sheet 12 for reinjection procedure.



Replacement of Sealing Fasteners
Figure 1 (Sheet 7 of 13).



Replacement of Sealing Fasteners
Figure 1 (Sheet 8 of 13).

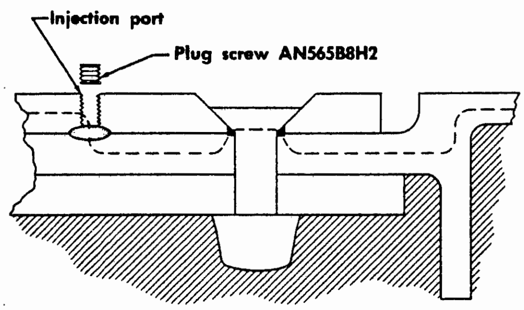


DETAIL A

Repair :
Replace with Rivnut and "O" ring.

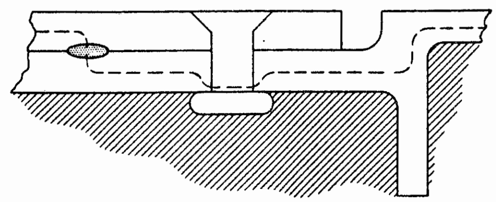
DETAIL B

Repair :
Rework to detail C
Install Rivnut and "O" ring. See detail AE for countersink and counterbore instructions.



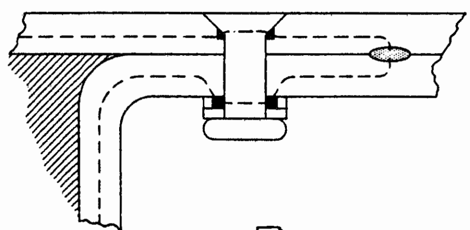
DETAIL C

Repair :
Replace with Rivnut and "O" ring.



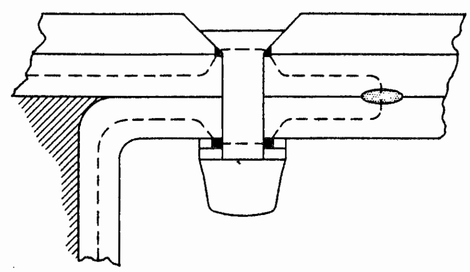
DETAIL D

Repair :
Rework to detail C .
Install Rivnut and "O" ring. See detail AE for countersink and counterbore dimensions.



DETAIL E

Repair:
Rework to detail X.
Install Rivnut and "O" ring. See detail AE for countersink, counterbore dimension.

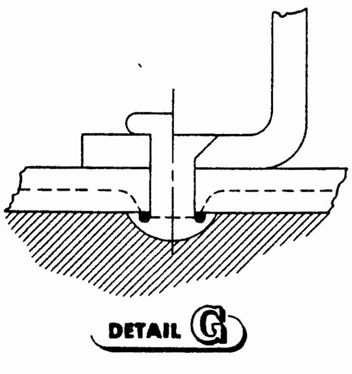


DETAIL F

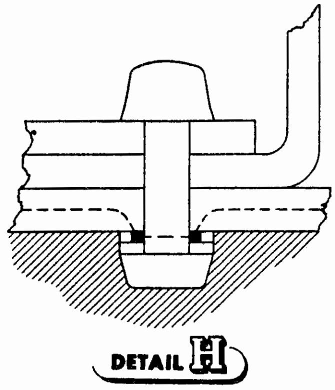
Repair:
Replace with NAS 1203 screw, "O" ring, Stat-O-Seal, washer and nut.

Fuel

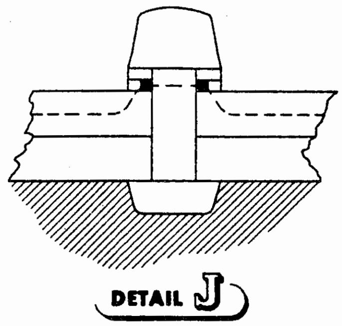
Replacement of Sealing Fasteners
Figure 1 (Sheet 9 of 13).



Repair:
Rework to detail C . Install Rivnut and "O" ring. See detail AE for countersink and counterbore instructions.

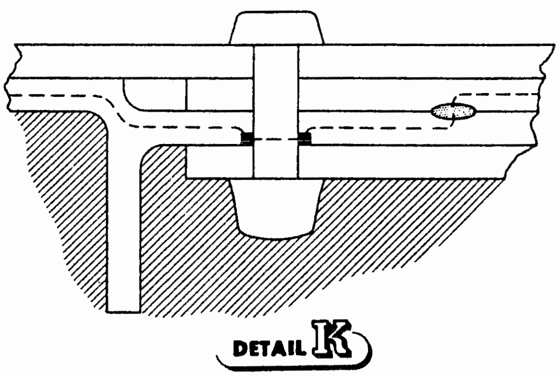


Repair:
Rework to detail M . Install Rivnut and "O" ring. See detail AA for bushing repair information.

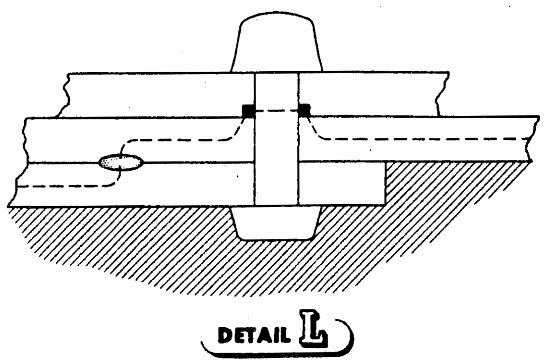


Repair:
Install Rivnut with Stat-O-Seal and washer same as in detail J .

Fuel



Repair:
Replace same fastener or with Rivnut. If leak persists, rework to detail M and install Rivnut and "O" ring. See detail AA .

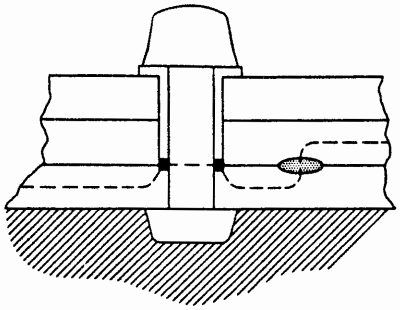


Repair:
same as for detail K

Fasteners used in fuel tanks are; MS20426AD5, MS20426AD6, G472AD5, and G472AD6 aluminum rivets; ASCT509-T6, ASCT509-T8, R3014-T6, SALP-T6 and SAL100-T8 steel huck lock bolts; HS47-6 Hi-shear rivet, and B1988,CH6B, and H6B B. F. Goodrich steel rivnuts. "O" rings used are G474, which have .025 diameter cross section and are used in counterbores shown in details AD, AE, AF. GS13H "O" rings of .050 diameter cross section are used in details K, L, M, S, AA. 600-001-10 and 600-015-10 Stato-O-Seals are washers with an inner ring of moulded rubber and are shown in details B, E, F, H, J and P.

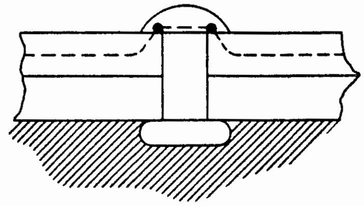
Replacement of Sealing Fasteners
Figure 1 (Sheet 10 of 13).

Fuel



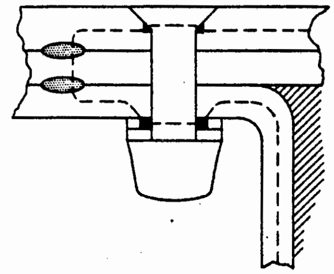
DETAIL M

Repair:
Replace "O" Ring.
Reinstall bushing
and install Rivnut.



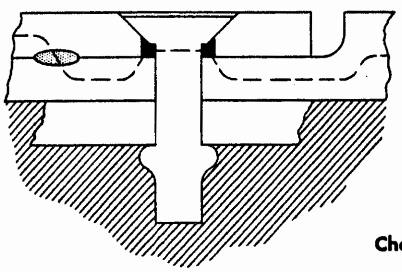
DETAIL N

Repair:
Install Rivnut, Stat-O-Seal
and washer as shown
in detail J.



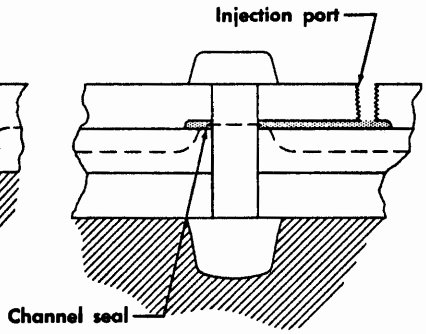
DETAIL P

Repair:
Countersink, counterbore
for NAS 1203 screw. Use
"O" ring, Stat-O-Seal and
washer as per detail "P."
See detail AE for
countersink, counterbore
instructions.



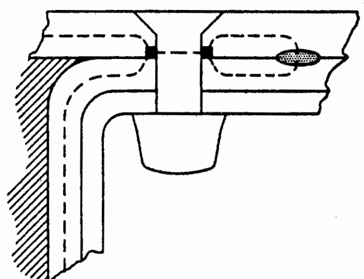
DETAIL Q

Repair:
Replace with GS13H-8
"O" ring and B1988 Rivnut.



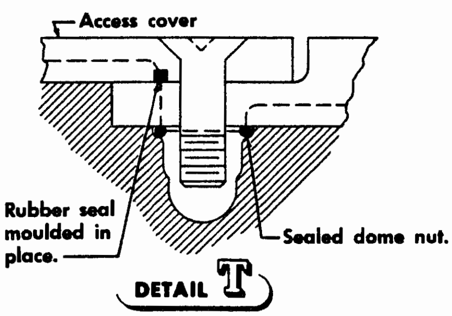
DETAIL R

Repair:
Reinject channel seal.
(Do not remove fasten-
ers.) Dow Corning
Q94-011 or Products
Research No. 703 Sealant
may be used in place of
Presstite No. 591.1.



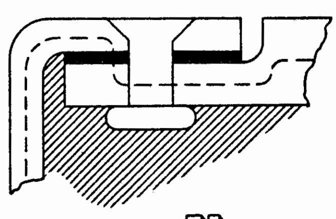
DETAIL S

Repair:
Replace fastener with
NAS1203 screw. If leak
persists, rework as
per detail AB.



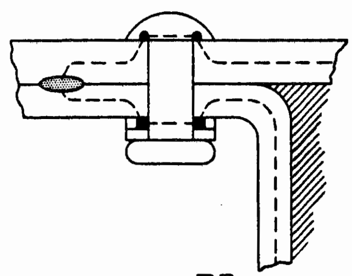
DETAIL T

Repair:
Apply thin layer of
Presstite No. 591.1
between faying
surfaces.



DETAIL U

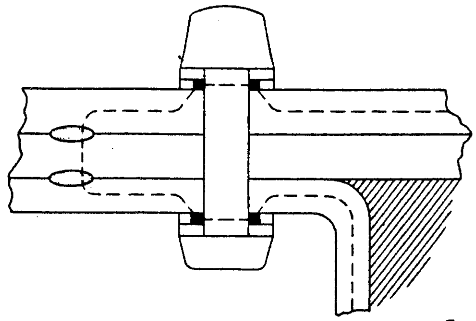
Repair:
Replace gasket.

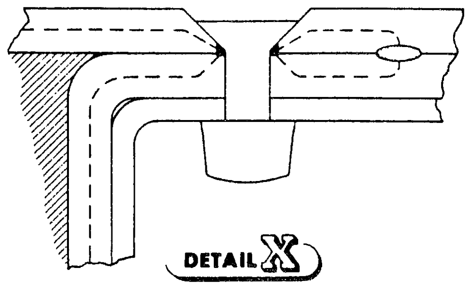


DETAIL V

Repair:
Same as for detail E.

Replacement of Sealing Fasteners
Figure 1 (Sheet 11 of 13).





DETAIL X

Fuel

Repair:
Replace with NAS 1203 screw and "O" ring, washer and nut.

DETAIL W

Repair:
This detail exists in only 3 places at inboard end of upper skin. Replace seals as per detail W. Access may be obtained through lower cover wing center section and dihedral brake rib.

Reinjection Procedure

1. Use Grover Smith Gun No. 223 (Grover Smith Manufacturing Co; San Gabriel, California) and sealant gun adapter, Grumman Tool No. 159GT1003.
2. Use an air supply of 30 to 40 PSI.

CAUTION

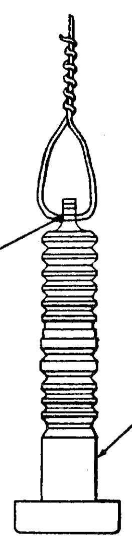
Do not use unregulated shop air.

3. Remove three (3) consecutive plug screws.
4. Inject Dow Corning Q94-011 or 591.1 sealant (Presstite-Keystone Engineering of America Products Division, Marietta Co. St. Louis, Missouri) into center hole until it flows from adjacent holes. Maintain flow of sealant through sealing grooves at approximately 6 inches per minute.

CAUTION

The maximum allowable injection pressure at the nozzle is 700 PSI.

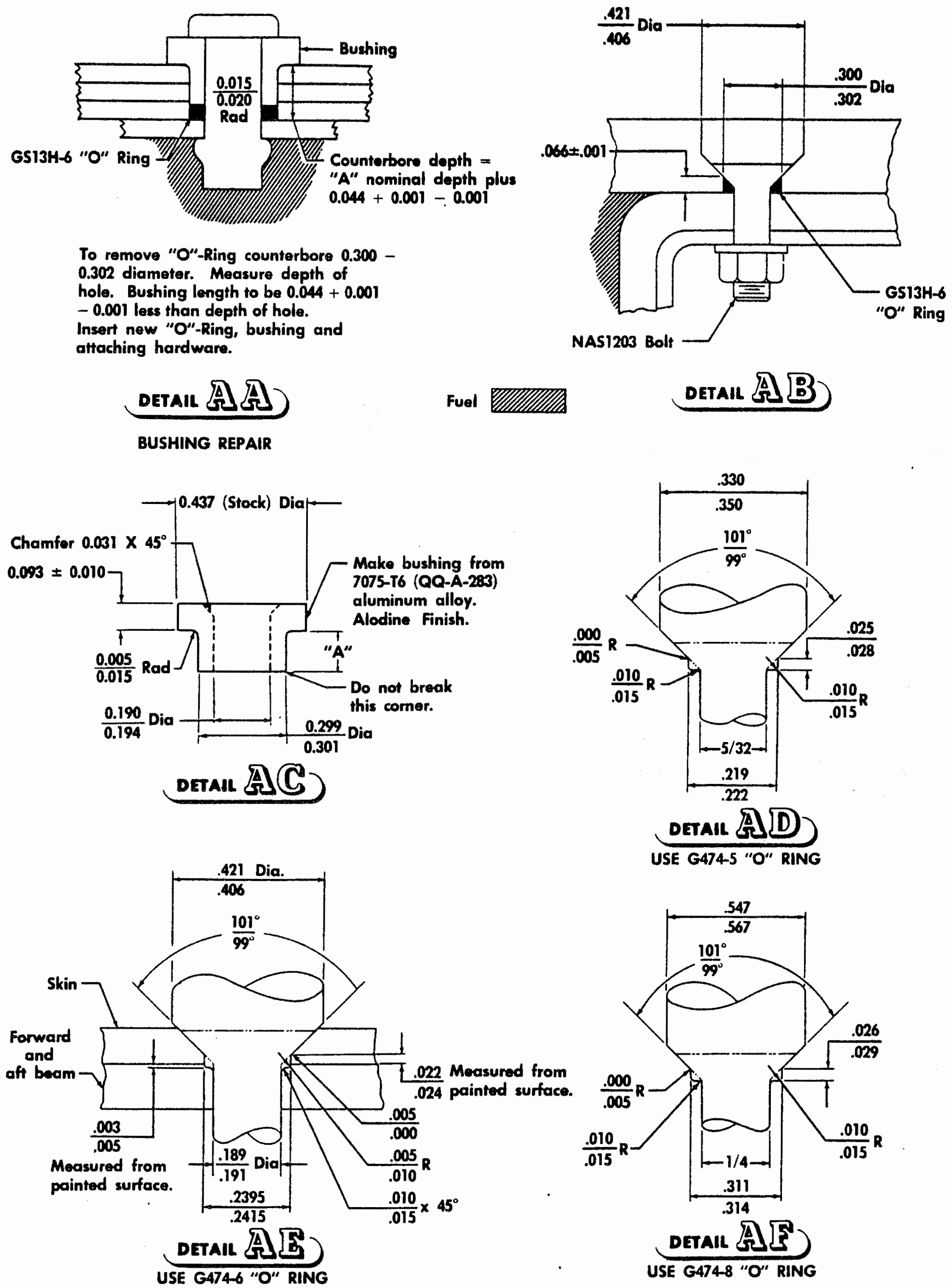
5. Install plug screws.
6. Dow Corning Q94-011 or Products Research No. 703 Sealant which is more viscous than Presstite No. 591.1 may be used to advantage to repair leaks on fitting whose attachment fasteners are sealed with sealant as per detail R.
7. The "Fishing Method" of replacing protruding head fasteners can be used to advantage. After removing the defective fastener, insert a wire through the fastener hole toward an access hole. Attach the wire to fastener as per detail Y. Pull fastener thru hole and install collar. This method may be used with Huck lock bolts and bolts, where the ground flats may be held by a wrench while tightening nut.



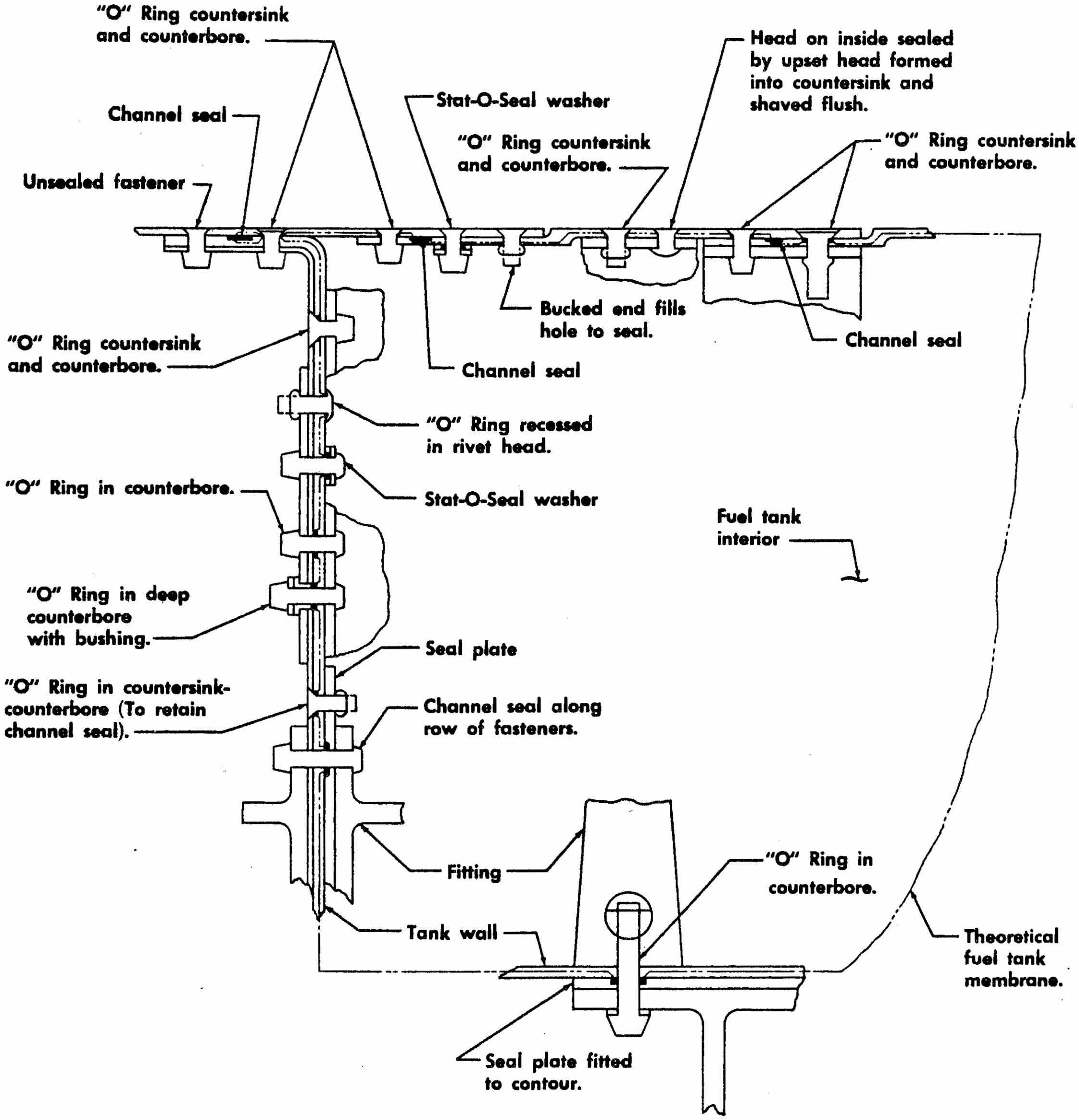
Method of Attaching Fishing Wire

DETAIL Y

Replacement of Sealing Fasteners
Figure 1 (Sheet 12 of 13).



Replacement of Sealing Fasteners
Figure 1 (Sheet 13 of 13).



Composite of Sealing Fasteners
Figure 2.

FUEL TANKS — MAINTENANCE PRACTICES

1. Reduction of Fuel Tank Vapor Hazards

A. Prior to working on fuel tanks, defuel them. Defueling should be done outdoors, with the aircraft at least feet from hangars or other aircraft. Residual fuel which cannot be drained by normal defueling must be drained from the tanks by opening all tank water drain valves.

WARNING: THERE IS LITTLE OR NO CONTROL OVER FUEL VAPORS BEING RELEASES. PROCEDURES FOR REMOVAL OF FUEL VAPORS MUST BE STARTED IMMEDIATELY.

IT IS RECOMMENDED THAT ADEQUATE FIRE FIGHTING EQUIPMENT BE STANDING NEAR THE AIRCRAFT DURING FUEL VAPOR REMOVAL AND UNTIL ALL MAINTENANCE IS COMPLETED.

There are three methods which are to be followed during aircraft ground handling in the interest of fire explosive prevention when it is desired to reduce the flammable vapor hazards of aircraft fuel tank atmospheres when such tanks contain or did contain a volatile fuel. The methods are as follows:

(1) **Inerting**: Inerting, as used herein, means the use of an inert gas to render the atmosphere of an enclosure non-explosive or non-flammable. Inerting, in effect, reduces the oxygen content of the air in the tank vapor space below the lowest point at which combustion can occur by replacing the oxygen in air with an inert gas.

NOTE: Some jet fuels are known as safety fuels but their flash points still fall between 100°F and 140°F.

(2) **Air Ventilation**: Air ventilation, as used herein, means to pass undiluted air (air not containing flammable vapors or inert gases) through an aircraft tank to render the atmosphere of the tank more suitable for human occupancy and to reduce the amount of flammable vapors in the tank to below the lower explosive limit of the fuel vapors involved. It is recognized that at sometime during, and possibly after, air ventilation, the tank may contain a flammable vapor-air mixture. During such periods a fire and explosion hazard exists which requires the elimination of ignition sources within the vapor-hazardous area.

(3) **Purging**: Purging, as used herein, means to remove the flammable vapor atmospheres, and any residual fuels capable of producing flammable vapors, in the tank and connected distribution lines so that subsequent natural ventilation will not result in the reinstatement of a flammable atmosphere unless or until a flammable liquid is again introduced into the tank or its connected distribution lines.

(4) **Inert Gases**: Nitrogen is a satisfactory medium and is also normally available at locations where work of this type is conducted. Greater quantities of nitrogen are required than of carbon dioxide to secure the desired inerting effect, but, generally nitrogen can be retained more easily in a sealed tank than carbon dioxide because of its lighter weight. Periodic checks (not to exceed 48 hours), should be conducted to ensure the maintenance of an inert atmosphere, particularly in non-metallic fuel cells, or in lieu of this, a positive pressure (within the safe working pressure of the tank) should be maintained on the inerted tank from the inert gas supply.

B. Inerting Procedure — General

(1) Aircraft should be parked no closer than 100 feet from hangars or other aircraft and "HAZARDOUS AREA" signs should be located around aircraft.

(2) All open flame and spark producing equipment or devices within the vapor hazard area should be shut down and not operated during the inerting procedures.

(3) Aircraft should be grounded to guard against the accumulation of static electrical charges.

(4) Whenever possible, the aircraft on which the inerting is being accomplished should have its electrical system deenergized and batteries removed.

(5) Electrical equipment used in the vapor hazard areas should be approved for use in hazardous locations as defined by the National Electric Code.

(6) All mechanics and other people, except those engaged in the inerting procedures, should remain clear of the aircraft, and no other maintenance activities should be conducted on the aircraft or tank until after the inerting has been accomplished.

(7) A suitable warning sign should be placed in a conspicuous location on aircraft to indicate that the fuel system has been inerted.

(8) Adequate portable fire extinguishing equipment should be provided for the hazards involved. Portable or mobile equipment should include a quick smothering type extinguishing agent (such as carbon dioxide or dry chemical) plus a permanent smothering type extinguishing agent (such as foam). The amount and nature of such equipment will be at the discretion of the Safety Department.

C. Pressure Siphon Inerting Procedure.

(1) The following equipment is required for use in performing the pressure siphon inerting procedure:

- Nitrogen gas bottle
- Pressure reducing regulator valve
- A needle type flow control valve
- A calibrated differential pressure gage or manometer
- An aircraft fuel servicing truck for refueling and defueling aircraft in a safe manner.

CAUTION: SPECIAL PRECAUTION MUST BE TAKEN TO AVOID DISCHARGE OF LIQUID NITROGEN.

(2) Fill main fuel tank to capacity.

NOTE: This renders the fuel tanks substantially free of air and eliminates any possibility of a fuel vapor-air mixture within the flammable range of the fuel vapor.

(3) Connect fuel truck to drain in rear of nacelle.

(4) Connect nitrogen supply line to overboard vent fitting and tee in a calibrated pressure gage of manometer.

NOTE: The calibrated pressure gage or manometer should be installed in the nitrogen supply line as close as possible to the aircraft inlet vent.

(5) Open pressure reducing valve on nitrogen bottle.

(6) Apply 3.0 $\pm$ 0.25 psi to overboard vent by adjusting needle control valve. This defuels main fuel tank into fuel truck.

NOTE: A continuous supply of nitrogen at 3.0 psi $\pm$ 0.25 must be maintained to permit continuous siphoning and inerting of the airspace created by the defueling operation. Do not open water drain valves.

(7) When tank is empty, disconnect nitrogen supply from overboard vent.

(8) Insert a solid plug, and cover it with sealing tape.

(9) Attach a red streamer to plug, to defueling drain, and to fuel gravity filler when inerting is complete.

WARNING: THE MAINTENANCE OF AN INERT ATMOSPHERE WITHIN THE TANK DEPENDS UPON RETAINING THE INERT GAS APPLIED TO THE TANK. IF ANY PART OF THE FUEL SYSTEM CONNECTED TO THE INERT TANK IS OPENED, THE TANK CANNOT BE CONSIDERED INERTED THE VAPOR CONTENT MUST BE TESTED. IF THE OXYGEN CONTENT IS ABOVE MINIMUM REQUIREMENTS, ADDITIONAL NITROGEN SHOULD BE ADDED TO REINSTATE THE SAFE TANK ATMOSPHERE.

NOTE: Periodic checks (not to exceed 48 hours) should be conducted to ensure the maintenance of an inert atmosphere, particularly in non-metalic fuel tanks.

D. Air Ventilation

CAUTION: AIRCRAFT ELECTRICAL CIRCUITS WHICH ARE IN VAPOR HAZARDOUS AREAS SHOULD BE DE-ENERGIZED.

NOTE: As mentioned in paragraph 1.A., air ventilation of fuel tanks is recommended for the sole purpose of rendering the atmosphere in an aircraft fuel tank more suitable for personnel to enter the tank area for inspection or work purposes. Air ventilation is not a method of inerting an aircraft fuel tank, and this distinction must be clearly understood.

(1) Precautions Against Exterior Ignition Sources: All precautions stated in paragraph 1.B. (Inerting Procedure - General) should be adhered to.

(2) Preparation for Air Ventilation

(a) Prior to conducting work on fuel tanks, it is necessary to defuel the tank or tanks to be inspected or maintained. Such defueling operations should be done outdoors. Aircraft should be parked no closer than 100 feet from hangars or other aircraft and "hazardous area" signs should be located around aircraft. Residual fuel which cannot be withdrawn by normal defueling procedures must be drained from the tanks by opening all water drain valves. With the opening of the tank, air ventilation procedures should be immediately instituted. This operation is inherently very hazardous as there can be little control over the vapors released. Residual fuel must be retrieved in the safest possible manner and the fuel prevented from excessively wetting the fuselage or dripping to the ground to form pools. This may be accomplished by siphoning out of the tank or by manually sponging or "mopping-up" from tank points or where trapped. Prior to entry into the tank to conduct any manual operation therein, tests should be conducted to determine that a flammable vapor air mixture does not exist or that toxic quantities of vapors are not present (unless adequate respiratory protection is provided and worn). Obviously, as much of this operation as is possible should be conducted outdoors to gain the maximum advantages of free circulation and the elimination of ignition sources.

(b) When air ventilation is done in an enclosed hangar and where a closed ventilating system to discharge vapors from tanks to outside the hangar is not used and tank vapors are discharged into the hangar, tests should be conducted to determine that the presence of such fuel vapor laden air in the enclosed hangar does not constitute a hazard, under the worst conditions that can normally be anticipated.

E. Air Ventilation Procedures.

CAUTION: WHEN ONLY AIR EXHAUST IS USED, BE CAREFUL TO PREVENT A BUILDUP OF NEGATIVE PRESSURE WHICH CAN RESULT IN TANK COLLAPSE. WHEN A BLOWER IS USED, PRESSURE OUT SHOULD BE BALANCED TO PREVENT AN ADVERSE EFFECT ON TANK STRUCTURE.

(1) Completely drain tank or tanks.

(2) Establish air circulator in tank.

NOTE: Proper air circulation rids the tank of hazardous quantities of fuel vapors.

(3) Perform combustion vapor tests in tank corners to ensure that no vapor pockets remain.

(4) Continue air ventilation during entire period that tanks are open and maintenance is being performed.

NOTE: If flammable vapors (as from cleaning solvents) are added, adjustments to the volume of air circulation may be required to maintain proper ventilation.

F. Purging Tanks.

The procedure for purging tanks is identical to the Air Ventilation Procedure, except that longer periods of air ventilation are required.

2. Sealing Procedures

(See Figure 1)

If a leak occurs in any channel seal groove, the tank can be quickly resealed by the reapplication of the Thiokol rubber sealant. The tool used for the sealant procedure is Grover Smith Sealant Gun. Model No. 223. This gun is used in conjunction with a fuel cell sealant gun adapter (Gulfstream Tool 159-GT-1003). A portable Semco Sealant Gun No.250-12 is also used at Gulfstream for this purpose. A typical resealing procedure is as follows:

A. Remove all screw plugs from the channel seal groove in the area where tank is leaking. (Remove at least one plug from each end beyond leaking area.)

B. Inject Presstite 591.1G sealant into outside recess hole. The practice of heating the sealant while in the gun in order to make the Presstite 591.1G more pliable is not an acceptable procedure. On the contrary, it can prevent proper tank sealing. Presstite 591.1G expands with heat, and if injected while warm tends to contract on cooling, thus creating air pockets in the channel groove. The air pocket may cause an eventual fuel tank leak. Fresh Presstite 591.1G is soft and pliable enough to flow at the recommended pressure and does not require heating above normal temperature. Sealant which is too hard to inject under normal conditions should be discarded. Care should be taken to tightly pack the sealant in the gun. Otherwise air bubbles may be introduced in the channel seal groove.

CAUTION: DO NOT USE MORE THAN 40 PSI AIR PRESSURE WHEN USING GROVER SMITH SEALANT GUN, MODEL 223. STRUCTURAL DAMAGE TO THE SKIN CAN RESULT OTHERWISE.

PRESSTITE 591.1G HAS THE SAME COLOR AND GENERAL APPEARANCE OF ZINC CHROMATE PASTE. ZINC CHROMATE PASTE IS NOT A SUBSTITUTE FOR PRESSTITE 591.1G AND CARE SHOULD BE EXERCISED TO ENSURE THAT THE PROPER SEALANT IS USED.

DO NOT USE PR-703 OR DOW CORNING Q94-011 WITH 60 PSI PRESSURE ANYWHERE ON THE WING INTERNAL FUEL TANK EXCEPT FOR THE LANDING GEAR RETRACT CYLINDER FITTING AND TWO (2) AFT UPPER ENGINE MOUNT FITTINGS BECAUSE OF THE POSSIBILITY OF STRUCTURAL DAMAGE.

For sealing landing gear retract cylinder fitting and (2) aft upper engine mount fittings, use Dow Corning Q94-011 or PR-703 sealant. Sealant gun pressure may be raised to 60 PSI maximum since sealant is more viscous than Presstite 591.1G.

C. As new sealant is pumped into hole, old sealant will be forced out of adjacent hole. Old sealant will maintain its shape and can be used as a guide to measure how far new sealant has been forced through channel groove by guiding ejected sealant in direction of sealant gun. Area between (2) recess holes can be considered resealed when sealant extends beyond hole into which new sealant is being applied or when new sealant is ejected from the adjacent hole.

D. Plug first hole and move to next hole. Repeat Step C. until all open holes have been sealed.

WARNING: PROPER PRECAUTIONS SHOULD BE OBSERVED WHEN USING METHANOL OR MIBK TO PREVENT INHALATION AND CONTACT WITH THE SKIN.

E. Remove any sealant from wing surface with a cloth dampened with methanol. Water methanol solution from water methanol tanks can be used for this purpose. Methyl isobutyl ketone (MIBK) can be used as a substitute for methanol. Use of methyl ethyl ketone (MEK) is not recommended for this procedure.

F. The wing beams, ribs, and planks are joined together with fasteners which must also be sealed to prevent the fuel from leaking out of attachment holes. Sealing the individual fuel tank fasteners in conjunction with channel seal groove provides means for sealing fuel tanks. Replacement of sealing fasteners is shown in Figure 1

G. Plank filler compound is inserted between wing planks to help provide a smooth airflow over wings, but does not aid in preventing fuel tanks from leaking. If plank filler requires replacement or repair, proceed as follows:

(1) Clean surface of skin with methyl isobutyl ketone (MIBK).

(2) Fill cavities with MIL-S-8802 compound. The mixture is forced into cavity with a piece of micarta, wood, or other tool which will not damage surrounding area. Mixing directions are provided on can.

CAUTION: DO NOT USE CROCUS CLOTH OR OTHER METALLIC ABRASIVE COMPOUND. CORROSION MAY RESULT FROM IMBEDDED PARTICLES.

(3) After compound dries and hardens, use aluminum wool, sandpaper, or emery cloth to restore original configuration. Paint as required.

H. Fuel Tank Leakage Limits and Repair Requirements

(1) **Fuel Tank Leakage criteria**: Fuel leaks are divided into five classes based on leakage rate. These leakage classes are Slow Seep, Seep, Heavy Seep, Drip and Running Leak. The description of these classes is as follows:

(a) **Slow Seep**: A fuel leak that wets an area around source up to 3/4 inch in diameter.

(b) **Seep**: A fuel leak that wets an area around source up to 1 1/2 inches in diameter, but does not run or drip.

(c) **Heavy Seep**: A fuel leak that wets an area around source up to 3 inches in diameter, but does not run or drip.

(d) **Drip**: Any fuel leak that causes dripping from aircraft structure at a rate up to 4 drops per minute.

(e) **Running Leak**: Any fuel leak that allows fuel to drip or run from aircraft structure at a rate of 4 drops per minute or greater.

NOTE: The size of the wetted area described for Slow Seep, Seep, or Heavy Seep is limited by evaporation of the fuel. When the fuel evaporates from the wetted surface at the leakage rate, the wetted area will not enlarge.

(2) Procedures for Determining the Class of Leaks

(a) Clean area around suspected leak with any approved external skin cleaner.

(b) Allow thirty minutes to 1 hour for leak to wet exterior surface then blow talcum powder on area. Talcum will outline wet area clearly.

(c) Measure average diameter of wetted area and establish leak class as described under Step H. (1).

(d) Observe all leaks for possible run paths and drip points.

CAUTION: ANY LEAK MAY BE AN INDICATION OF A STARTING STRUCTURE FAILURE CONSEQUENTLY THE SOURCE OF A LEAK MUST BE LOCATED AND THE CAUSE DETERMINED PRIOR TO RESEALING.

(3) All fuel leaks in fuel tanks, regardless of location, subject aircraft to following operational limitations.

(a) **Slow Seep, Seep, and Heavy Seep**: Clean surface, record leak location, and inspect frequently.

(b) **Drip**: Return to a suitable maintenance base, investigate cause and repair before further flight.

(c) **Running Leak** Ground aircraft and repair immediately.

3. Fuel Tank — Cleaning Procedures

A. General

As a result of reports on corrosion in integral fuel tanks for turbine fuel, full scale investigations were conducted by many agencies to determine the cause and find a deterrent for this condition. The causative agent has been recognized as a micro-organism type of contamination which will promote a breakdown of the protective coating and initiate a corrosion reaction with aluminum in the presence of water, salts and rust. Certain chemicals retained in the fuel after refinement will also contribute to the overall condition.

While these contaminants do not generally create a problem by themselves, they are usually found in a combined state in storage tank bottoms and can be easily injected into an aircraft fuel cell by improper handling techniques or inadequate pumping and filtering equipment.

Under favorable conditions, the micro-organisms will thrive and propagate at the fuel water interface. Eventually, they become firmly rooted to the structure surface and hold the accompanying contaminants in place. By-products of metabolism in conjunction with the iron in the rust subsequently establish a galvanic cell and the aluminum becomes the corroding anode.

Although considerable effort is being expended towards improved coating systems and biocidal additives, the most practical and effective way of combating this particular problem is to maintain and provide clean, dry fuel.

The protective system presently incorporated in the Gulfstream I fuel cell will prevent corrosion if incipient contamination is introduced into the system. However, unless prompt action is taken to flush and remove the contaminants, they will eventually spread and anchor to the surface. When this occurs, much more extensive and costly repair procedures are necessitated.

B. Microbial Contamination Control

(1) Fuel only with clean fuel (water and salt free). .

(2) Use water drains regularly; after each flight, just prior to takeoff if fueled over 2 hours.

(3) Inspect every 3 months and clean if necessary.

(4) Anti-microbial additive Methyl Cellosolve may be used in accordance with the following schedule:

(a) **Initial treatment to kill**: 0.25 % by volume (this solution must be drained before engine operation or it may be diluted to a concentration not exceeding 0.15 % by volume and burned off in the engine, provided it is not contaminated with microbiological debris.)

(b) **Continuous treatment for sterilization maintenance**: 0.15% by volume (maximum).

(5) Biobor JF anti-microbial additive may be used in accordance with the following schedule:

(a) **Initial treatment to kill**: 270 PPM maximum (4 ounces per 1,000 pounds of fuel).

(b) **Continuous treatment for sterilization maintenance** 135 PPM maximum (2 ounces per 1,000 pounds of fuel). It is recommended this concentration be achieved by premixing in storage, and not by direct addition to aircraft tanks.

C. General Cleaning Procedure

The following procedure is recommended for fuel tanks as a general cleaning method where sediment, flakes of tank liner material, and other non-microbial contaminants are present. A conscientious effort should be made to thoroughly clean and rinse every inch, corner, seam and relatively inaccessible area of the tank interior. Use boroscopes and mirrors to see into hidden areas and long handled brushes as needed to facilitate cleaning.

(1) Defuel aircraft, see Section 28-0-1.

(2) Remove tailpipe and heatshield. Remove all access covers from tops of the wings.

NOTE: Provide liquid tight covers for openings left by removal of components.

(3) Remove fuel quantity probes from each wing through the rear beam. Remove associated wiring and fittings.

(4) Remove fuel pumps from each wing.

(5) Remove fuel ejectors and adjacent fuel lines.

(6) Seal off all electrical connectors which might become wet.

(7) Prepare a cleaning solution as follows. Mix material conforming to MIL-C-25769 with warm (100°F to 150°F) water. Use one part cleaner to five parts water by volume.

NOTE: The use of powdered detergents and other cleaners is not recommended unless ingredients are known. Many contain abrasives and inert filler which may leave sediment, and

excessive amounts of foaming agents which may present difficulties when rinsing tanks out.

(8) Through access openings inboard of Wing Station 307, distribute 25 gallons between front and rear beams. Distribute remaining solution between front and rear beams through access openings inboard of engine nacelles.

CAUTION: DO NOT ALLOW CLEANING SOLUTION TO DRY IN TANK.

(9) Working from outboard to inboard, clean interior of the tank, section by section. Pay particular attention to lower planks, and to recesses in lower plank splices and straps. Flush each area with warm (100°F to 150°F) water after cleaning to remove sediment and cleaning solution.

NOTE: Bristle (fiber) brushes may be used on stubborn areas providing tank liner is not damaged or loosened.

(10) Open fuel pump mounting holes and begin draining cleaning solution. Simultaneously, begin directing a spray of warm water on top and sides of tank to rinse off detergent and contaminants. Repeat spraying operation until all contamination and cleaning agents are removed. Continue until a bottle of water collected at fuel pump mounting holes shows no sudsing action when shaken vigorously.

(11) Allow tank to drain.

(12) Inspect interior of tank to ensure all contamination has been removed and for foreign objects such as cleaning materials. Repeat cleaning procedure if contamination still exists.

CAUTION: USE ONLY A LOW PRESSURE AIR SOURCE SUCH AS A FAN OR CENTRIFUGAL BLOWER. DO NOT USE SHOP AIR FOR DRYING TANK INTERIOR.

(13) Use forced-air heat (150°F maximum) to dry tank. Approximate drying time is 4 hours.

(14) Thoroughly inspect lower wing surfaces for corrosion and breakdown of tank liner. See Interior Tank Coating System — Repair, this Section if repairs are necessary.

(15) Install access covers beneath tailpipe and heatshield. Remove all plugs, caps, and liquid tight covers and install components removed above.

(16) Inspect tanks for foreign matter before closing.

(17) Install access covers on tops of wings.

(18) Remove fuel filters, clean fuel lines and filter area with clean fuel, and install a new filter.

(19) Fuel aircraft (See Chapter 12) and perform Boost Pump — Pressure Check, see Section 28-2-1.

(20) Perform Fuel System — Bleed Procedure, see Section 28-0-1 and perform Engine — Operational Test, see Chapter 71.

(21) Remove and inspect fuel filter.

(22) Perform Fuel System — Bleed Procedure, see Section 28-0-1 and perform Engine — Operational Test, see Chapter 71.

4. Tank Interior Coating System — Repair

The following repair method may be used to repair small or easily accessible areas of the tank interior where the required repair is obviously local in nature and no damage to skin or structure is involved.

A. Clean tank interior and decontaminate if required, see Fuel Tank — Cleaning Procedures, this Section.

CAUTION: IN PROCEDURE BELOW, USE MINIMUM AMOUNT OF SOLVENT REQUIRED TO REMOVE COATING. AVOID SPILLS AND USING ENOUGH SOLVENT TO DRIP AND RUN INTO AREAS NOT TO BE TREATED.

B. Using a cloth or brush, wet with Methyl Isobutyl Ketone (MIBK), Federal Specification TT-M-268, remove coating in affected area.

C. Perform a close inspection of area from which coating has been removed. If evidence of corrosion is found, determine depth of penetration. If corrosion extends to unstripped area, remove coating as required to determine extent of corrosion. Superficial corrosion may be removed as follows:

CAUTION: USE OF CARBORUNDUM PAPERS SHOULD BE AVOIDED. EMERY PAPER AND CROCUS CLOTH ARE PROHIBITED, SINCE IMBEDDED IRON OXIDE CAN CAUSE SERIOUS CORROSION PROBLEMS. DO NOT REMOVE ANY MORE METAL THAN ABSOLUTELY NECESSARY. ENSURE ALL CORROSION SALTS AND OTHER DEBRIS ARE REMOVED FROM TANK.

(1) Superficial corrosion may be removed by lightly sanding with aluminum oxide paper, No. 300 grit or finer.

(2) Prepare a solution of 1.5 to 3.0 ounces by weight of Brush Alodine No. 1200 MIL-C-5541 per gallon of water. The pH should be within 1.2 to 1.8.

CAUTION: AREA TO BE TREATED MUST BE THOROUGHLY CLEANED.

(3) Use a damp cloth to remove all loosened corrosion and other debris from area to be treated.

(4) Thoroughly wet surface to be treated with clean water. Wipe any excess from untreated recesses or seams.

NOTE: Do not allow brush alodine solution to dry on surface.

(5) Apply Brush Alodine solution to area, using a cellulose sponge or bristle brush. Allow solution to remain on the surface from 2 to 5 minutes, repeating application if needed to prevent drying.

(6) Remove excess alodine solution with a cellulose sponge or clean cloth dampened in clean water.

NOTE: Do not wipe surface dry.

(7) Allow surface to air dry completely before proceeding.

D. Reactivate coating at least 1/2-inch wide surrounding area to be coated by wiping with a MIBK dampened cloth just before applying coating in next step.

E. Apply a brush coating of EC776, Pro-Seal, 442 or coating conforming to MIL-S-4383. Overlap existing coating treated in D. above. Wait 20 minutes or until coating sets up before proceeding.

F. Apply a second coating as in E. above. Allow at least 1 hour drying time before closing tank.

5. Wing Tank Leakage — Test

A. Description

Either of the two wing tank leakages test procedures given below may be used, or they may be used alternately to confirm a leak (full tank test procedure) and to test a repair (empty tank test procedure). The empty tank test will more accurately pinpoint leak location, and will reveal smaller leaks which may not be readily evident in a full tank test. The full tank test, however, may reveal leaks not evident without the added burden of fuel.

B. Special Tooling Equipment

The following special tools and equipment are required to perform the leak test procedure:

(1) A source of dry shop air or nitrogen gas.

(2) A pressure reducing and regulating valve capable of maintaining 3.5 psi.

(3) A calibrated pressure gage or manometer.

(4) Lines, tees, and fittings to interconnect the above.

(5) A solution of water and mild soap or other sudsing agent.

C. Full Tank - Test

NOTE: The following procedure can be used to test either tank.

(1) Clean outer tank surfaces.

(2) Fill the fuel tank to capacity. (See Chapter 12, Fueling Procedure).

CAUTION: ENSURE ALL VALVES CALLED OUT IN STEP (3) BELOW ARE CLOSED. FAILURE TO COMPLY WILL PREVENT TANK PRESSURIZATION, AND COULD RESULT IN FUEL BEING FORCED INTO OTHER TANK OR ENGINE.

(3) Close high pressure fuel cock, crossfeed valve, and fuel shutoff valve.

(4) Install filler cap on wing tank.

(5) Attach nitrogen gas source to pressure reducing and regulating valve.

CAUTION: AVOID THE DISCHARGE OF LIQUID NITROGEN.

(6) Connect a line from the pressure reducing and regulating valve to a tee. Connect a line from the tee to the tank vent line. The line to the tank vent fitting should be as short as possible.

(7) Attach a pressure gage or manometer to open port of tee:

(8) Open the pressure reducing and regulating valve and apply pressure not to exceed the following:

(a) Aircraft having ASC 125, 3.5 psi maximum.

(b) Aircraft not having ASC 125, 4.5 $\pm$ 1 psi.

(9) While maintaining required pressure, inspect exterior surfaces of tanks for leaks.

(10) Repair any leaks as directed in this chapter.

D. Empty Tank - Test

NOTE: The following procedure can be used to test either tank.

(1) Clean outer tank surfaces.

(2) Drain fuel tank. (See Chapter 12.)

NOTE: All valves called out in Step (3) below must be properly closed to attain proper tank pressurization.

(3) Close high pressure fuel cock, crossfeed valve, fuel shutoff valve and drain valves.

(4) Install filler cap on wing tank.

(5) Attach nitrogen gas source to pressure reducing and regulating valve.

(6) Connect a line from the pressure reducing and regulating valve to a T-fitting. Connect a line from the T-fitting to the tank vent line. The line to the tank vent fitting should be as short as possible.

(7) Attach a pressure gage or manometer to open port of T-fitting.

(8) Open the pressure reducing and regulating valve and apply pressure not to exceed the following:

(a) Aircraft having ASC 125, 3.5 psi maximum.

(b) Aircraft not having ASC 125, 4.5 $\pm$ 1 psi maximum.

(9) While maintaining required pressure, apply soap solution to wings and exterior surfaces to tanks. Observe soap solution for the formation of bubbles.

6. Fuel Cells — Removal / Installation

(Aircraft having ASC 125) (See Figure 201)

NOTE: The procedures given below are general in nature and apply to all fuel cells. However, the difference in location and size of access openings may required modification of the procedures to accommodate removal and installation of each individual fuel cell. See Figure 201 for location of fuel cells, access openings and other components of wing fuel system.

For handling and packaging procedures; See Chapter 82, Water / Methanol Tanks - Maintenance Practices (Aircraft 1 - 106 and 114 not having ASC 125). Additional handling procedures, as well as procedures for storage and repair of fuel cells, are contained in Uniroyal Corporation Report FC-1473-73, third revision, June 1979, available from Uniroyal Corporation, Engineered Systems Department, Mishawaka, Indiana 46544.

A. Removal

(1) Remove all electrical power from aircraft.

(2) Defuel and purge the fuel tanks.

(3) Remove access cover(s) for fuel cell(s) to be removed.

(4) Working through the access opening(s), accomplish the following:

(a) Remove nut and washer from common vent line (6).

(b) Remove nut and washer from common fuel drain line connector (11).

(c) Remove nut and washer from interconnector (3,8 or 8A).

(d) For fuel cells 1 and 4 (13 and 16) remove safety-wire, attaching hardware and bypass vent lines (4).

(e) If removing fuel cell 1 (13) remove bypass vent line from cell.

CAUTION: WHEN REMOVING DAGE FITTINGS FROM REAR BEAM, PREVENT FITTINGS FROM TURNING BY HOLDING FITTINGS FROM INSIDE CELL WHILE REMOVING NUTS. FAILURE TO COMPLY MAY RESULT IN DAMAGE TO FITTING.

(f) If removing fuel cell 2 or 4 (14 or 16), disconnect fuel quantity probe electrical connector. Remove attaching hardware and fuel quantity probe (9).

(g) Disconnect snap hangers securing top of cell to wing upper skin.

(h) Collapse cell uniformly, and roll lengthwise, making as small a package as practical to facilitate removing cell from wing through opening.

(i) Remove cell through wing access opening.

B. Installation

CAUTION: ENSURE INTERIOR OF WING CAVITY IS FREE FROM SHARP EDGES AND ROUGHNESS. CAVITY INTERIOR SHOULD BE CLEANED AND INSPECTED BEFORE CELL INSTALLATION.

(1) Clean fuel cell exterior to remove grit, metal chips or shavings, etc., then coat fuel cell exterior with talcum powder.

(2) Roll fuel cell lengthwise ensuring roll is small enough to fit through wing access opening. Ensure fuel cell is oriented properly, insert through wing access opening.

(3) Guide cell into position and properly seat cell openings and cutouts at respective fitting attachment locations.

(4) Connect snap hangars to wing upper skin.

(5) Secure upper common vent line connector with nut and washer (6).

(6) Secure lower common fuel drain line connector (11) with nut and washer.

(7) Secure interconnector with nut and washer; use special wrenches.

(8) When installing fuel cell 2 or 4 (14 or 16), install fuel quantity probe (9) and secure with screws.

(9) In fuel cells 2 or 4, carefully install dage fittings in rear wing beam, and electrical wires to probe.

(10) When installing fuel cell 1 or 4, (13 or 16) install bypass vent line and fittings (4), secure with screws and safety-wire.

(11) Inspect area for foreign objects, security of all attachments.

(12) Secure access cover.

(13) Fuel aircraft as required and check for leakage.

7. Outboard / Inboard Flapper Valves — Removal / Installation

A. Removal

(1) Remove access cover in wing fillet.

(2) Defuel tank using normal procedures.

(3) If required, remove fuel boost pump.

(4) Remove flapper valve.

B. Installation

(1) Install flapper valve.

(2) Check flapper valve for freedom of movement.

(3) If removed, install fuel boost pump.

(4) Add approximately 50 gallons of fuel to tank.

(5) Perform Fuel System — Bleed Procedure, Section 28-0-1.

(6) Perform Fuel System — Integrity Test, Section 28-0-1.

(7) Inspect area for foreign objects, security of all attachments.

(8) Install access cover.

8. Fuel Tank Flapper Valve — Inspection

A. Check flapper valves for freedom of movement.

B. Check for security of mounting.

9. Fuel Filler Cap O-ring — Removal / Installation

A. Removal

(1) Remove fuel cap from wing filler.

(2) Remove cotter pin and nut from cap.

(3) Remove O-rings from cap.

B. Installation

(1) Install new O-rings as follows:

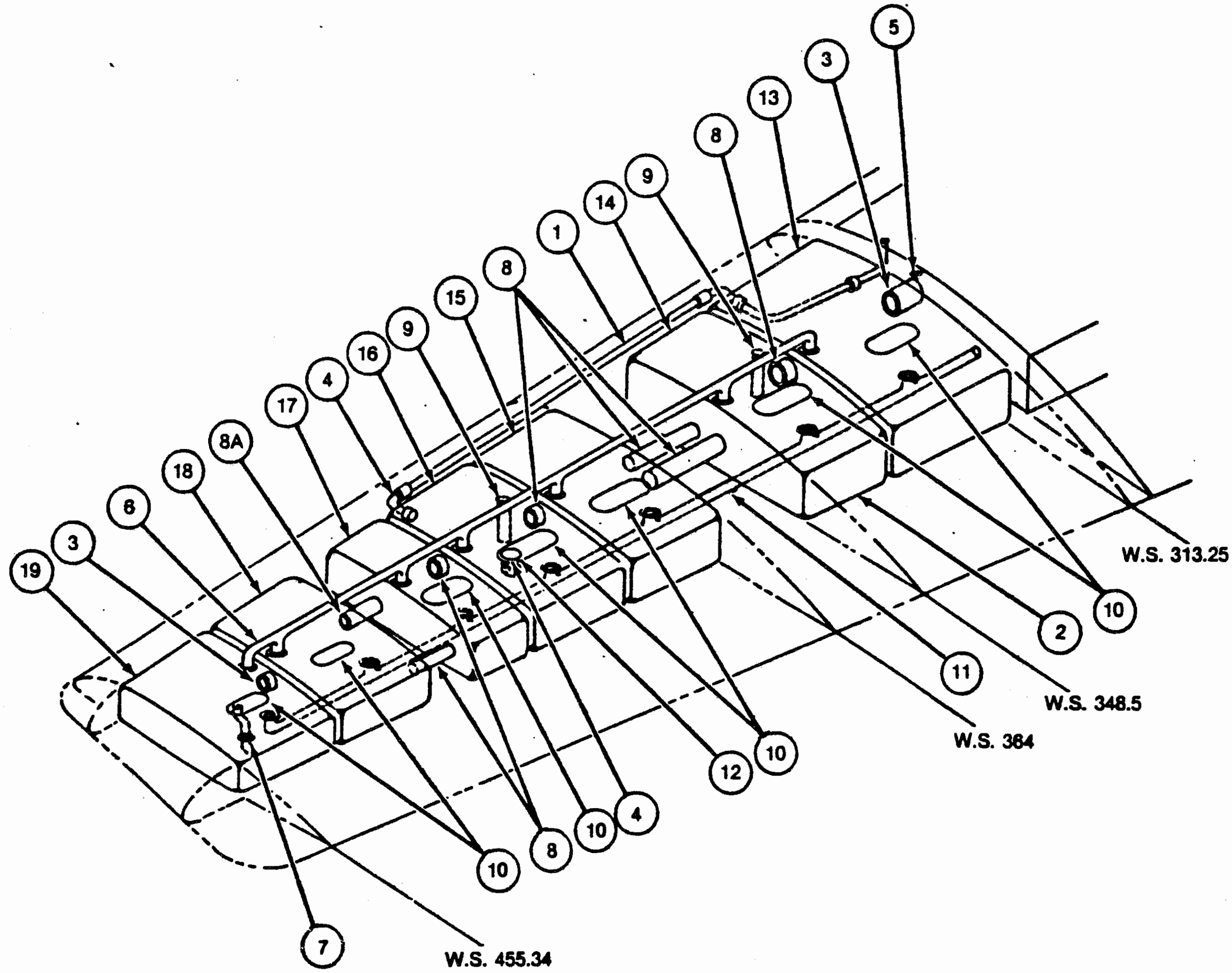
(a) On Aircraft having ASC 125 use MS29513-10 and MS29513-232 O-rings.

(b) On Aircraft not having ASC 125 use MS29513-10 and MS29513-240 O-rings.

(2) Assemble cap; install nut and cotter pin.

(3) Inspect area for foreign objects.

(4) Install fuel cap.



1. WING FRONT BEAM
2. WING REAR BEAM
3. INTER-CONNECT FITTING / FLAPPER VALVE
4. BYPASS VENT LINE
5. FLOAT VALVE
6. COMMON VENT LINE
7. VENT
8. INTER-CONNECT FITTING (W/O FLAPPER VALVE)
8A. INTER-CONNECT FITTING (W/O FLAPPER VALVE-RIGHT WING)
9. FUEL QUANTITY PROBE (TANK UNIT)
10. ACCESS PLATE
11. COMMON FUEL DRAIN LINE
12. FILLER CAP
13. FUEL CELL NO. 1
14. FUEL CELL NO. 2
15. FUEL CELL NO. 3
16. FUEL CELL NO. 4
17. FUEL CELL NO. 5
18. FUEL CELL NO. 6
19. FUEL CELL NO. 7

Location of Fuel Cells
Figure 201.

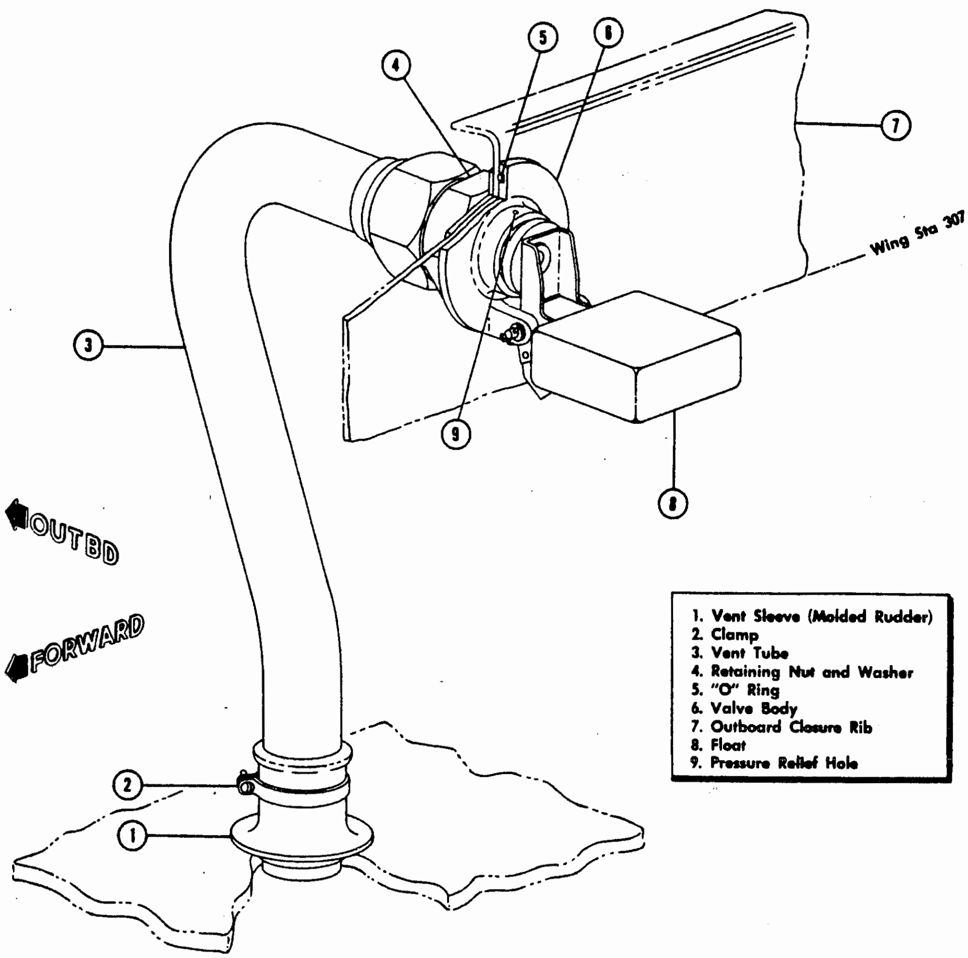
FUEL TANK VENT SYSTEM — STANDARD SYSTEM — DESCRIPTION / OPERATION

1. Description

The integral fuel tanks are vented to the atmosphere by a fuel tank vent system which consists of an internally located vent float valve in each tank and all necessary tubing. (See Figure 1) During flight, each integral fuel tank is slightly pressurized by ram air introduced into the scarfed end of the vent tubing which extends approximately 1/4 inch below the surface of the lower cover. A vent float valve is attached to the inboard face of each outboard closure rib. On Aircraft 1 - 93 and 114 having ASC 156 and Aircraft 94 - 200, 322 and 323, a 5/32 inch diameter hole, incorporated in the valve body, prevents excessive pressure build-up within the wing fuel tanks.

CAUTION: DURING GROUND OPERATIONS INVOLVING BOOST PUMP OPERATION ON ONE SIDE ONLY THE FUEL CROSSFEED SHUTOFF VALVE SHOULD BE CLOSED UNLESS A SYSTEM CROSSFEED CHECK OF SHORT DURATION (5 MINUTES MAXIMUM) IS BEING MADE.

The vent tubing extends from the closure rib to a point within the gap band area. A high level of fuel at the outboard end of the tanks will lift the vent valve floats to the closed position to prevent loss of fuel through the vent. Access to the tank valves and tubing is made through access holes at Wing Station 302 and 311 in the upper cover of each wing.



Fuel Tank Vent Installation
Figure 1.

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FUEL TANK VENT SYSTEM — STANDARD SYSTEM — MAINTENANCE PRACTICES

1. Fuel Tank Vent Valve / Line — Removal / Installation

A. Removal

(1) Drain tank(s) to a maximum of 4,000 lbs. of fuel.

(2) Remove access covers at Wing Station 302 and 311.

(3) Loosen coupling holding vent tube to vent valve.

(4) Remove retaining nut and washer holding vent valve to outboard closure rib.

(5) Loosen clamp holding vent tube in rubber boot on lower cover.

(6) Turn neck of vent tube so that it faces outboard and then lift through access hole.

(7) Remove valve through access hole at Wing Station 302.

B. Installation

(1) Replace O-ring on valve.

(2) Insert valve through access hole at Wing Station 302 and position on closure rib.

(3) Install washer and retaining nut on valve.

(4) Install vent tube and tighten coupling and clamp.

(5) Inspect area for foreign objects, security of all attachments.

(6) Install access covers.

(7) Refuel and check for leaks.

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FUEL TANK VENT SYSTEM (AIRCRAFT HAVING ASC 125) — DESCRIPTION / OPERATION

1. **Description**

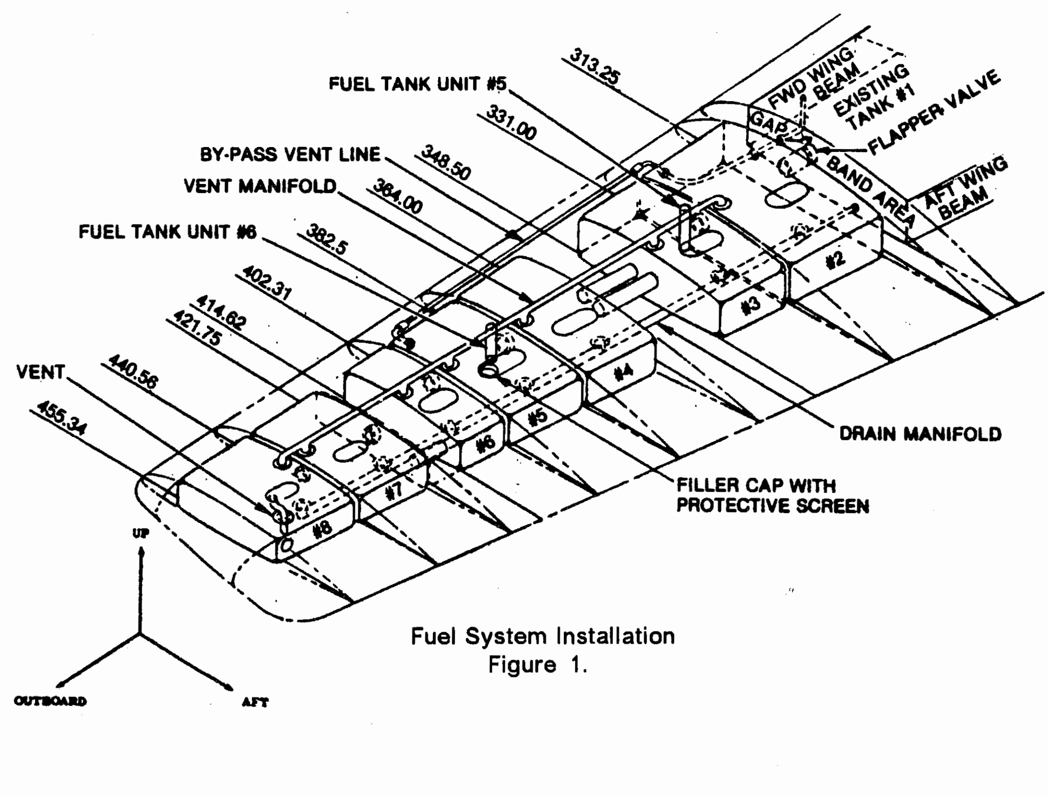
A bypass vent line vents the integral No. 1 tank of each wing to the outboard, bladder type cell No. 5. A common vent line, connecting all the bladder type cells (No. 2 to No. 8) of each wing, vents all wing tanks (including integral No. 1 tank) to the outboard bladder cell No. 8, which is vented overboard under this cell. (See Figure 1) A float valve, in the integral tank bypass vent line, prevents fuel surge outboard to bladder cells, during aircraft maneuvers.

2. **Operation**

The vent system as installed uses the outboard cell as the venting airspace. All vents terminate here either directly or indirectly. As fuel is put into the aircraft, the air normally trapped in the upper outboard end of the integral tank is allowed to escape through the bypass vent into the upper part of tank No. 5. As the fueling proceeds, air from the other tanks (2,3,4,5,6 and 7) is displaced by the fuel through the vent manifold. This manifold terminates in the upper side of cell No. 8. An overboard vent exits through the bottom of cell No. 8 to atmosphere. This line is open at both ends and has a positive scarf to afford air pressure for venting during flight. During the flight then, as fuel is used, all tanks are vented with equal positive pressures.

In addition to the above system, a precautionary or secondary vent has been installed. This vent is know as a differential pressure vent and is located in the upper aft outboard portion of cell No. 8, and exits through the rear beam. This valve is simply a flapper valve which will open when tank pressure is lower than ambient pressure. Since the primary system affords a positive pressure to t he tanks which is somewhat higher than ambient; use of the secondary vent would only come into use if the primary system were blocked and a positive pressure was not available.

Operation of this vent is automatic and requires no action on the part of the pilot.



Fuel System Installation
Figure 1.

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FUEL TANK VENT SYSTEM (AIRCRAFT HAVING ASC 125) — MAINTENANCE PRACTICES

1. **Fuel Vent Valve — Removal / Installation**

A. Removal

NOTE: Failure to comply with (1) below will result in a negative pressure which may be severe enough to pull bladder off hanger.

(1) Drain tanks to a maximum of 4,000 lbs of fuel.

(2) Remove access cover at Wing Station 448 (Fuel Cell No. 8).

(3) Remove 5-shaped vent tube.

(4) Remove vent tube through upper access hole.

(5) Remove check nut and washer from vent fitting.

(6) Remove differential pressure vent.

(7) Remove fuel cell.

(8) Remove screws securing vent fitting to doubler.

(9) Remove vent fitting through upper access hole.

B. Installation

(1) Install fuel vent fitting through lower wing cover; ensure angle faces forward. Secure to doubler.

(2) Install screws securing vent fitting to doubler.

(3) Install fuel cell.

(4) Install differential pressure vent.

(5) Install check nut and washer to vent fitting.

(6) Install vent tube with top of S at access cover.

(7) Inspect area for foreign objects.

(8) Install access cover, using puller, to hold in place when securing screws.

(9) Refuel and check for leaks.

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FUEL CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

The electrically operated fuel control system consists of four low-pressure pumps (two forward and two aft), two main fuel shutoff valves, one crossfeed valve, and all necessary tubing. This system controls the flow of fuel to the engines from the integral tanks. The low-pressure pumps supply fuel at the required rate and pressure to the Flow Control Unit (FCU) of each engine.

The left tank rear pump and the right tank rear pump are identified as the normal pumps: the left tank forward pump and the right tank forward pump are identified as the auxiliary pumps. The normal and auxiliary designations of the pumps are given only as a manner of nomenclature. Neither pump is subordinate to the other and both may be operated alternately during normal flight. When both pumps within a fuel cavity bay are operating, fuel will flow out the discharge ports and manifold into a single feed line. This pump manifold line incorporates a T-fitting for connection to the engine fuel feed line located within the tank. The engine fuel feed line extends outboard and aft from this fitting and exits through the rear beam, at the outboard side of the nacelles.

The main fuel shutoff valves installed in each fuel feed line, provide the means for rapid cutoff of fuel supply in event of engine failure or fire. Closing the shutoff valves also eliminates the necessity of defueling the aircraft when removing components in the fuel feed lines downstream of these valves. The left and right engine main fuel shutoff valves are among the components operated when the FIRE PULL T-handles are pulled.

The fuel control system incorporates a crossfeed system so that both engines can feed from any one tank at the same time. It is not possible, however, to transfer fuel directly from one tank to the other. A FUEL CROSSFEED switch is installed on the right hand side of the lower overhead panel and is located to the right of the FUEL BOOST PUMP switches. (See Figure 1) This is a two-position switch (OPEN and CLOSE). An indicator light is located between the pump switches and the FUEL CROSSFEED switch. The indicator light comes on when the crossfeed gate valve is open. (This is a positive indication.) The crossfeed system incorporates a simple line run which interconnects the left and right integral tank engine feed lines. In each integral tank the fuel from both pumps combines to flow into the engine feed and the aircraft crossfeed line. The crossfeed line attaches to the rear leg of the aft Wiggins T-connection. Using the left tank as a starting point the line is routed aft and exits from the tank through the rear beam. A crossfeed shutoff valve is located aft of the wing beam on the left side of the aircraft. The crossfeed line follows the rear beam, enters the fuselage, and is routed under the floor of the fuselage. Emerging from the fuselage on the opposite side the crossfeed line follows the rear beam of the right wing and enters the inboard bay of the right wing tank. This line terminates by attaching to the rear leg of the right wings rear pump Wiggins T-fitting.

The bends in the crossfeed line, adjacent to the fuselage, are the lowest points of the crossfeed system. A normally-closed, spring-loaded water drain valve is installed in this bend on each side of the fuselage. Access holes are provided in the fairing which make these valves accessible for water draining during preflight.

CAUTION: WHEN WATER DRAINING, IT IS IMPERATIVE THAT THE CROSSFEED VALVE IS CLOSED AND ONE BOOSTER PUMP IN EACH TANK IS OPERATING.

A pressure seal is provided between the fuselage and the crossfeed line to prevent the loss of cabin pressure.

Normally the aircraft flies with the crossfeed system turned off, and one pump operating in each tank. Under these conditions the crossfeed lines on either side of the crossfeed shutoff valve are pressurized. When it is desired to crossfeed from the left tank to the right engine, both pumps in the left tank are turned on. The FUEL CROSSFEED switch is actuated to the OPEN position, and the right boost pumps are turned off. (See Figure 2) The crossfeed gate valve opens and allows fuel to flow from the left tank to the opposite engine feed line. Some of this fuel will enter the right tank through the ejector.

Fuel within a given tank is recirculated in that tank. Operation of the pumps on one side with crossflow open results in fuel being added to the opposite tank at the motive flow rate.

The ejector utilized in the aircraft has the following flow characteristics when operating against a 10 inch head.

Resistor size in motive flow line

Pump pressure	Motive flow	Induced flow	Total flow
18	0.78 gpm	2.37 gpm	3.15 gpm
21	0.84 gpm	2.68 gpm	3.52 gpm

The indicator light comes on when the gate valve is opened.

NOTE: It is recommended that the crossfeed system be turned on, anytime the low level warning lights come on, if the fuel pressure warning light comes on, or if it is desired to balance the fuel load in the wings. (Maximum unbalanced fuel allowable is 1500 pounds.)

The crossfeed system should immediately be turned on following any low fuel pressure warning light indication without low fuel level warning. Assume that the left normal pump is operating and the left low-pressure warning light comes on. If the light goes out after the crossfeed system is turned on, it indicates failure of that pump. Continued crossfeed operation results in an unbalanced fuel load as fuel is supplied to both engines by the right tank. Prolonged crossfeed operation, due to a single boost pump failure, is not necessary because the left tank auxiliary boost pump is still available to supply pressure to the left engine. This second pump should be turned on and the crossfeed system turned off.

WARNING: DO NOT CLOSE CROSSFEED VALVE UNTIL PUMPS ARE OPERATING IN BOTH TANKS. FAILURE TO HEED THIS WARNING COULD RESULT IN ENGINE FLAMEOUT.

Failure of both pumps in the same tank is indicated if the low-pressure warning light comes on again without low fuel level warning. This is not likely to happen since both pumps are run off separate electrical buses. However, there is always a remote possibility that a failure of this type can occur and if it should happen, the crossfeed switch must immediately be turned on again to prevent a possible flameout. Both engines continue to be fed from the tank in which boost pumps are operational. Continued use of the crossfeed system results in an unbalanced fuel load since the fuel in the inoperative tank is not being used. This can be avoided by having the aircraft descend to a lower altitude where suction from the engine-driven pump draws fuel from the malfunctioning tank. The crossfeed system can be turned off.

If the weight of fuel is greater on one side than the other, the crossfeed system can be used to rebalance the load. Both boost pumps in the heavier wing tank should be turned on and the crossfeed valve opened. After the crossfeed valve indicator light comes on, the pump in the lighter tank should be turned off. After the fuel load is balanced, due to both engines operating on heavy tank, one boost pump is turned on in what was previously the light tank, and one boost pump in the heavy tank is turned off. The crossfeed shutoff valve is closed by actuating the selector switch to the CLOSE position.

WARNING: DO NOT CLOSE CROSSFEED VALVE UNTIL PUMPS ARE OPERATING IN BOTH TANKS. FAILURE TO HEED THIS WARNING COULD RESULT IN ENGINE FLAMEOUT.

The crossfeed system should be checked and operated before flight. This check can be done before starting the engines. A boost pump is operated in either of the wings. The low-pressure warning light on that side goes out. The FUEL CROSSFEED switch is turned on. The crossfeed OPEN indicator light should come on and low.pressure warning light on the opposite side should go out. If the opposite engines low-pressure warning light does not go out, it is an indication that the crossfeed system is malfunctioning.

CAUTION: IF THE CROSSFEED OPEN LIGHT DOES NOT COME ON WHEN THE CROSSFEED SWITCH IS PLACED IN THE OPEN POSITION, TEST BULB WITH EITHER WARNING LIGHT TEST BUTTON. IF LIGHT STILL DOES NOT COME ON - DO NOT COUNT ON CROSSFEED AS VALVE IS NOT FULLY OPEN!

THE ABOVE IS ONLY A CHECK ON THE CROSSFEED, NOT A COMPLETE BOOSTER PUMP CHECKOUT.

Each pump in the fuel control system is individually controlled via its own circuit breaker, switch, and relay. (See Figure 3) An ON - OFF switch energizes a relay through which power is supplied to the pump. The relays for the pumps are located in the mid relay box (under floorboard 8), in forward cabin compartment (Fuselage Station 193 to 232). The right and left normal pump control switches are tied together and fed from PUMP CONT. A circuit breaker (essential dc bus). The right and left auxiliary pump control switches are also tied together and fed from PUMP CONT B circuit breaker (main dc bus).

The left normal fuel pump is also used to supply fuel to the APU. To accomplish this, one pole of the four pole APU MAST switch is tied in parallel with the L NORM fuel boost pump switch.

A low-pressure main fuel shutoff valve, located on rear beam, outboard of the nacelle, controls a flow of fuel from the fuel tank. (See Figure 2) The valve is normally energized open. The operation of the valve is controlled by a FIRE PULL T-handle switch located on the eyebrow panel in the cockpit. When a fire occurs in zones or II of an engine, the appropriate FIRE PULL T-handle lights up. The engine fire procedure includes pulling this FIRE PULL T-handle to cause the valve to close, stopping the flow of fuel at that point.

Pulling the left FIRE PULL T-handle accomplishes the following:

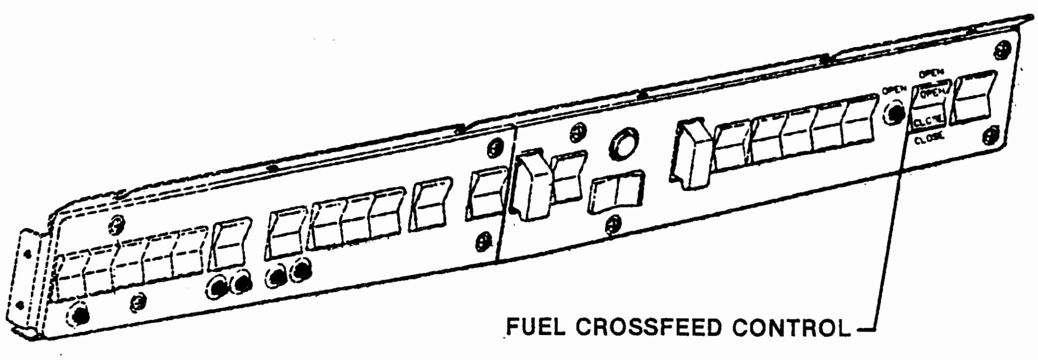
- Closes left tank main fuel shutoff valve.
- Closes left tank water/methanol shutoff valve (if open).
- Closes left engine hydraulic shutoff valve.
- Takes left engines generator off the line.
- Deactivates left engines alternator field.
- Provides access to fire extinguishing switch for that side.

Pulling right FIRE PULL T-handle will result in a similar action for the right engine.

All circuit breakers for fuel control system are located in left circuit breaker panel.

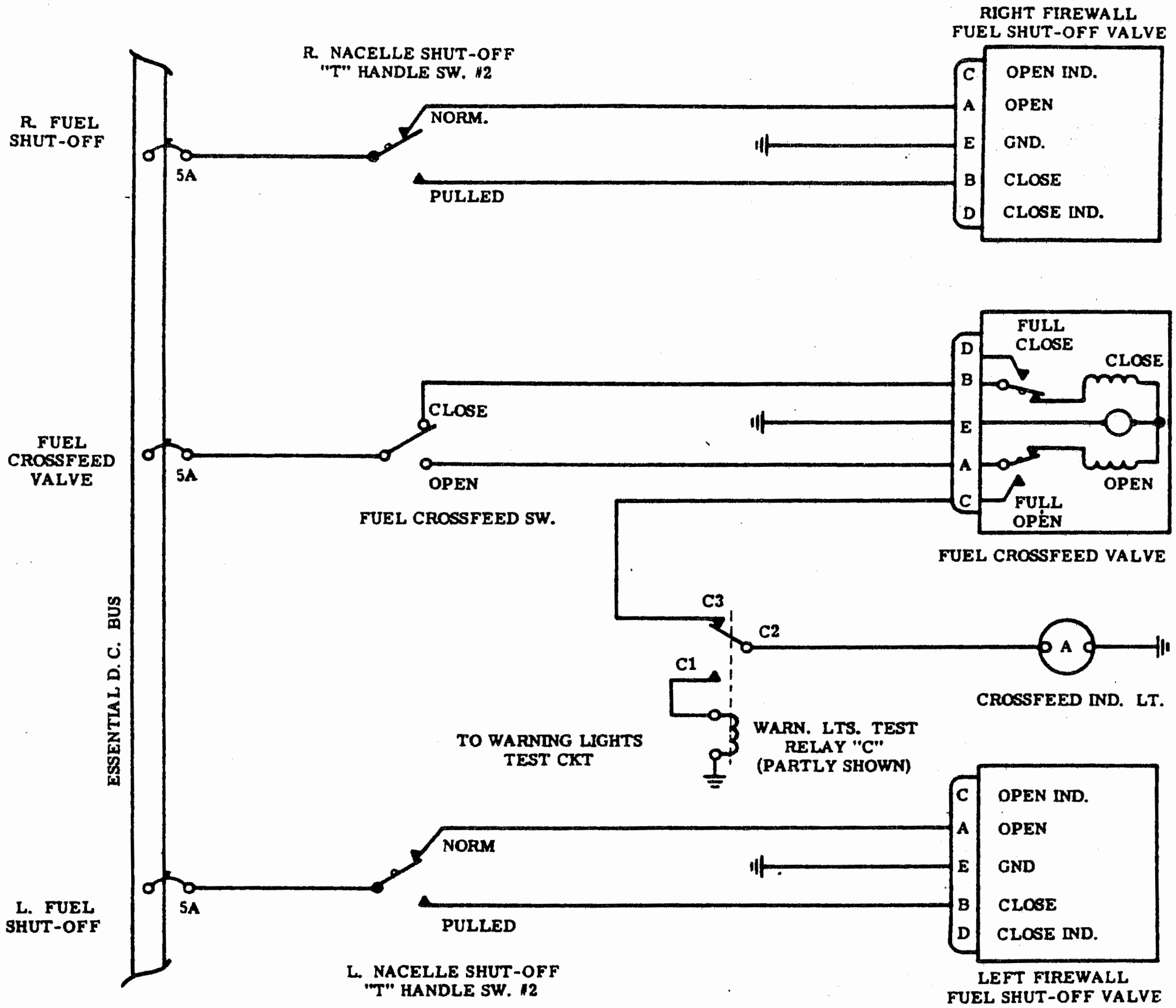
2. **Emergency DC Operation — Effect on System**

- Left and right normal fuel pumps are operative.
- Left and right auxiliary fuel pumps are inoperative.
- Fuel crossfeed is available, including crossfeed indicator light.
- Fuel shutoff is available (through the FIRE PULL T-handle).

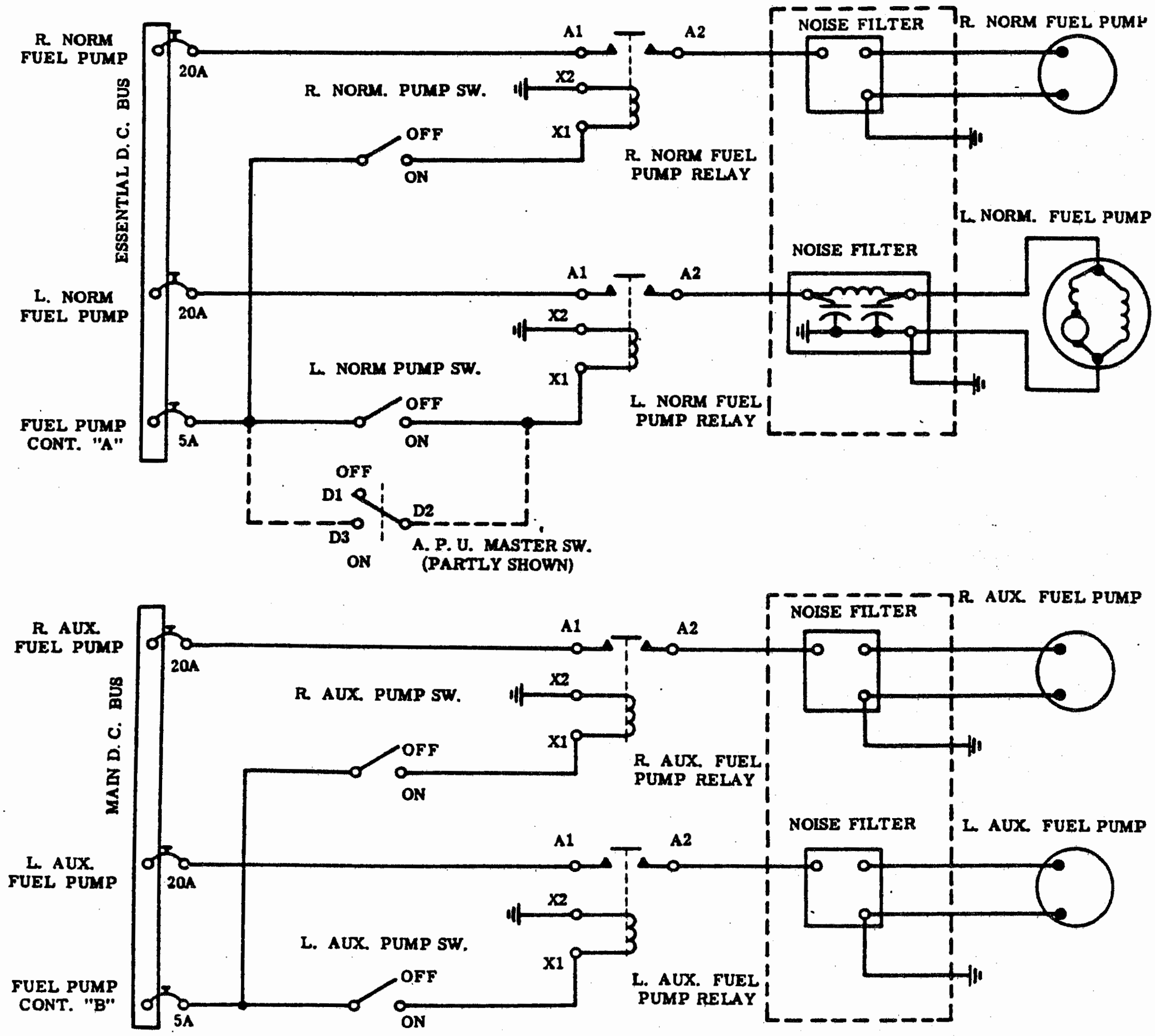


LOWER OVERHEAD PANEL

Fuel Crossfeed Control
Figure 1.



Fuel Control System, Crossfeed and Shutoff Valves — Schematic
Figure 2.



Fuel Control System, Fuel pumps — Schematic
Figure 3.

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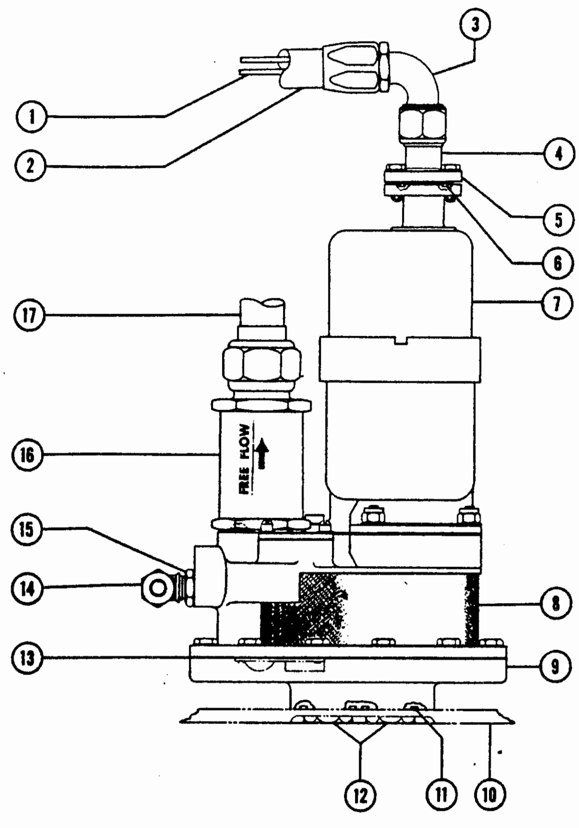
LOW PRESSURE FUEL PUMPS — DESCRIPTION / OPERATION

1. **Description**

Two low pressure fuel pumps, one "normal" and one "auxiliary", are installed in the fuel cavity bay of each integral tank. (See Figure 1) Each pump is driven by an explosion proof, dc motor, and is capable of supplying fuel at the required pressure to both engines when used in conjuction with the engine driven fuel pumps. The left "normal" pump also supplies fuel to the auxiliary power unit (APU). Each pump in the same tank is provided with a different source of power. The forward ("auxiliary") pumps are powered from the 28 volt main dc bus, and the aft ("normal") pumps are powered from the 28 volt essential dc bus. The electrical leads of each pump, contained within flexible conduits, extend aft and exit through the rear beam. The discharge port of each pump is equipped with a check valve to prevent fuel from flowing back into the tank when the pump is inoperative. Access to the pumps in the fuel cavity bay is gained through an access hole in the upper cover of each wing between Wing Station 45 and 55.

On Aircraft having ASC 146, a second source of fuel for the APU has been provided by the addition of fuel lines and check valves, which are connected between the right "normal" pump and the existing fuel supply line from the left "normal" pump. The installation of the second source of fuel to the APU will aid in fuel management whenever APU operations is required. Without incorporation of ASC 146, the fuel system cannot be balanced if the APU is operating since the APU MAST switch keeps the left "normal" pump operating regardless of the position of the L NORM boost pump switch, it is also impossible to operate the APU if the left "normal" pump should fail.



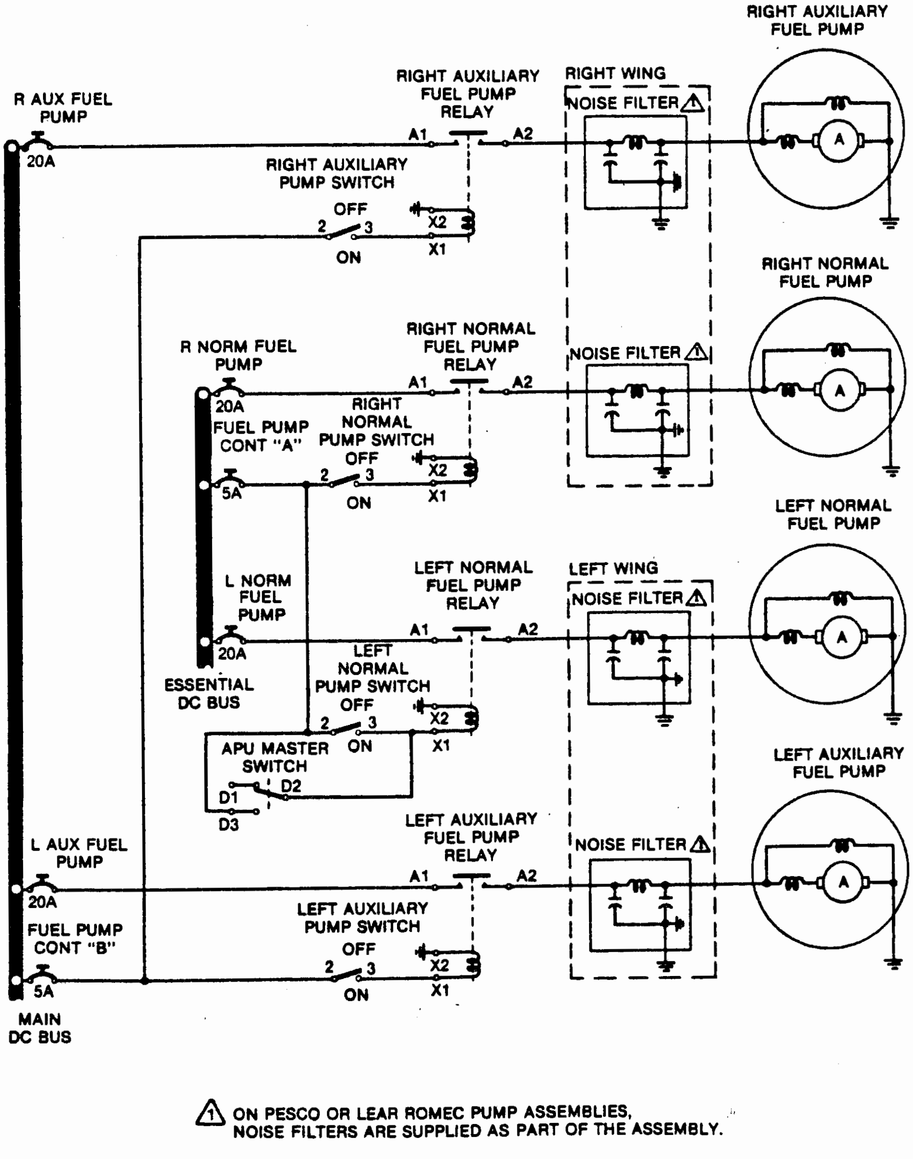
1. Pump Electrical Leads
2. Flexible Conduit
3. 90° Fitting
4. Adapter
5. Swivel Flange
6. Swivel Flange "O" Ring
7. Pump Motor Housing
8. Impeller Screen
9. Mounting Adapter
10. Lower Plank
11. "O" Rings
12. Attachment Bolts
13. Gasket
14. Elbow (To APU Feed Line)*
15. Elbow "O" Ring
16. Flow Check Valve
17. Feed Line

*Used on aft pump LH side only. Plug and "O" Ring used on 3 remaining pumps. If ASC 146 is installed LH and RH aft pumps are the same as shown

Boost Pump
Figure 1.

2. Operation

Each pump is powered through its own 20 ampere circuit breaker, control switch, and relay. (See Figure 2) The circuit breakers for the pumps are located in the left circuit breaker panel. The right "normal" and left "normal" pump control switches are tied together and are fed from the essential dc bus through the 5 ampere PUMP CONT A circuit breaker. The right "auxiliary" and "left" auxiliary pump control switches are similarly tied and fed from the main dc bus through the 5 ampere PUMP CONT B circuit breaker. The pump control switches are located on the lower overhead panel in the cockpit. Setting the pump control switch to ON energizes a relay which allows the dc power to be directed to the pump motor.



Low Pressure Fuel Pump — Schematic
Figure 2.

LOW PRESSURE FUEL PUMPS — MAINTENANCE PRACTICES

1. Pressure Check

A. Prestart Check. (FUEL CROSSFEED switch in CLOSED Position.)

NOTE: Pressure is available at the engine pumps inlet port if either the normal or auxiliary pump is in operation. As a result, with both pump switches turned ON one of the two pumps might fail and this failure would not be indicated in the cockpit. A prestart pressure check should be accomplished to ensure the operation of both pumps prior to flight.

(1) Place L and R NORM boost pump switches to ON. Low-pressure warning lights should go out.

(2) Place switches to OFF. Low-pressure warning lights should come on.

(3) Place L and R AUX boost pump switches to ON. Lights should go off.

NOTE: If at any time the warning lights do not go out when any boost pump is operating, it is an indication of pump or system failure and remedial measures should be taken. (Malfunction of the boost pumps, pressure switch, warning lights, etc., might cause this type of indication.)

Recorded pressure for each pump should be a minimum of 20 psi if a direct reading pressure indicator is used for shop checks.

B. Ground Run Check.

During ground run, turn off aircraft boost pumps while engines are operating to ensure that engines function satisfactorily without aid of low-pressure pumps. (Low.pressure warning light should come on or flicker during this operation as suction of engine driven fuel pump does not maintain 9 $\pm$ 1/4 psi required by the pressure switch to prevent a low-pressure indication in the cockpit.)

2. Fuel Differential Pressure Switch — Removal / Installation

A. Removal

(1) Open engine cowl.

(2) With power on aircraft, pull FIRE PULL T-handle to close main fuel shutoff valve.

(3) Remove electrical power from aircraft.

(4) Disconnect electrical connector from pressure switch.

(5) Disconnect sensing line at HP port.

(6) Disconnect lines from T-fitting at low pressure port.

(7) Cap lines.

(8) Remove pressure switch. Remove unions and O-rings from switch and cap switch ports.

B. Installation

(1) Lubricate O-rings with paraffin and install on unions.

(2) Install unions in switch.

(3) Install pressure switch.

(4) Remove caps and connect lines to HP port and T-fitting.

(5) Connect electrical connector to pressure switch.

(6) Perform Fuel Differential Pressure Switch - Functional Test, this Section.

(7) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

(8) Close engine cowl.

3. Fuel Differential Pressure Switch — Functional Test

A. Open engine cowl.

B. Energize main and essential dc buses.

C. Pull FIRE PULL T-handle to close main fuel shutoff valve; ensure fuel heater switch is in AUTO.

D. On Aircraft not having ASC 108A, pull LANDING GEAR WARNING HORN and WARNING LIGHTS circuit breakers. On Aircraft having ASC 108A, pull LG NUTCRACKER circuit breaker.

E. Disconnect pressure sensing line at high side of pressure switch.

F. Connect air pressure source and pressure gage to high side of switch.

G. Slowly increase air pressure until hot air solenoid valve, on underside of engine clicks open.

NOTE: Use a mercury manometer to observe switch actuation pressure.

H. Observe pressure when valve actuates. Pressure readings should be:

(1) Switch P/N 32028-1

(a) Increasing pressure - 3.87 in Hg (1.9 psig maximum).

(b) Decreasing pressure - 2.04 in Hg (1.5 psig minimum).

(2) Switch P/N 32028-3

(a) Increasing pressure - 5.09 in Hg (2.5 psig maximum).

(b) Decreasing pressure - 4.07 in Hg (2.0 psig minimum).

I. Note that amber warning light below switch in cockpit comes on simultaneously with actuation of solenoid.

NOTE: Aircraft with 2 minute ± 10 second or 5:45 ± 30 second timers, valve will remain energized and amber light will continue to glow after pressure is released due to timer operation. Untimed systems (not having ASC 117A or 117B) valve will deenergize and light will go out when pressure is lowered.

J. Reset circuit breakers.

K. Perform Fuel System — Bleed Procedure, see Section 28-0-1.

L. Remove electrical power from aircraft.

M. Inspect area for presents of foreign objects, security of all attachments and close engine cowl.

4. Fuel Low Pressure Switch — Removal / Installation

A. Removal

(1) Open engine cowl.

(2) With electrical power on, pull FIRE PULL T-handle to close main fuel shutoff valve.

(3) Remove electrical power from aircraft.

(4) Disconnect pressure sensing line at elbow.

(5) Disconnect electrical plug from pressure switch.

(6) Remove pressure switch. Remove union and O-ring from switch and install plugs.

B. Installation

(1) Lubricate O-ring with paraffin and install on union.

(2) Install union in switch.

(3) Install pressure switch.

(4) Connect electrical plug to pressure switch.

(5) Connect pressure sensing line at elbow.

(6) Perform Fuel Low Pressure Switch - Functional Test, this Section.

(7) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

(8) Inspect area for presents of foreign objects, security of all attachments and close engine cowl.

5. Fuel Low Pressure Switch — Functional Test

A. Energize main and essential dc buses.

B. Pull FIRE PULL T-handle to close main fuel shutoff valve.

C. Open engine cowl.

D. Disconnect 1/4 inch sensing line at pressure switch.

E. Connect an air pressure source and direct reading gage to pressure switch.

F. Slowly increase air pressure until approximately 13 psi is reached.

NOTE: When increasing pressure, warning light should go out at or before 11 psi.

G. Slowly decrease pressure, warning light must come on at 9 $\pm$ 1/4 psi.

H. Return system to normal configuration.

I. Perform Fuel System — Bleed Procedure, see Section 28-0-1.

J. Remove electrical power from aircraft.

K. Inspect area for presents of foreign objects, security of all attachments and close engine cowl.

6. Fuel Boost Pump — Removal / Installation

A. Removal

(1) Defuel fuel tank using normal defueling procedures.

(2) Remove access cover in wing fillet.

(3) On right side only, remove landing light cover access panel and air conditioning duct in rear beam.

(4) Remove electrical power from aircraft.

CAUTION: ENSURE THAT SPANWISE ENGINE FUEL LINE IS NOT DAMAGED.

NOTE: When removing forward (auxiliary) fuel boost pump, first remove rear (normal) fuel boost pump and all connecting tubing before attempting to remove forward pump.

(5) Disconnect electrical leads at radio noise filters on rear beam.

(6) Disconnect rear pump flexible conduit at base of swivel fitting by removing bolts and washers securing pump adapter and swivel flange to pump. Discard O-ring.

CAUTION: WHEN REMOVING FORWARD PUMP, DISCONNECT FLEXIBLE CONDUIT ONLY AT BASE OF SWIVEL FITTING.

(7) Tie guide string to end of electrical leads. Pull leads through and leave string in conduit as leads are withdrawn to aid in installation.

(8) Disconnect fuel crossfeed line.

(9) Disconnect fuel line at pump check valve.

CAUTION: DO NOT REMOVE T-FITTING FROM ENGINE FUEL FEED LINE. LEAVING T-FITTING ON LINE WILL PREVENT DAMAGE TO LINE DURING REMOVAL.

(10) Disconnect and remove fuel line at aft side of Wiggins T-fitting between forward and rear pumps. When removing forward pump, disconnect fuel lines on both sides of T-fitting.

(11) If left rear pump is involved, disconnect APU fuel feed line from pump; if right rear pump is involved and having ASC 146, perform same procedures.

(12) Disconnect bonding jumpers.

(13) Remove hollow bolts that hold rear pump and adapter to lower wing plank. Discard O-rings.

(14) Remove bolts in pump mounting adapters that attach rear pump to tank brackets.

(15) Pull rear pump slightly forward and remove.

(16) Remove hollow bolts holding forward pump and adapter to lower wing plank. Discard O-rings.

(17) Remove bolts in pump mounting adapter that attach forward pump to tank brackets.

(18) Pull forward pump gently around fuel quantity probe and engine fuel line; then remove.

(19) Inspect flapper valves for freedom of movement and security of mounting.

B. Installation

NOTE: Before installing pump(s), check condition of adapter gasket and replace as necessary.

(1) Install new O-ring MS29513-019 on mounting adapter and conduit mounting flange.

NOTE: Ensure that check valve is installed correctly with free flow direction arrow pointing away from pump.

If only rear (normal) pump is being installed, continue on with Step (8).

(2) Install forward pump and draw electrical leads through conduit, and at same time position pump

(3) over attachment holes and install hollow bolts with new MS29513-214 O-rings. Torque hollow bolts to 300-500 inch pounds.

(4) Install bolts in pump mounting adapter that attach pump to tank brackets.

(5) Connect bonding jumpers on forward pump.

(6) Install fuel line between forward pump and Wiggins T-fitting on engine fuel feed line and safety wire.

(7) Connect flexible conduit at forward pump adapter and swivel flange.

(8) Connect electrical leads at radio noise filter on rear beam (outboard leads).

(9) Install rear pump and draw electrical leads through conduit and at same time position pump over attachment holes. Install hollow bolts with new MS29513-214 O-rings. Torque hollow bolts to 300-500 inch pounds.

(10) Install bolts in pump mounting adapter to attach pump to tank brackets.

NOTE: If fitting was removed from pump, replace MS29512-6 O-ring.

(11) Connect APU feed line to left rear pump or right pump if having ASC 146.

(12) Connect bonding jumpers.

(13) Install fuel line between rear pump and Wiggins T-fitting on engine fuel feed line and safety wire.

(14) Connect fuel crossfeed line and safety wire.

(15) Connect fuel line at pump check valve.

(16) Connect flexible conduit at rear pump adapter and swivel flange.

(17) Connect electrical leads at radio noise filter on rear beam (inboard leads).

(18) Inspect for foreign objects then install access covers.

(19) Fill tanks with approximately 50 gallons of fuel.

(20) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

(21) Perform Fuel System — Integrity Test, see Section 28-0-1.

(22) Place appropriate boost pump switch on. Respective low fuel pressure light should go out.

7. **Fuel Booster Pump — Pressure Check**

A. Connect an adapter that will attach to the 1 1/4 inch fuel drain valve in nacelle and to a 0-30 psi direct reading pressure gage.

B. Energize main and essential dc buses.

C. Start one pump, record pressure and shut down pump.

D. Start second pump, record pressure and shut down pump. Recorded pressure for each pump should be a minimum of 20 psi.

8. **Normal / Auxiliary Fuel Boost Pump Check Valves — Removal / Installation**

A. Removal

NOTE: If check valve is being removed at same time as fuel boost pump, the two components should be removed intact and disassembled on the bench; otherwise check valve may be removed from pump while in tank.

(1) Defuel tank using normal defueling procedures.

(2) Open access cover in wing fillet.

(3) If required, remove fuel boost pump(s).

(4) Remove check valve.

B. Installation

(1) Install new O-ring on check valve.

(2) Inspect area for foreign objects.

(3) Install check valve and ensure free flow direction arrow is pointing away from pump.

(4) Install fuel boost pump(s) if removed.

(5) Perform Fuel System — Integrity Test, see Section 28-0-1.

(6) Check pump check valve as follows:

(a) To check left auxiliary pump check valve, operate left normal pump. Left low fuel pressure light should go out.

(b) To check left normal pump check valve, operate left auxiliary pump. Left low fuel pressure light should go out.

(c) Repeat Step (1) or (2) above, as required, for right side.

(7) Inspect area for foreign objects, security of all attachments and secure access covers.

9. **Boost Pump — Check (Static)**

NOTE: This check can be combined with the static crossfeed check.

A. Place BATT switch to NORM.

B. Ensure FUEL CROSSFEED switch is CLOSED.

C. Place L NORM and R NORM boost pump switches to ON, Land R FUEL PRESS warning lights should go out.

D. Place L NORM and R NORM boost pump switches to OFF.

E. L and R FUEL PRESS warning lights should come on.

F. Place L and R AUX boost pump switch to ON, L and R FUEL PRESS warning lights should go out.

G. Place Land R AUX boost pump switches OFF, Land R FUEL PRESS warning lights should come on.

H. Place BATT switch to OFF.

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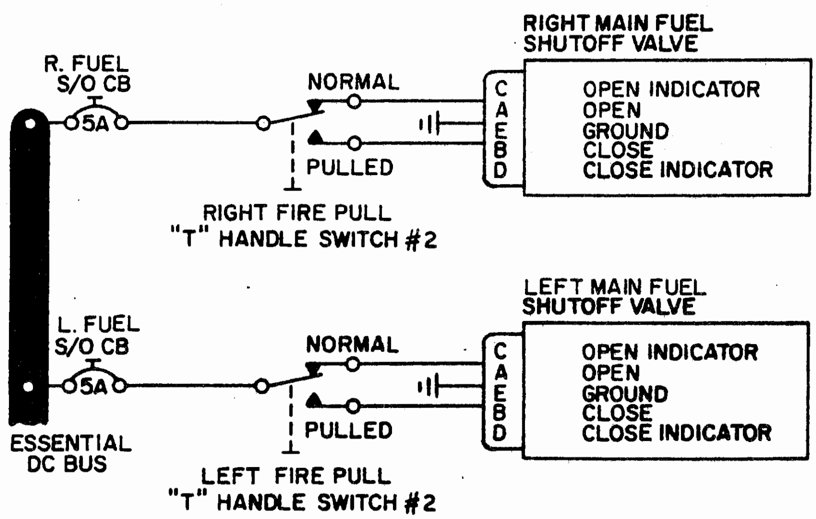
FUEL SHUTOFF VALVES — DESCRIPTION / OPERATION

1. Description

An electric motor driven, sliding gate shutoff valve is installed in each engine fuel feed line. The valves are attached to the rear beam outside of the integral tanks at Wing Station 161.75. A flag lever and OPEN and CLOSED markings are provided on each valve to give a visual indication of the sliding gates position. Under normal conditions the valves are in the open position and cannot be closed by applying manual force to the flag levers. The left and right FIRE PULL T-handles are the only means of operating the valves. Each valve is powered from the 28 volt essential dc bus through a 5 ampere circuit breaker located on the left circuit breaker panel. (See Figure 1) Access to the fuel shutoff valves is through the wing flap bay area.

The valve standard is as follows:

- Line Thermal Relief 30-55 psi both directions. Relief Valve Assembly No. 102913 AW.
- Body Relief 30-55 psi. Relief Valve Assembly No. 103852 BF.
- Actuator Assembly No. 102842 ME.



Fuel Shutoff Valve — Schematic
Figure 1.

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FUEL SHUTOFF VALVES — MAINTENANCE PRACTICES

1. **Fuel Shutoff Valve — Removal / Installation**

   A. Removal

      (1) Defuel tank using normal defueling procedures.

      (2) Remove electrical power from aircraft.

      (3) Disconnect valve electrical connector.

      (4) Disconnect fuel line at elbow.

      (5) Remove valve.

   B. Installation

      (1) Use new adapter O-rings.

      (2) Position valve and adapter on rear beam and install hardware.

      (3) Connect fuel line at elbow.

      (4) Connect valve electrical connector.

      (5) Apply electrical power to aircraft.

      CAUTION: PULL FIRE EXT SHOT NO. 1 AND NO. 2 CIRCUIT BREAKERS BEFORE PULLING FIRE PULL T-HANDLE. CIRCUIT BREAKERS SHOULD ALSO BE LABELED TO PREVENT INADVERTENT DEPRESSION DURING TEST.

      (6) Pull FIRE PULL T-handle, check that shutoff valve flag levers are in closed position.

      (7) Reset FIRE PULL T-handle, check that shutoff valve flag levers are in open position.

      (8) Fuel the tank and check for leaks.

      (9) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

2. **Fuel Shutoff Valve — Operational Test**

   A. Energize essential dc bus.

   B. Gain access to fuel shutoff valves in left and right wing flap bay areas.

   CAUTION: PULL FIRE EXT SHOT NO. 1 AND NO. 2 CIRCUIT BREAKERS BEFORE PULLING FIRE PULL T-HANDLE. CIRCUIT BREAKERS SHOULD ALSO BE LABELED TO PREVENT INADVERTENT DEPRESSION DURING TEST.

   C. Pull FIRE PULL T-Handle; check that shutoff valve flag levers are in closed position.

   D. Reset FIRE PULL T-handle; check that shutoff valve flag levers are in open position.

   E. De-energize essential dc bus.

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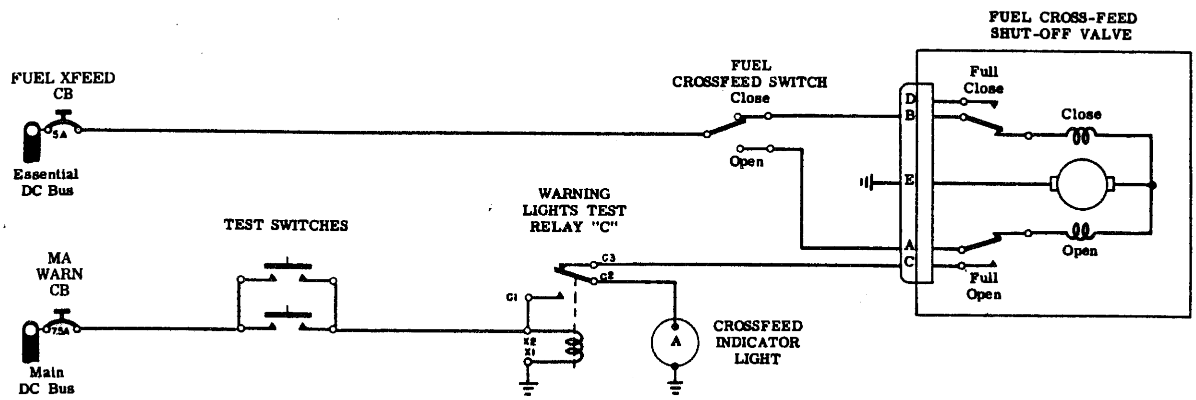
FUEL CROSSFEED VALVE — DESCRIPTION / OPERATION

1. Description

The crossfeed shutoff valve is attached to the rear beam outside of the left integral fuel tank at Wing Station 41. This valve is interchangeable with the left and right fuel tank shutoff valves. This valve is normally closed to allow both lines on each side of the valve to be pressurized. A crossfeed switch marked OPEN and CLOSE is located to the right of the four low pressure pump switches on the lower overhead panel. A crossfeed indicator light installed to the left of the crossfeed switch will come on when the crossfeed valve is opened. The valve is powered from the essential dc bus through the 5 ampere FUEL XFEED circuit breaker on the left circuit breaker panel. (See Figure 1) Access to the shutoff valve is through the left wing flap bay area.

The valve standard is as follows:

- Line Thermal Relief 30-55 psi both directions. Relief Valve Assembly No. 102913 AW.
- Body Relief 30-55 psi. Relief Valve Assembly No. 103852 BF.
- Actuator assembly No. 102842 ME.



Fuel Crossfeed Valve — Schematic
Figure 1.

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FUEL CROSSFEED VALVE — MAINTENANCE PRACTICES

1. **Fuel Crossfeed Valve — Removal / Installation**

   A. Removal

   CAUTION: DO NOT RUN PUMPS DRY.

   (1) Drain left and right tanks using normal defueling procedures.

   (2) Remove electrical power from aircraft.

   (3) Gain access to valve on inboard left rear wing beam.

   (4) Disconnect valve electrical connector.

   NOTE: Residual fuel in crossfeed line will drain out. Take proper precautions.

   (5) Remove valve.

   B. Installation

   (1) Use new adapter O-rings.

   (2) Position valve and adapters on closure rib and install hardware.

   (3) Connect valve electrical connector.

   (4) Apply power to aircraft essential bus; battery connected, BATT switch in NORM or EMER.

   (5) Check valve for proper operation as follows:

   - With FUEL CROSSFEED switch in CLOSE position, visually observe indicator tab on valve body is in closed position; amber crossfeed indicator light adjacent to switch should be OFF.
   - Place FUEL CROSSFEED switch to OPEN position; valve should circle to full open position on valve body indicator tab and crossfeed amber light in cockpit should come on.
   - Cycle valve several times to ensure proper operation and indication of amber light in cockpit.

   (6) Leave valve in CLOSE position.

   (7) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

2. **Fuel Crossfeed Valve — Operational Test**

   A. Energize essential dc bus.

   B. Gain access to fuel crossfeed valve located left wing rear beam, Fuselage Station 41.

   C. Place crossfeed switch to OPEN.

   D. CROSSFEED light should come on, check that flag lever is in open position.

   E. Place crossfeed switch to CLOSE.

   F. CROSSFEED light should go out, check that flag lever is in closed position.

   G. Remove electrical power.

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FUEL EJECTOR — DESCRIPTION / OPERATION

1. **Description**

The fuel ejector or jet pump is a low pressure, high volume pump with no moving parts. The ejector utilizes the energy of a small mass flow, high velocity stream (motive flow) to induce a large mass flow, low velocity stream (induced flow).

The ejector consists of an inlet tube, a mixing tube, an outlet and a motive flow inlet and nozzle. Only the fluid moves. Wear (by erosion) from the moving fluid is completely eliminated by hard surfacing the fluid passages.

The ejector makes use of the power now wasted in excess capacity of boost pumps by tapping into the fuel crossfeed line. The excess boost pump flow, instead of being relieved back to tank with all pressure energy lost, is bypassed at boost pump pressure through a restrictor to the ejector. The ejector utilizes this fuel flow as a motive energy to power the ejector for moving fuel from remote sections to make it more readily available for use.

The installation of the fuel ejector system in the Gulfstream increases the available supply of fuel to the fuel boost pumps at low fuel state conditions with the aircraft in an aborted landing configuration. The ejector is located in the aft section of the fuel pump cavity.

2. **Operation**

The induced flow comes about by the passage of the high velocity motive flow through the venturi throat which creates a low pressure area into which external pressure pushes the fluid to be pumped. The high velocity stream impinges upon this fluid in the throat and transfers energy to it. With proper proportioning of the motive stream, throat and mixing tube, the energy of the high velocity stream can be transferred to the induced stream to produce a large mass, low velocity flow from the rear inboard portion of wing to the base of the aft pump.

The ejector utilized in the Gulfstream has the following flow characteristics when operating against a 10 inch head.

Restrictor size in motive flow line 0.086 Dia

Pump pressure	Motive flow	Induced flow	Total flow
18	0.78 gpm	2.37 gpm	3.15 gpm
21	0.84 gpm	2.68 gpm	3.25 gpm

CAUTION: DURING GROUND OPERATIONS INVOLVING BOOST PUMP OPERATION OF ONE SIDE ONLY, THE FUEL CROSSFEED SHUTOFF VALVE SHOULD BE CLOSED UNLESS A SYSTEM CROSSFEED CHECK OF SHORT DURATION (5 MINUTES MAXIMUM) IS BEING MADE. THIS WILL PRECLUDE POSSIBLE STRUCTURAL DAMAGE OF THE WING FUEL TANKS.

Fuel within a given tank is merely recirculated in that tank. Operation of the pumps on one side with crossfeed OPEN results in fuel being added to the opposite tank at the motive flow rate.

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FUEL EJECTOR — MAINTENANCE PRACTICES

1. Fuel Tank Ejector — Removal / Installation

A. Removal

(1) Defuel tank using normal defueling procedure see Section 28-0-1.

(2) Gain access to ejector through most inboard upper chamber wing hand hole cover.

(3) Disconnect motive flow line from fitting.

(4) Remove ejector.

B. Installation

(1) Install ejector.

(2) Connect motive flow line to fitting.

(3) Inspect area for foreign objects, security of all attachments.

(4) Close hand hole cover and return system to normal configuration.

(5) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

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FUEL FLOWMETER SYSTEM — DESCRIPTION / OPERATION

1. Description

A fuel flowmeter system is installed in each fuel feed line to provide a visual indication of the rate of fuel consumption in pounds per hour. Each system consists of an indicator, a transmitter, and an instrument transformer. Power for each system is supplied from 115/26V ac instrument transformers which receive power from the 115V ac instrument bus through 3/4 ampere circuit breakers located on the right circuit breaker panel. The fuel flow indicators are hermetically sealed, single pointer instruments. Both indicators are normally mounted in the cockpit right.center instrument panel and have dials graduated to indicate a fuel consumption of 100 to 1500 pounds per hour. In some aircraft the indicators are located at the bottom of the cockpit left-center instrument panel. The transmitters receive fuel from the fuel system, register the rate of flow, and discharge fuel to the respective Fuel Control Unit (FCU). Each transmitter is equipped with a bypass valve to permit the flow of fuel to the engine in the event of internal failure. The transformers are 115/26 volt auto-transformers of the open type treated with environment resistant material. The transformers are located under floorboard No. 8 and are used to step down 115 volt, 400 cycle, one phase power to 26 volt, 400 cycle, one phase power. The fuel flow transmitters are accessible through the main gear doors between Fuselage Station 293 and 331.

2. Operation

Fuel supplied from the fuel system enters the metering chamber of the transmitter which is located in the main fuel line of the engine, in the main wheel well downstream from the main fuel shutoff valve. The fuel causes a swinging vane to move against a calibrated spring. The position of the vane is then converted to electrical signals by means of an attached ring magnet. These signals are then received by the indicator and the variations thus produced in the three coil windings positions the pointer of the indicator. (See Figure 1)

The indicator, although calibrated in pounds per hour, is subject to error since the transmitter is a gravimetric measuring device, not a true mass flow device. The spring is calibrated for fuel at room temperature and therefore fuel temperature (especially low fuel temperature) has a very pronounced effect on the calibration of the fuel flow transmitter.

The curve in Figure 2 shows the effect of varying fuel temperature on the transmitter and also includes the fuel temperature effect on specific gravity. The error shown is characteristic of this type of transmitter under varying fuel temperature conditions, and is in addition to any calibration error at room temperature of the transmitter or indicator.

As the fuel temperature is lowered, the system will read MORE than is actually flowing.

This problem will manifest itself only when fuel in the aircraft integral fuel tanks has been cooled sufficiently by either exposure to low ambient temperatures while parked, or by low ambient temperatures experienced during extended cruising flight at high altitudes.

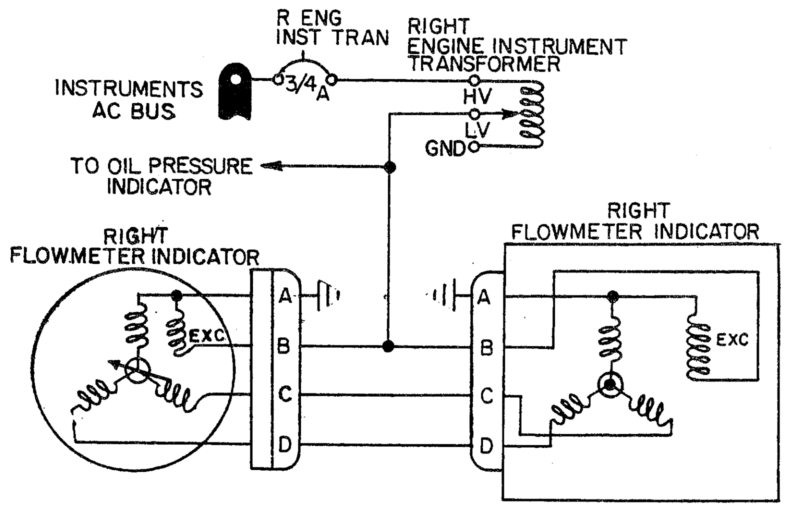
To apply corrections to fuel flow the graph given in Figure 2 must be used. To illustrate, assume that there are no transmitter or indicator errors at room temperature. At a fuel temperature of - 40°C the fuel flow transmitter will generate a signal 24 lbs/hour more than the flow passing through the transmitter. The 24 lbs/hour then, must be subtracted from the indicated cockpit fuel flow to obtain the actual flow.

The operation of the fuel heater will not affect the fuel flow transmitter, since the fuel heater is located downstream of the transmitter. Therefore, the correction applied to fuel flow prior to the selection of fuel heat will apply during the applications of fuel heat.

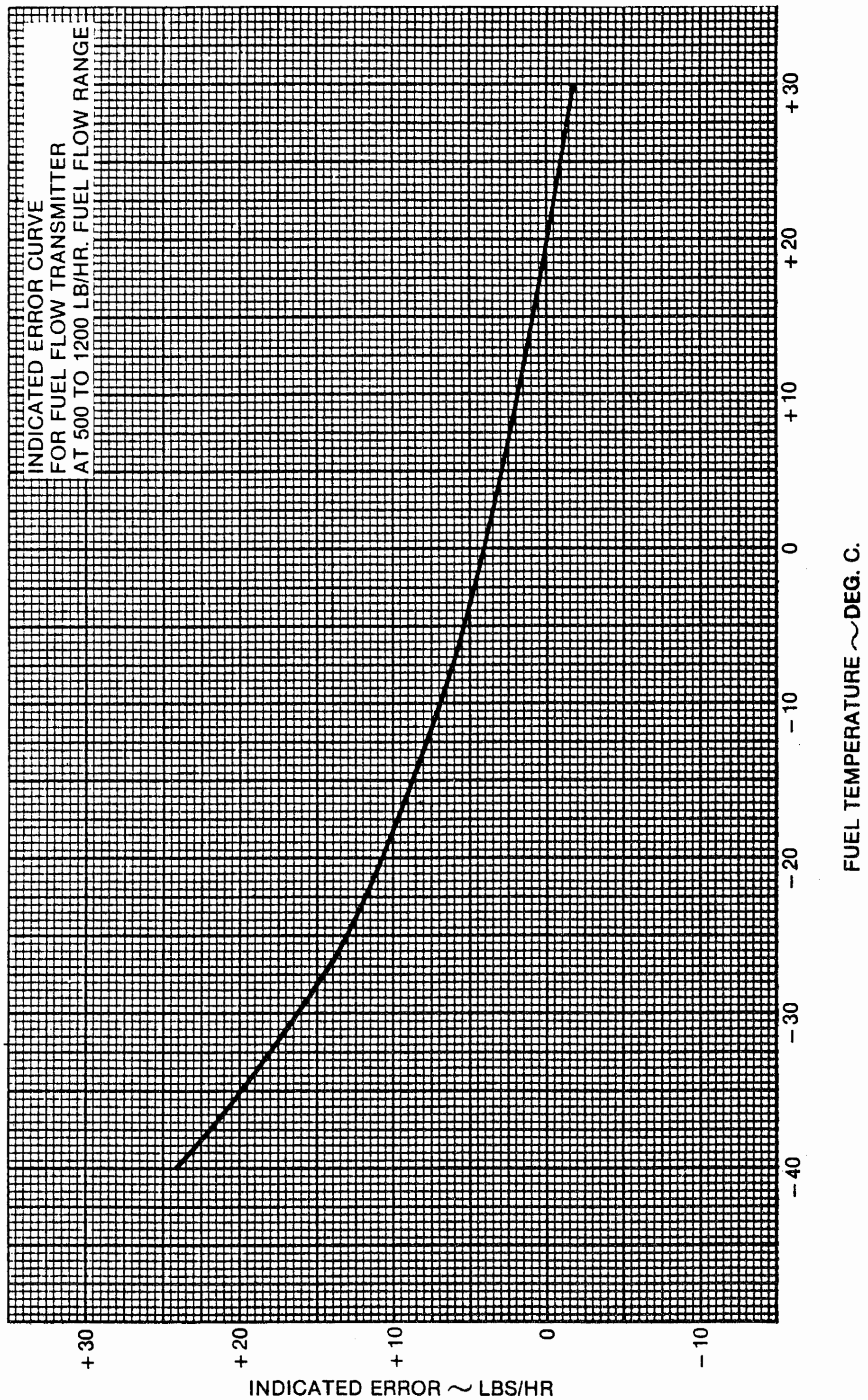
Note that at maximum recommended cruise power (14,200 RPM), the engine is still to be operated to the controlling parameter which occurs first; either the limiting TGT, or the limiting fuel flow as specified for that engine (with cold fuel, the correction must be added to the chart value of limiting fuel flow).

The fuel flowmeter indicating system is operative in Emergency Operation

CAUTION: WHEN DRAINING FUEL ON AIRCRAFT 164 - 200 AND THOSE HAVING ASC 178 NOT INCLUDING 322 AND 323, DO NOT OPERATE THE INVERTERS. WITH THE RELOCATION OF THE FUEL DRAIN VALVE THE FUEL PASSES THROUGH THE FUEL FLOW TRANSMITTER WHEN DRAINING. THE RATE OF FLOW IS MUCH HIGHER THAN THE INDICATORS CAPABILITY WHICH CAN CAUSE DAMAGE TO THE INDICATOR IF THE INSTRUMENT AC BUS ENERGIZED BY THE "E", "A" OR "B" INVERTER.



Flow Flowmeter System — Schematic
Figure 1.



Indicator Error Curve for Flow Transmitter at 500 to 1200 LBS/HR Fuel Flow Range
Figure 2.

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FUEL FLOWMETER SYSTEM — MAINTENANCE PRACTICES

1. **Fuel Flowmeter Transmitter — Removal / Installation**

   A. Removal

      (1) With power on aircraft pull FIRE PULL T-handle to close main fuel shutoff valve.

      (2) Remove electrical power from aircraft.

      (3) Disconnect electrical connector from transmitter.

      (4) Disconnect fuel line at each end of transmitter and drain fuel from line.

      (5) Remove transmitter.

   B. Installation

      (1) Seal fittings with Loc Tite or equivalent.

      (2) Install transmitter.

      (3) Connect fuel lines to transmitter.

      (4) Connect electrical connector to transmitter.

      (5) Apply power to aircraft; reset FIRE PULL T-handle to open main fuel shutoff valve.

      (6) Remove electrical power from aircraft.

      (7) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

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FUEL QUANTITY INDICATION SYSTEM — DESCRIPTION / OPERATION

1. General

Two types of fuel indication system are utilized in the aircraft. They are as follows:

- Standard System - Integral Tank Only
- Increased Fuel Load System - Aircraft having ASC 125.

The two systems are covered separately, in this chapter, after the general description of capacitance fuel quantity system, which applies to all systems of this type.

2. Capacitance Fuel Quantity Systems — General Description

Fuel quantity is measured by two separate, capacitor type systems, one for each tank system (left and right). They are identical and independent of each other. One side will be discussed, the other being identical. Each system consists of an interconnected network of capacitance sensing devices, known as probes, or tank units, strategically located in the fuel system. The composite capacitance reading of the probes for the side involved is transmitted by electrical wiring to a combination transistorized indicator power unit, located on the cockpit center instrument panel. The indicator power unit (hereafter called the indicator) translates the capacitance information from the probes into visual indication of fuel quantity in terms of pounds of fuel. The indicator also contains a low level warning switch, operated by a cam on the pointer shaft, which completes a circuit to the L and R FUEL QUAN lights of the master warning system. The warning system indicates to the crew by illuminating its appropriate warning light, when the indicator hand is at 720 pounds or below. The indicator dial is also marked with a red band in the same area (720 to 0 pounds). A test circuit is supplied which will test the system. A single FUEL GAGE TEST button, on the pilots side of the eyebrow panel in the cockpit, tests both left and right fuel indication systems simultaneously. This test system is the only thing in common between left and right systems. The indicator has no stops and all internal mechanisms are capable of allowing 360 degree rotation of the indicating hand. When the FUEL indicator TEST button is depressed, both indicators slave downscale at approximately 30 degrees per second pointer movement. Releasing the button will cause the pointers to return to their original position.

NOTE: Because of the 360 degree capabilities of pointer travel, when the test button is released, the pointer may return to its original position by a clockwise or counterclockwise route, depending on its position when the button was released. This is not a malfunction. The important thing is that the pointers return to their original position.

The indicator is servo motor controlled, and desensitized, and thus will give stable readings, even in turbulent air conditions. Power for the operation of each system is obtained from the ac instrument bus, this requires an inverter to be operating in order for this system to function. Since the ac instrument bus is one of the two ac buses capable of being powered from the E-inverter, the indicators are operative in emergency dc. The fuel indicator test function, which is dc operated, is powered from the essential dc bus, and is also fully operative in emergency dc. However, the low fuel warning lights in the master warning system are inoperative in emergency.

3. Capacitance Fuel Quantity Systems — Principle of Operation

The system operation is based on the principle that capacitance for a given capacitor is determined by the dielectric constant of the insulating medium between the electrodes. The tank probes are capacitors whose capacitance can be varied by changing the dielectric constant between the electrodes. The dielectric in the tank probes is either liquid fuel, a mixture of air and fuel vapor, or a combination of the two. As the level in the tanks varies, so also does the capacitance of the probes.

The liquidometer capacitance type fuel quantity indication system operates on self-balancing capacitance bridge principle (rudimentary capacitance bridge) as shown in Figure 1 It consists of a transformer to supply alternating current, a tank unit C1 (variable capacitor), a fixed capacitor C2 and a meter connected from the center of the transformer to the junction of the two capacitors. By selecting the proper fixed capacitor, the circuit may be designed so that the meter will indicate a certain minimum value when the tank in which the tank unit is located is empty. As the tank is filled the bridge will become more and more unbalanced due to the

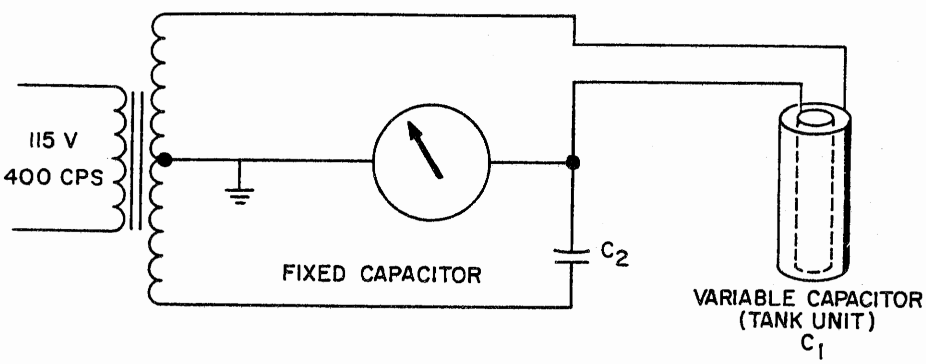
increase in tank unit capacitance. The meter pointer position when the tank is full can than be considered the full point of the indicator scale. This circuit represents a rudimentary fuel indicator.

Greater accuracy can be designed into the fuel gage shown in Figure 1 by allowing the capacitance bridge to null (reach a minimum output) at the fuel level to be measured. To do this, the voltage across the fixed capacitor C2 (see Figure 2) is varied to balance the capacitance voltage product of the tank unit in the other leg of the bridge. When the bridge equation $C_1V_1 = C_2V_2$ is satisfied, the bridge will balance (null). Electrically, this required the addition of a balancing potentiometer, as shown in Figure 2. Here the potentiometer is hand operated, and in addition to moving the potentiometer, the knob also turns a pointer on a scale. Thus, no matter what the fuel quantity may be, the potentiometer may be moved to the point that will bring the voltage bridge output to a minimum (null) value. At this point, the dial may be marked with the amount of fuel in the tank. By repeating this from empty to full, a dial can be calibrated so that by nulling the bridge, a fuel quantity indication is obtained. By selecting a meter that is extremely accurate at its zero position, greater accuracy can be introduced into the gaging system.

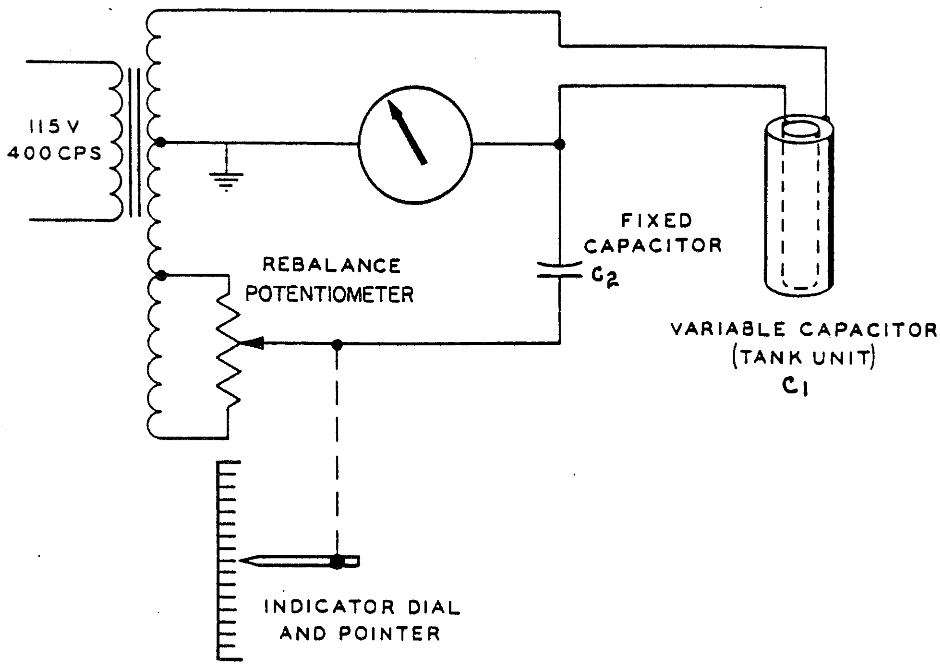
When the meter has been removed and the bridge output fed to an amplifier, as shown in Figure 3, the amplifier increases the strength of the output signal so that it can drive a phase sensitive induction motor. The motor replaces the hand driven knob and through a high ratio gear train moves the potentiometer automatically to the rebalance point. The pointer is also geared to the motor so that it indicates a potentiometer position, in terms of fuel quantity, on a suitable graduated dial. This is a basic capacitor type fuel gage.

In order to compensate for changes in fuel density, which is important to the accuracy of the system, the reference leg of the bridge must be corrected. This is accomplished by an additional capacitor unit (known as a compensator) mounted in the tank. It is usually placed at the bottom of the first probe to get wet and the last to go dry (in the case of multiple probe systems).

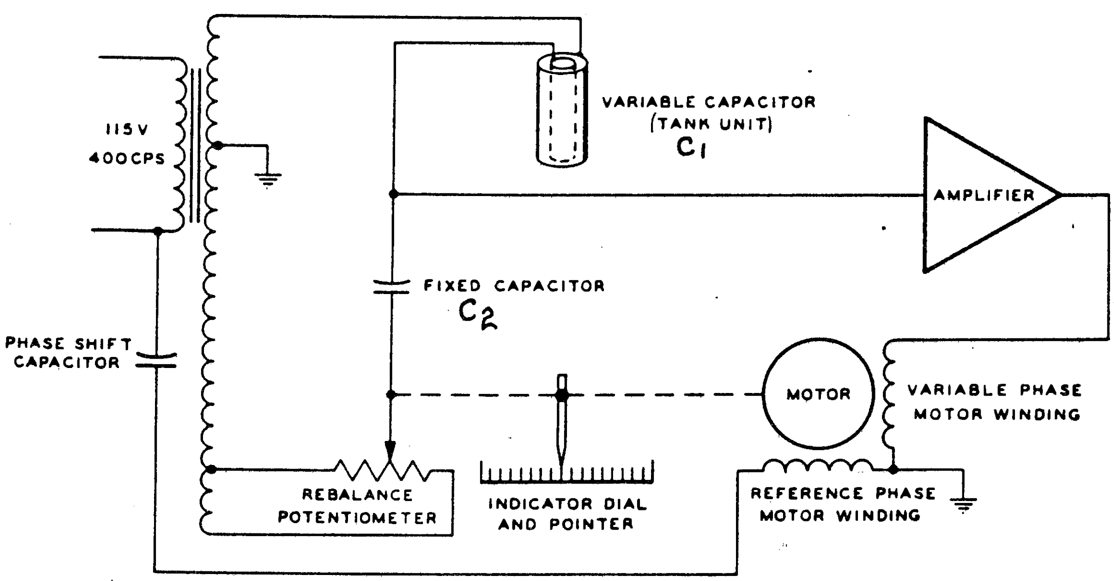
Since the variable leg of the bridge (probes) are continuously sensing dielectric constant changes of the fuel, or fuel-air mixture, the reference capacitor in the fixed leg of the bridge must also sense this change in dielectric constant in order to maintain a well balanced bridge with minimum error. This is the function of the compensator. Therefore, in any accurate capacitance type fuel gaging system, a compensator capacitor performs this important function.



Capacitance Bridge
Figure 1.



Simple Null-Balancing Capacitance Bridge — Schematic
Figure 2.



Basic Capacitor Fuel indicator — Schematic
Figure 3.

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FUEL QUANTITY INDICATION SYSTEM — DESCRIPTION / OPERATION

1. General Description

A. Standard System

The standard fuel indication system consists of two identical systems, one for the left integral tank and one for the right. One side will be described, the other being identical in all respect.

Four tank units are strategically mounted in the integral tank. They are referred to by number, with the most inboard as No.1, and extreme outboard as No.4. The indicator is located in the center instrument panel in the cockpit, and is calibrated from 0 to 5600 pounds. (See Figure 1) Low level warning red band and low level switch warning is from 720 pounds to 0 pounds. The probes are interchangeable side to side with its counterpart in the other tank, (No.1 is the same part number in the left and right tanks, etc.) but can only be used in the same position in the respective tank. Since each probe is calibrated on manufacture for its position in the integral tank, you cannot replace a No.1 with a No.3 probe or vice versa.

The four probes are electrically connected in parallel by special connectors at the rear wing beam. One shielded wire and one unshielded wire connect the probe syste′ to its appropriate indicator. (See Figure 2) The No.1 probe also contains a third wire, for the use of the compensator unit, and it also is connected to the indicator. The probes are suspended in a vertical position, held in the tank by bra-cketry on the upper and lower wing planking. They are accessible through access holes in the upper cover. (the No.4 probe is accessible through the fuel tank filler neck.) Special fuel resistant wiring manu-factured as part of each probe connects to fuel proof fittings which exit from the tank through holes in the rear wing beam. The probe to fitting wires are part of the probe and cannot be repaired or replaced except at an approved overhaul agency equipped for this type of work.

The indicator contains all other components necessary for the operation of the system, with the excep-tion of the FUEL GAGE TEST button which is located on pilots side of the eyebrow panel and the fuel indicator test relay. (COPILOTS RELAY BOX)

Power is supplied to the system through the L and R FUEL indicator circuit breakers, located on the copilots (right) circuit breaker panel. The FUEL GAGE TEST button receives dc power from the essential dc bus through the MA WARN circuit breakers. All connections to the indicator are made by two plugs at its rear. One is a miniature, multiple pin, twist lock connector, and the other is a dage single conductor, shielded wire connector.

B. Modified System

Aircraft having ASC 125 have the same fuel quantity indication system as the standard aircraft with the following exceptions:

(1) Six tank units (probes), instead of four, are strategically mounted in the tank system. They are referred to by numbers with the most inboard as number 1 and extreme outboard as number 6. They are connected in parallel, by special connectors at the wing rear beam.

(2) The indicator is calibrated from 0 to 6300 pounds, instead of 0 to 5600 pounds.

(3) The number 5 and 6 probes are left and right and cannot be interchanged side to side, even in the same location.

NOTE: Later models are equipped with modified No.1 and No.3 tanks units identified with ″M″ suffix. Probes having the same part No. with or without the suffix ″M″ are interchangeable with each other.

The characteristics of probes 5 and 6 are different because of the difference in size of the number 6 bladder cells. The left number 6 cell is slightly smaller, due to the fact that allow-ance must be made on the left side for passage of the aileron trim cables. Since the overall tank capacities are different, the two outboard probes must be different and mated to that particular side.

2. Major Components

A. FUEL QUANTITY indicators (2), Liquidometer type B118-75, one for each tank, mounted in the center instrument panel.

B. Fuel indicator test relay, located in the copilots relay box.

C. FUEL GAGE TEST button, located in the eyebrow panel, left side.

D. Fuel probes (tank units) as follows (one of each in each side):

Position	Part No. (Liquidometer)	Name
#1	EA-818-1901 or 1901M	Tank Unit with Compensator
#2	EA-772-1902	Tank Unit
#3	EA-772-1903 or 1903M	Tank Unit
#4	EA-772-1904	Tank Unit *
#4	EA-772-2356	Tank Unit
#5	EA-772-2357R (right) EA-772-2357L (left)	Tank Unit* Tank Unit
#6	EA-772-2358R (right) EA-772-2358L (left)	Tank Unit Tank Unit
*** AIRCRAFT HAVING ASC 125.:**		

3. Fuel Probe Capacitances

A. Dry (measured at probe plug)

Probe No.	Tank Part No.	Dry Capacitance (MMfd)	
		Standard (four probes)	Modified (six probes)
1	EA-818-1901 *EA-818-1901M	22.70 ±0.5 *22.01 ±0.5	22.70 ±0.5 *22.01 ±0.5
2	EA-772-1902	42.95 ±0.5	42.95 ±0.5
3	EA-772-1903 or 1903M	22.51 ±0.5	22.51 ±0.5
4	EA-772-1904 EA-772-2356	20.94 ±0.5 N/A	N/A 20.95 ±0.5
5	EA-772-2357 R or L	N/A	8.66 ±0.5
6	EA-772-2358 R or L	N/A	8.85 ±0.5
Total dry capacitance with probes in parallel.		109.10 ±20	124.62 ±3.0
* Total dry capacitance with "M" types probes in parallel (Aircraft 63 - 200, 322 and 323, excluding 114).		108.41 ±2.0	123.93 ± 3.0

B. Wet Capacitances: (4 probes in Parallel With S.G. 0.807 - 6.72 lb /gal)

Standard type: (Aircraft 1 - 62 and 114)

Dry	109.10	± 2.0 mmfd
Added	133.56	± 0.0 mmfd
Wet	242.66	± 2.0 mmf

"M" type: (Aircraft 63 - 200, 322 and 323)

Dry	108.41	± 2.0 mmfd
Added	133.56	± 0.0 mmfd
Wet	241.97	± 2.0 mmf

C. Wet Capacitances: (6 probes in Parallel With S.G. 0.807-6.72 lb/gal)

Standard type: (Aircraft 1 - 62 and 114)

Dry	124.62	± 3.0 mmfd
Added	149.61	± 0.0 mmfd
Wet	274.23	± 3.0 mmf

"M" type: (Aircraft 63 - 200, 322 and 323)

Dry	123.93	± 3.0 mmfd
Added	149.61	± 0.0 mmfd
Wet	273.54	± 3.0 mmf

D. Compensator Capacitance Measured at No.1 Probe Plugs

Dry	25.00	± 0.5 mmfd
Added	31.05	± 0.0 mmfd
Wet	56.05	± 050 mmf

NOTE: The compensator capacitance (dry and wet) for aircraft having ASC 125 is the same as aircraft not having ASC 125

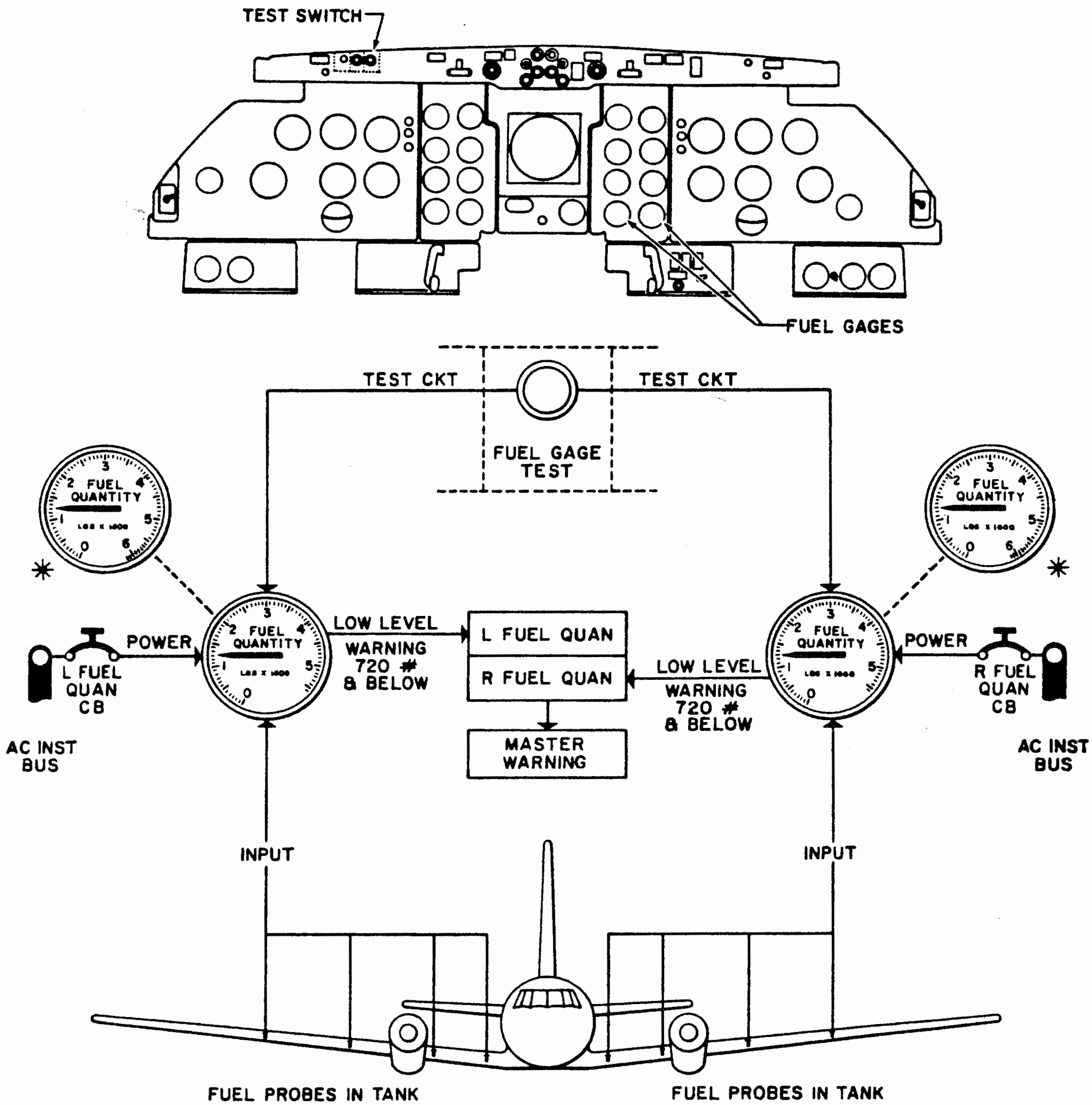
4. Low Fuel Level Warning System

NOTE: The low level warning system is the same for standard and modified (ASC 125) aircraft.

Low fuel level warning is a function of indicator needle position. A low level warning switch is part of each indicator, and is operated by a cam attached to the indicator needle shaft. The switch is set to close when the needle is in the red-banded area on the dial; 720 pounds and below. In this area, a circuit is completed to the master warning system to illuminate the appropriate L or R FUEL QUAN capsule and associated master warning lights. The switch and cam assembly cannot be adjusted in the field. Two different variations to this system are utilized, depending on which master warning system is installed as follows:

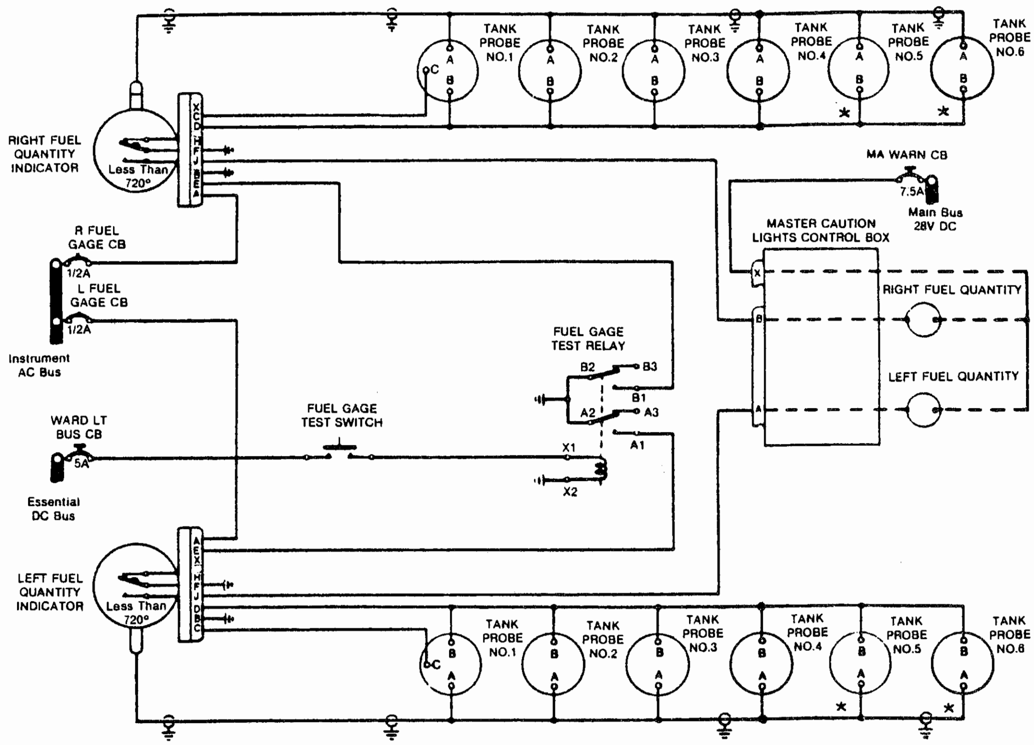
- Aircraft 1 - 120, 322 and 323 (30 capsule master warning system) (See Figure 2). A negative switching circuit is employed. The switch completes a ground to the appropriate section of the master warning system. The positive feed is common within the master warning system, and is provided from the main dc bus through the MA WARN circuit breaker.
- Aircraft 121 - 200 (32 capsule master warning system) (See Figure 3). A positive switching circuit is employed. The switch completes a positive feed from the MA WARN circuit breaker to the appropriate section of the master warning system. The ground feed is common within the master warning system.

The low fuel level warning lights function of the fuel quantity system is inoperative in emergency dc, however, all other functions of the indication system are fully operative.



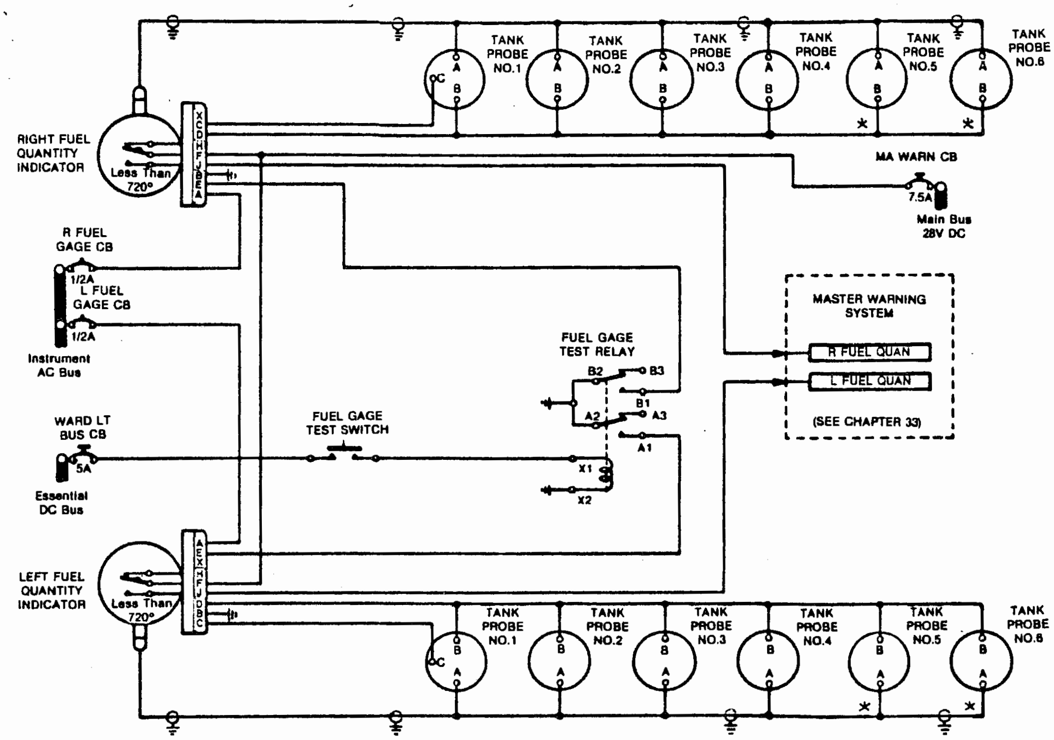
✳ Aircraft 1 through 200 including 322 and 323 having ASC 125 incorporated

Fuel Quantity System (Standard System)
Figure 1.



*Aircraft 1 through 200 including 322 and 323 having ASC 125 incorporated.

Fuel Quantity System — Schematic
(Aircraft 1 - 200, 322 and 323)
Figure 2.



*Aircraft 1 through 200 including 322 and 323 having ASC 125 incorporated

Fuel Quantity System — Schematic
(Aircraft 121 - 200 only)
Figure 3.

FUEL QUANTITY INDICATION SYSTEM — FAULT ISOLATION

1. Fuel Quantity Indication System Indicator

NOTE: The following table covers standard and modified (ASC 125) system.

FAULT	POSSIBLE CAUSE	CORRECTION
Indicator pointer does not move when tanks are being filled.	Power not supplied to system.	Check circuit breaker and 115 volt, 400 cycle power source.
Low level warning light remains on when more than 720 pounds of fuel is indicated.	Defective switch in indicator	Replace indicator.
	Defective wiring.	Check wiring.
Fuel tanks filled, indicator pointer fails to move counter-clockwise when test switch is depressed.	Defective indicator.	Replace indicator.
	Defective test circuit.	Check test circuit.
Indicator pointer moves counter-clockwise, low level warning light fails to come on at 720 pounds or less.	Defective indicator switch.	Replace indicator.
	Defective switch wiring.	Check circuit.
	Defective warning light lamps.	Depress warning light test switch. If light fails to come on, replace defective lamp or lamps.
Indicator pointer oscillates.	Defective indicator.	Replace indicator.
Indicator pointer continues to rotate in counterclockwise direction.	Open circuit between probe and indicator	Repair or replace defective wiring.
	Shielded lead from probes is grounded.	Minimum resistance to ground is 20 megohms, if less, replace lead.
Indicator pointer creeps	Probe connectors shorted.	Repair or replace.
Indicates rotates counterclockwise continuously (single indicator, single failure).	High impedance short to ground at probe or leads.	Isolate and eliminate short (external. Replace probe (internal).
	Low impedance short to ground at probe or leads.	Isolate and eliminate short (external). Replace probe (internal).
Indicator rotates counterclockwise continuously (single indicator, single failure).	Compensator shorted to ground at probes or leads. (Wet No. 1 probe.)	Isolate and eliminate short (external). Replace No. 1 probe (internal).
	Open probe leads (high impedance, low impedance, or compensator). (Wet No. 1 probe.)	Isolate and repair open probe lead (external). Replace probe (internal).
	Short to ground in indicator plug pin "E" or in wiring to this pin.	Isolate and eliminate short. Replace plug.
	Loose or defective Dage connection at probes or indicator.	Tighten loose connector, or replace defective connector.
	Broken wire in high impedance, low impedance, compensator leads between indicator and probe connectors.	Isolate and repair broken wire, (external). Replace probe (internal).
	Defective indicator.	Replace indicator.

FAULT	POSSIBLE CAUSE	CORRECTION
Both indicators rotate counterclockwise continuously (single failure).	Fuel indicator test relay - both contacts frozen closed.	Remove and replace relay.
	Hot feed on fuel indicator test relay contact X1.	Locate and remove hot feed.
	Shorted fuel gage test switch.	Locate and eliminate short. Replace switch.
Indicator rotates clockwise continuously (single indicator, single failure).	High impedance shorted to low impedance at probe or leads.	Isolate and eliminate short (external). Replace probe (internal).
	Defective indicator	Replace indicator.
Indicator reads high but does not rotate clockwise (single indicator, single failure).	Water or moisture in one or more Dage connectors.	Blow dry with air.
	Water or moisture in one or more probes.	Blow dry with air, or remove probe and permit to dry.
Indicator reads high but does not rotate clockwise (single indicator, single failure).	Open compensator or lead with low fuel levels. (Possible full clockwise rotation with high fuel levels.)	Isolate and eliminate trouble (external). Replace No. 1 probe (internal).
	Short between compensator and low impedance lead at probes or leads.	Isolate and eliminate short (external). Replace No. 1 probe (internal).
Indicator does not move (single indicator, single failure).	No power at indicator. (Wire broken, circuit breaker blown or pulled.)	Check power source wiring, and circuit breakers.
	Defective indicator.	Replace indicator.
Both indicators do not move (single failure).	Inverter not running. Instrument AC Bus not energized.	Check inverter operation. Check instrument AC Bus.

FUEL QUANTITY INDICATION SYSTEM — MAINTENANCE PRACTICES

1. Fuel Quantity System — Calibration

A. Calibration procedure using liquidometer B575 tester: (and B575-2 adapter for aircraft having ASC 125)

(1) Drain tank involved using normal defueling procedures.

(2) Complete tank draining using water drains. Close drains when dry.

(3) Add exactly 5 U.S. gallons to tank involved.

(4) Remove indicator. (See this Section.)

(5) Open adjustment screw covers at rear of instrument.

(6) Attach harness between instrument and plugs, placing indicator so that adjustment screws are accessible for adjustment and connect test cable to tester.

(7) Set box selector to position 3 (system test).

(8) Turn power on in aircraft (battery switch - NORM, external power plugged in with EXT PWR switch on). Turn E-inverter on by placing ESSENTIAL AC bus switch to NORM.

(9) Allow system to warm up for 5 minutes.

(10) Adjust EMPTY screw on back of indicator until 0 indication is obtained.

(11) Place test box selector switch to POSITION 4 (system test).

(12) Indicator hand should go clockwise and stop at approximately:

(a) 5,600 pounds for Aircraft not having ASC 125.

(b) 6,300 pounds for Aircraft having ASC 125.

(13) Adjust FULL screw on back of indicator until following indication is obtained:

(a) 5,600 pounds for Aircraft not having ASC 125.

(b) 6,300 pounds for Aircraft having ASC 125.

(14) Remove compensator lead from inboard probe by disconnecting 90° angle fitting at aft wing beam of tank which is being calibrated, leaving connector disconnected. Note that indicator moves from the following and may rotate continuously.

(a) 5,600 pounds for Aircraft not having ASC 125.

(b) 6,300 pounds for Aircraft having ASC 125.

NOTE: If above action does not occur, do not continue with procedure. Repair malfunction and repeat procedure from Step (7) above.

(15) Connect compensator lead connector to receptacle at aft wing beam. Indicator should return to:

(a) 5,600 pounds for Aircraft not having ASC 125.

(b) 6,300 pounds for Aircraft having ASC 125.

(16) Depress and hold FUEL GAGE TEST button on pilot side of eyebrow panel. Both fuel indicators should drive down scale. Check that low fuel warning light comes on when applicable fuel indicator indicates pounds or below (red band area on dial).

(17) Release test button. Indicators should return to their previous position.

(18) Repeat Steps (7), (10), (11), (12) and (13) until error at:

(a) 0 and 5,600 range is eliminated. Aircraft not having ASC 125.

(b) 0 and 6,300 range is eliminated. Aircraft having ASC 125.

(19) Turn off inverter and disconnect test harness from indicator and plugs. Install adjustment screw covers at rear of instrument and safety covers.

(20) Connect plugs at rear of indicator and install indicator.

(21) Turn on inverter and check that indicator hand goes to 0 and that test switch drives down scale.

(22) Turn off inverter and power.

B. Calibration procedure using capacitance injection unit other than B575 tester:

(1) Test harness must be fabricated to enable capacitance injection box to be placed in parallel with probe leads with mating plugs to join with indicator plugs, and also to enable injection unit to be placed in or out of circuit. (Shielded wire integrity must be maintained).

(2) Box must be capable of injecting the following accurately in parallel with probes:

(a) 97.6 to 98.7 pF for Aircraft not having ASC 125.

(b) 111.4 pF for Aircraft having ASC 125.

(3) With harness hooked up but box not hooked in, turn aircraft power on. (Battery switch - NORM and external power plugged in with EXT PWR switch on). Turn E-inverter on by placing ESSENTIAL AC bus switch to NORM.

(4) Allow system to warm up for 5 minutes.

(5) Adjust EMPTY screw on back of indicator until 0 indication is obtained.

(6) Place injection unit in parallel with probe leads at harness.

(7) Set injection unit to:

(a) 97.6 pF for Aircraft not having ASC 125.

NOTE: Aircraft with M-type No.1 probes set in 98.7 pF.

(b) 111.41 pF for Aircraft having ASC 125.

(8) Indicator should slave upscale to approximately:

(a) 5,600 pounds for Aircraft not having ASC 125.

(b) 6,300 pounds for Aircraft having ASC 125.

(9) Adjust FULL screw on back of indicator until the following indication is obtained:

(a) 5,600 pounds for Aircraft not having ASC 125.

(b) 6,300 pounds for Aircraft having ASC 125.

(10) Remove compensator lead from inboard probe by disconnecting 90° angle fitting at aft wing beam of tank which is being calibrated, leaving connector disconnected. Note that indicator moves from the following and may rotate continuously:

(a) 5,600 pounds for Aircraft not having ASC 125.

(b) 6,300 pounds for Aircraft having ASC 125.

NOTE: If above action does not occur, do not continue with procedure. Repair malfunction and repeat procedure from Step (5) above.

(11) Connect compensator lead connector to receptacle at aft wing beam. Indicator should return to:

(a) 5,600 pounds for Aircraft not having ASC 125.

(b) 6,300 pounds for Aircraft having ASC 125.

(12) Depress and hold FUEL GAGE TEST button on pilot side of eyebrow panel. Both fuel indicators should drive down scale. Check that low fuel warning light comes on when applicable fuel indicator indicates 720 pounds or below (red band area on dial).

(13) Release test button. Indicators should return to their previous position.

(14) Disconnect injection box from harness. Indicator should return to approximately zero.

(15) Repeat Steps (5), (6) and (7) above until all errors are out of system and needle points exactly to:

(a) 0 and 5,600 range is eliminated. Aircraft not having ASC 125.

(b) 0 and 6,300 range is eliminated. Aircraft having ASC 125.

(16) Turn off inverter and remove all test harnesses. Install adjustment screw covers at rear of instrument and safety covers.

(17) Install indicator (See this section).

C. Fueling check after adjustment:

NOTE: Perform the following check after adjustment is completed. It is intended to ascertain overall accuracy of indication system versus actual amounts of fuel in tank.

(1) Jack aircraft to 0° laterally and longitudinally using normal jacking procedures.

(2) Sample fuel to be used with hydrometer and ascertain number of pounds per gallon. Compute how much fuel shall be added in order to obtain following actual weight of fuel based on that density of fuel:

100 pounds	1,000 pounds
200 pounds	1,500 pounds
300 pounds	2,500 pounds
400 pounds	3,500 pounds
500 pounds	4,500 pounds
	5,500 pounds (Aircraft having ASC 125)

(3) Set truck or measuring unit to zero.

CAUTION: WHEN ADDING FUEL AT LOW LEVELS (0-1000 POUNDS), LET FUEL SETTLE FOR AT LEAST 10 MINUTES BEFORE READING INDICATOR. FROM 1000 - 2500 POUNDS, WAIT AT LEAST 5 MINUTES FOR FUEL TO SETTLE BEFORE READING INDICATOR. ABOVE 2500 POUNDS, INDICATOR MAY BE READ IMMEDIATELY.

(4) Add fuel to tank in amount calibrated to give 100 pounds.

(5) Let fuel settle for at least 10 minutes.

(6) Turn on electrical power, EXT PWR switch and E-inverter.

(7) Read indicator and check against tolerance listed below.

(8) Add additional fuel so that actual total added fuel amount equals

Actual Fuel Load (Pounds)	Tolerance (Pounds)
0	±0
100	±25
200	±53
300	±80
400	±103
500	±132
1000	±155
1500	±175
2500	±215
3500	±255
4500	±294
5500	±340 (Aircraft having ASC 125)

(9) Return aircraft to normal configuration.

(10) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

(11) Remove electrical power from aircraft.

2. Fuel Quantity Indicator — Operational Test

A. Energize main and essential dc buses and instrument ac bus.

B. Depress and hold FUEL GAGE TEST button on eyebrow panel.

C. Observe that both indicators slave downscale (counterclockwise).

D. Observe that appropriate FUEL QUAN warning light on master warning panel comes on when associated indicator hand is in red band area (720 pounds or below).

E. Release test button. Indicators shall return to their original position.

F. Remove electrical power from aircraft.

3. Fuel Flow Indicator — Removal / Installation

A. Removal:

(1) Remove electrical power from aircraft.

(2) Disconnect electrical connector.

(3) Remove indicator.

B. Installation:

(1) Install indicator.

(2) Connect electrical connector to indicator.

(3) Perform engine run and check indicator for proper operation.

4. Fuel Quantity Indicator — Removal / Installation

A. Removal:

(1) Remove electrical power from aircraft.

(2) Gain access to indicator in cockpit center instrument panel.

(3) Disconnect connectors at rear of indicator.

(4) Remove indicator.

NOTE: If indicator is replaced, perform FUEL QUANTITY CALIBRATION.

B. Installation:

(1) Install indicator.

(2) Connect plugs at rear of indicator.

(3) Energize main and essential dc buses and instrument ac bus.

(4) Check that indicator tests properly and lower level warning light comes on in red band area.

(5) Release test switch and indicator shall return to zero.

(6) Remove power.

5. Fuel Quantity Probes — Removal / Installation

A. Removal:

(1) Remove electrical power from aircraft.

(2) Locate probe by finding rear beam connection. Probe is located under closest chordwise upper camber handhole. Do not open handhole at this time.

NOTE: For identification only, probes are numbered one through four for Aircraft not having ASC 125 and one through six for aircraft having ASC 125. The most inboard probe is No.1 and proceeding spanwise, respectively for each wing to No.2,3,4 (No.5 and No.6 for Aircraft having ASC 125). No.1 - No.4 are in inner panel. No.5 and No.6 are in two outer panel cells for Aircraft having ASC 125.

Probes No.1, No.2 and No.3 have same part numbers as Aircraft having/not having ASC 125. Probe No.4 part number is different for Aircraft having ASC 125.

(3) Defuel wing involved using normal defueling procedures until probe involved is dry. (No.1 probe replacement involves complete defueling).

(4) Open handhole.

(5) Disconnect Dage connectors at rear beam (three connectors for No.1 probe, two for all others).

(6) Disconnect wing beam connection by having one man hold connector inside of tank (working through handhole) and other loosening and removing locknut and washer at rear beam.

(7) Remove probe mounting bolts from supports inside tank, remove probe.

B. Pre-installation Check: (Recommended but not mandatory)

(1) Before installing probe, check probe insulation factor using a 50V dc megger or TF-20 or equivalent. Minimum resistance is 500 megohms for all checks listed below. Check all connections as follows:

NOTE: Probe for test must be dry.

(a) High impedance (white plug insert) to ground (mounting ear).

(b) Low impedance (orange plug insert) to ground (mounting ear).

(c) Compensator (yellow plug insert) to ground (mounting ear) (No.1 probe only).

(d) High impedance (white plug insert) to low impedance (orange plug insert).

(e) High impedance (white plug insert) to compensator (yellow plug insert) (No.1 probe only).

(f) Low impedance (orange plug insert) to compensator (yellow plug insert) (No.1 probe only).

C. Installation:

(1) Mount probe to brackets inside of tank.

(2) Ensure probe connector has an O-ring installed in recess before installing connector through wing beam.

(3) Remove lockwasher and locknut from connector and with O-ring in recess, insert connector through wing beam. Outside man then secures connector with washer and locknut while other man holds connector inside tank. Torque locknut to 15 - 20 inch-pounds max.

(4) Connect outside plugs to wing beam connectors.

(5) Install wing handhole. Ensure all screws are tight.

(6) Energize main and essential dc buses and instrument ac bus and check indicator for normal test reaction.

(7) Perform Fuel System — Bleed Procedure, see Section 28-0-1.

6. Probes — Insulation Check

NOTE: This check involves the use of 50 volt megger. The MD-2 tester or equivalent may be used.

CAUTION: WHETHER THE PROBE IS INSTALLED IN TANK OR TESTED ON THE BENCH, THE FOLLOWING CONDITIONS ARE MANDATORY.

PROBE MUST BE DRY.

RELATIVE HUMIDITY SURROUNDING PROBE MUST BE LESS THAN 50%.

A MEGGER OF NO HIGHER THAN 50V DC MUST BE USED.

IF INSTALLED IN AIRCRAFT, PROBE MUST BE DISCONNECTED FROM REST OF SYSTEM BY REMOVING CONNECTORS AT REAR BEAM. THIS CHECK IS VALID FOR THE PROBE AND ITS LEADS ONLY, NOT THE AIRCRAFT INTERCONNECTING WIRING.

NOTE: The unshielded wire connection of each probe (connector color coded orange) is called the low impedance side. The shielded wire connection of each probe (connector color coded white) is called the high impedance side.

The third connection for the No.1 probe (color coded yellow) is referred to as the compensator lead.

Ground refers to aircraft structure (if installed in tank) or metallic body of probe (if on the bench).

A. Connect megger leads as indicated, checking all combinations as follows:

Megger readings must be 500 megohm minimum for all:

(1) High impedance to ground
(2) Low impedance to ground
(3) Compensator to ground
(4) High to low impedance
(5) High to compensator
(6) Low impedance to compensator

7. **System Wiring — Insulation Check**

CAUTION: NEVER USE MEGGER ON THE SYSTEM WITH THE INDICATOR PLUGS CONNECTED

NOTE: This check involves the use of a 50V megger. The system wiring insulation resistance is measured with all system components disconnected (indicator, all probes and compensator).

A. Minimum allowable values are as follows:

(1) Ground to high impedance wiring (shielded cable and connectors) 100 megohms

(2) Ground to low impedance wiring (unshielded cable and connectors) 30 megohms

8. **Fuel Quantity Tank Units — Rear Wing Beam Electrical Connections**

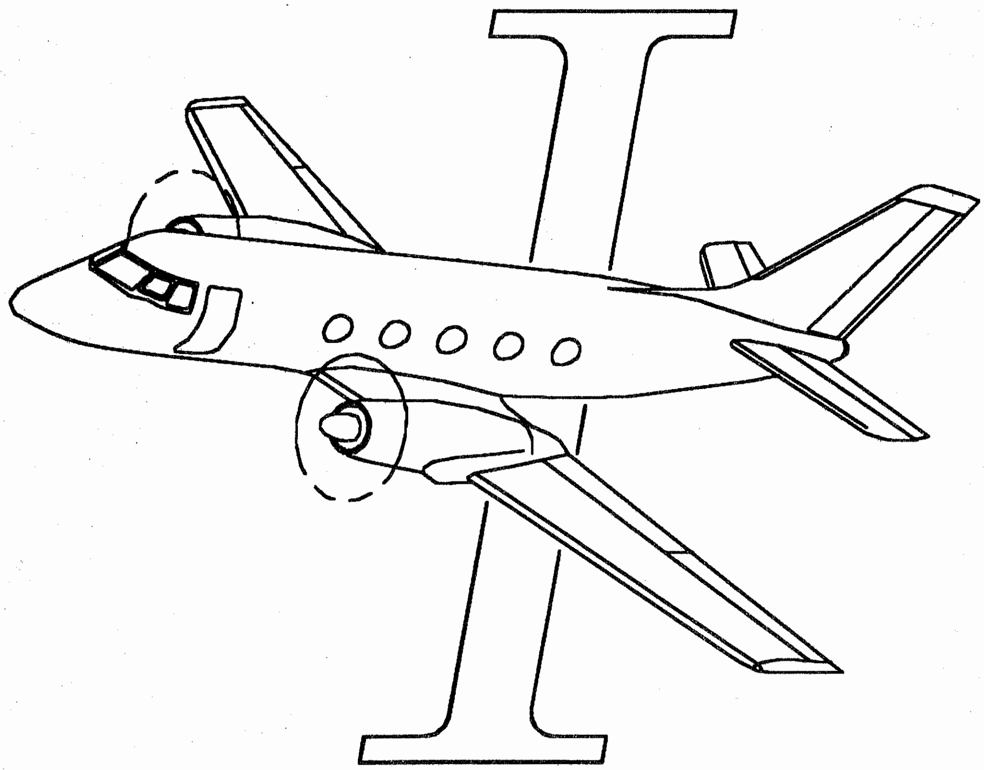
A. When installing tank units, each electrical connection (supplied as part of tank unit wiring) requires an O-ring in the recess of each receptacle. Each connection also requires a lockwasher and locknut (supplied as part of the tank unit) to secure connections to rear wing beam.

B. When installing these connections through rear wing beam, Hold portion of connector inside tank securely while tightening locknut. Torque locknut to 15 - 20 inch-pounds.

GULFSTREAM I

MAINTENANCE MANUAL

GULFSTREAM



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CHAPTER 29

HYDRAULIC POWER

HYDRAULIC POWER SYSTEMS — DESCRIPTION / OPERATION

1. General

The Gulfstream I has a 1500-psi hydraulic power system. The system is divided into three subsystems - main, auxiliary and emergency (See Figure 1 and Figure 2).

The main hydraulic power system supplies 1500 psi fluid pressure to operate the following systems: wing flaps, landing gear, speedbrakes, wheel brakes, windshield wipers and nosewheel steering.

The auxiliary hydraulic power system supplies 1500 psi fluid pressure to operate the following systems: emergency wing flaps, main and nose gear doors (for ground servicing), propeller brake (left engine only), parking and emergency brakes, auxiliary brakes, door and stairs.

The emergency air system supplies 1950 psi pneumatic pressure for emergency extension of the landing gear only.

2. Reservoirs

The dual-chambered hydraulic reservoir has a total capacity of 4.9 US gallons (4.08 Imperial gallons; 18.55 liters) of hydraulic fluid which is not readily flammable. The working capacity of the main system chamber is 3.1 gallons; the capacity of the auxiliary system chamber is 1.8 gallons, which is reserve. The reservoir is located on the aft side of Fuselage Station 169 bulkhead. The reservoir is a welded aluminum unit with an emergency filler neck and filler plug, and a normal filler neck and filler plug. The normal filler is used at regular periods or when needed during ground servicing. The emergency filler may be used during flight. Screens are incorporated in the filler necks to filter out dirt and foreign objects. A glass sight gage is installed in the reservoir for use by ground and flight personnel in determining fluid level. The lowest marking on the sight gage is the REFILL LEVEL, the middle marking is the GROUND FULL LEVEL and the highest marking is the FLIGHT FULL LEVEL. The filler plugs are attached to the filler necks with chains to prevent their loss. Hydraulic fluid is supplied to the main hydraulic system through two outlet port fittings at the right and left side near the bottom of the reservoir. Hydraulic fluid is supplied to the electrically-driven hydraulic pump through one auxiliary outlet fitting at the bottom center of the reservoir. Two separate drain fittings, one for the main and one for the auxiliary chamber, are provided for draining hydraulic fluid from the reservoir. The drain fitting for the main chamber only is on the reservoir. The drain fitting for the emergency system is on the suction line tee at the auxiliary hydraulic pump. Hydraulic fluid from all the systems is returned to the reservoir through five inlet ports. The reservoir should be filled when the aircraft is on the ground, or on jacks and with the landing gear fully extended, the airstair door open and the brake accumulator charged. The reservoir should be kept filled to the GROUND FULL LEVEL shown on the sight gage. The FLIGHT FULL LEVEL should be checked with the aircraft in a clean configuration; landing gear up and locked, airstair door closed and brake accumulator charged.

A vent and relief valve is provided to protect the reservoir against excessive positive and negative pressure. It is located near and above the reservoir. By maintaining a head on the hydraulic fluid in the reservoir, the valve minimizes cavitation in the engine driven hydraulic pumps and the electrically driven auxiliary hydraulic pump. The relief valve is designed to crack when air pressure trapped in the reservoir is 16 $\pm$ 1 psi above outside barometric pressure. The vent valve is designed to open when cabin pressurization is 0.5 psi above the trapped air in the reservoir. A filter is installed in the air inlet to eliminate dirt and foreign objects. The relief and vent ports are open to cabin and atmosphere respectively. The reservoir port is connected to the top center fitting of the hydraulic reservoir.

3. Hydraulic Line Identification

To facilitate servicing and maintaining the hydraulic power system, each line in the system is tagged with an identification band which indicates the blueprint number, line number, line code and the tubing size (see listing and example which follow). The bands are placed near each end of every line and at locations where they will assist in identifying the system.

CODE LETTERS	DEFINITION	CODE LETTERS	DEFINITION
P	Pressure	WW	Windshield Wiper
S	Suction	LGE	Landing Gear Emergency
R	Return	LGD	Landing Gear Down
V	Vent Drain	LGU	Landing Gear Up
BE	Brake Emergency	LWB	Left Wheel Brake
BR	Brake Return	RWB	Right Wheel Brake
DC	Door Close	LGED	Landing Gear Emer Down
DO	Door Open	WFD	Wing Flap Down
ER	Emergency Return	WFED	Wing Flap Emer Down
PB	Prop Brake	WFU	Wing Flap Up
D	Drain	WFEU	Wing Flap Emer Up

A. Example:

```
 ______________________________Denotes aircraft model
|     _________________________Hydraulic
|    |     ____________________Drawing number
|    |    |     _______________Line number
|    |    |    |     __________Line code letter
|    |    |    |    |     _____Tubing size
|    |    |    |    |    |
|    |    |    |    |    |
|    |    |    |    |    |
159  H10002 -1  -P   -4
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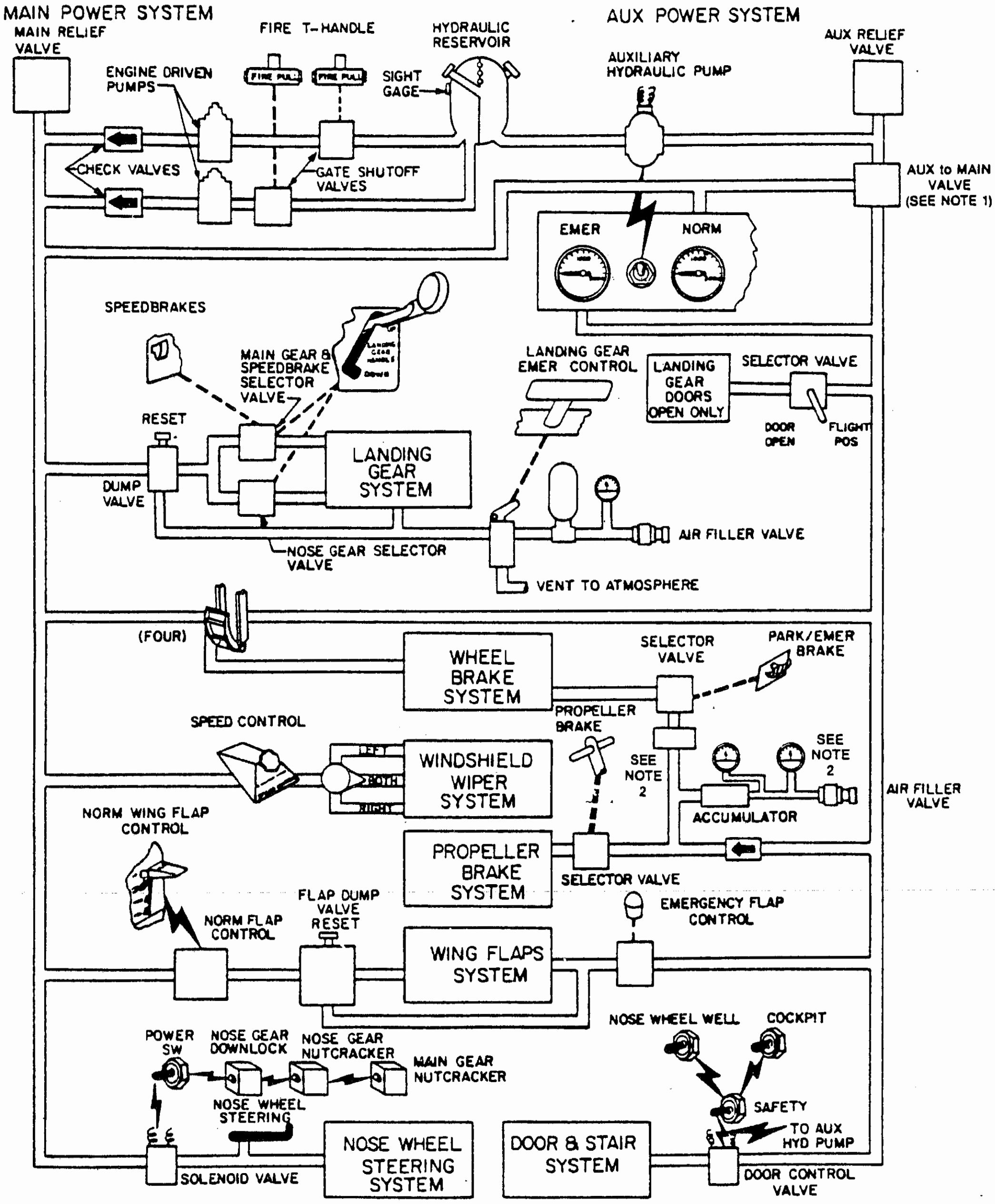
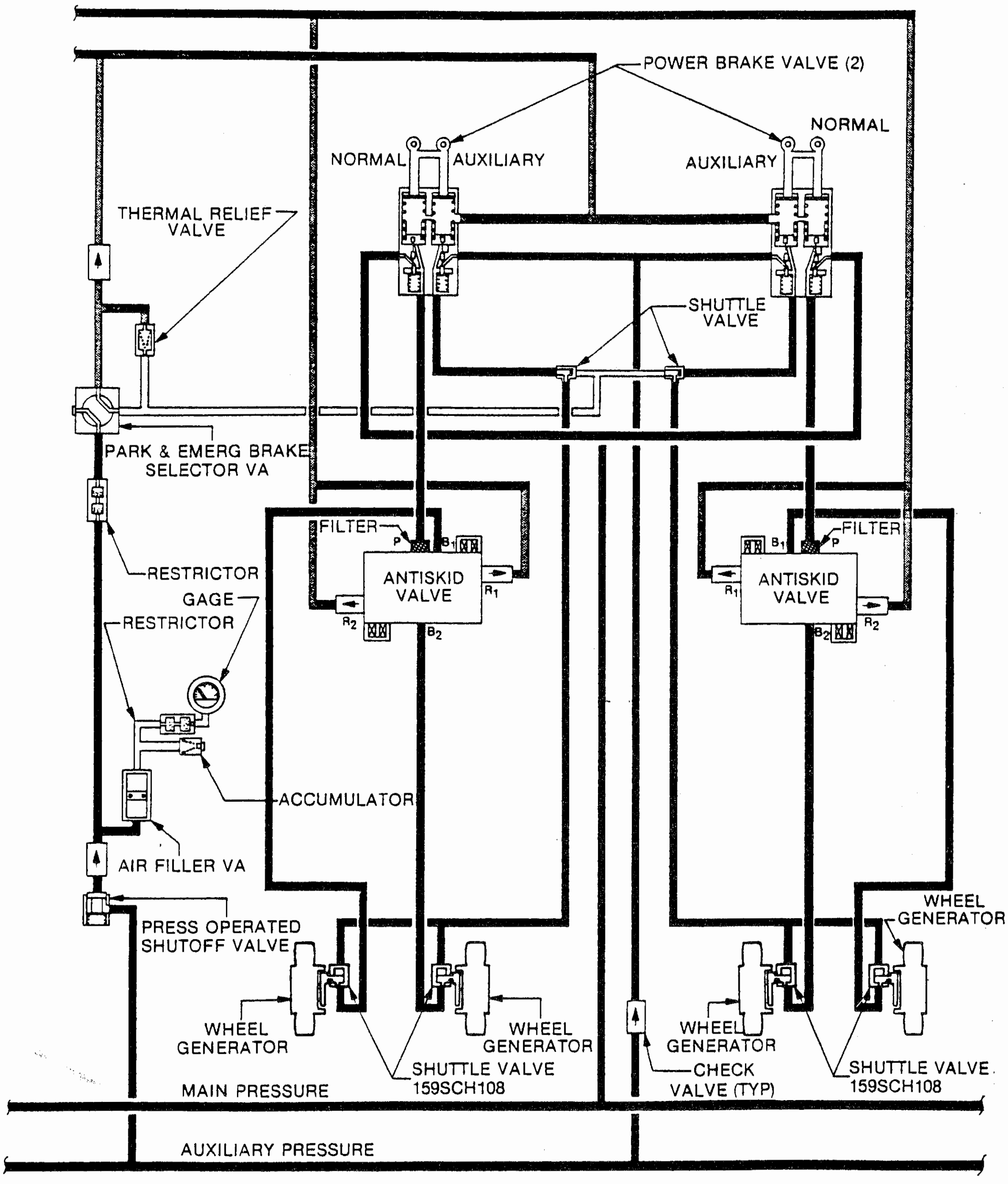


NOTES:

1 Aircraft 1 — 200, including 322 and 323, having ASC 109.

2 Aircraft 1 — 200, including 322 and 323, having ASC 214.

General Hydraulic Schematic
Figure 2.



Antiskid Hydraulic Schematic
Figure 3.

HYDRAULIC POWER SYSTEMS — FAULT ISOLATION

1. General

Common malfunctions of the main and auxiliary power systems usually are caused by: air in the system; leakage; clogged lines or fittings. Fault in a particular unit usually can be traced to one or more of the following conditions; leaks, either internal or external; foreign particles clogging or holding open some part of a unit; improper adjustment; mechanical damage or structural failure; or excessive clearances resulting from wear.

When the fault is isolated to a particular unit, remove that unit from the aircraft. Thoroughly flush the unit with clean hydraulic fluid, identical to that used in the aircraft hydraulic system, then perform a functional test to check its operation.

NOTE: When a hydraulic unit has been changed, or if hydraulic lines have been disconnected for any reason, cycle the unit to return trapped air to the reservoir. With the system under pressure, check for leaks. Ensure that system is operating properly and that there are no leaks before placing the aircraft in service.

A. Tubing Leaks and Failures

Fault in a tubing system may be classified in three broad groups: leaks, clogged lines and failures.

(1) Causes of leaks at flared joints:

- Poor flare, rough surface, cracks and splits.
- Cracks in coupling sleeve.
- Improper torque.
- Insufficient or improper support of lines.
- Damage to flares.
- Foreign material under flares.
- Bad fitting or mismated parts.
- Careless assembly.
- Threads seized or galled.

(2) Causes of leaks at threaded joints fitted with synthetic rubber type gasket:

- Improper positioning of gasket on fitting.
- Fitting on boss not properly positioned.
- Not enough torque to squeeze gasket and make seal.
- Damage to surface contacted by O-rings.
- Careless assembly.

(3) Tubing Failures.

Tubing failures generally are the result of improper bends, nicks, dents or because the tubing is too short. The tubing used in the hydraulic systems is selected to withstand several times the operating pressure to which it will be subjected. Vibration resulting from chattering or insufficient support is also a common cause of failure.

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HYDRAULIC POWER SYSTEMS — MAINTENANCE PRACTICES

1. **General**

Take every precaution to ensure that the fluid in the system remains clean. Micronic-type filters are installed in the system to remove fine particles of metal which result from natural wear of parts. Only hydraulic units which have been thoroughly cleaned and protected by capping of the openings immediately after cleaning should be installed. Take care to see that all threads mate easily without causing metal particles to enter the system. Do not twist hoses when installing.

The importance of keeping water from entering the system cannot be overemphasized. In low temperature operation, very small quantities of water can cause serious malfunction or stoppage in units having small orifices.

Use only water-free hydraulic fluid, when adding to or refilling the system. Add fluid only through the filler assembly on the reservoir.

Water in hydraulic fluid can be detected visually. Pure hydraulic fluid is clear, water will cause fluid to cloud. Drain system and refill with clean fluid if there is any evidence of cloudiness. If purity of the fluid is in doubt, compare a sample from the aircraft supply with fluid known to be water-free. Samples should be held in test tubes, but any clear glass containers may be used. Temperature within the operating range of the aircraft will not affect the validity of the clarity test.

NOTE: Air in solution can also cloud hydraulic fluid.

Hydraulic units which are functioning properly need not be removed for inspection. Units which are heating (denoting internal leakage), or actuating cylinders which are abnormally slow in action must be replaced or repaired. Internal leakage at actuating cylinders usually is due to leakage past O-ring seal, which may have been cut by metal particles, or which are worn by use. In these cases, replacement of the O-rings is all that will be required. Scuffing of the O-rings due to a scored cylinder or excessive clearance between piston and cylinder usually is reason for replacement of the cylinder. O-rings should be installed carefully in order not to damage them. Coat the O-ring and any surface over which it must slide with aircraft hydraulic fluid before installation. Be careful not to cut O-rings on sharp edges and when passing them over threads, cover the threads with a metal sleeve. Use only proper size O-rings.

WARNING: IF HYDRAULIC UNIT HAS AN ELECTRICAL CONNECTION, ENSURE ALL AIRCRAFT ELECTRICAL POWER IS OFF.

2. **Removal / Installation — Procedures and Precautions**

WARNING: THE GAS MUST BE RELEASED FROM ALL UNITS WHICH ARE CHARGED WITH NITROGEN BEFORE ANY DISASSEMBLY IS ATTEMPTED. COMPRESSED NITROGEN IS VERY DANGEROUS AND SHOULD BE HANDLED WITH EXTREME CAUTION.

NOTE: No special instructions pertaining to removal of pneumatic and hydraulic components are required. Relieve hydraulic pressure by operating a high-displacement hydraulically-operated component, preferably the brakes. release any residual pressure by loosening filler caps from reservoir. At removal, tag lines and units that they can be properly identified and installed. After installation of hydraulic units and replacement of filter elements, operate the hydraulic system to which installed units are connected. An external hydraulic test rig should be used, to return entrapped air to reservoir, to detect leaks and to ensure proper operation.

3. **Pre-Installation Procedures**

WARNING: USED SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY AWAY FROM FLAMES.

A. Clean all parts thoroughly with dry cleaning solvent.

B. Coat all unplated surfaces which have been cleaned with VV-P-236, Skydrol or recommended alternates.

C. Tag all line ends and fittings prior to removal of one or more hydraulic components. If line is removed from aircraft, tag all clips and clamps with end locations; e.g., filter outlet to check valve line.

D. Old fittings which are available and in good condition may be cleaned and reused. Clean new fittings thoroughly before they are installed.

CAUTION: DO NOT USE OIL OR GREASE TO INSTALL SEALS AND PACKINGS IN HYDRAULIC SYSTEMS.

E. Coat all seals, O-rings and packings with Monsanto Company MCS-352 lubricant or Gulfstream Aerospace approved and recommended hydraulic fluid before assembly. Felt or other porous packings should be soaked in Gulfstream approved and recommended hydraulic fluid. Coat threaded on other rough surfaces over which seals, O-rings or gaskets must pass using the same type of lubricant. Use care in assembly to prevent damage or distortion to soft parts. Use MCS-352 lubricant sparingly on male thread areas of fitting and fasteners during assembly.

4. **Tubing codes**

Dash No.	Wall Thickness In Inches	OD In Inches	Material	Service
-4	0.028	1/4	Type 321 corrosion resistant steel	Wheel brakes only.
-4	0.035	1/4	52S0 aluminum alloy	All except wheel brakes.
-6	0.049	3/8	52S0 aluminum alloy	All.
-8	0.065	1/2	52S0 aluminum alloy	All.
-12	0.095	3/4	52S0 aluminum alloy	All except return and suction.
-12	0.049	3/4	52S0 aluminum alloy	Return and suction.
-16	0.049	1	52S0 aluminum alloy	Return and suction.
-20	0.049	1-1/4	52S0 aluminum alloy	Return and suction.

5. **Torques Values For 18-8 Corrosion-Resistant Steel and 52S0 Aluminum Alloy Tubing**

Dash No.	OD in Inches	Torque in Inch-Pounds
-4 stainless	1/4	125-150
-4 aluminum / stainless	1/4	40-65
-6	3/8	75-125
-8	1/2	150-250
-12	3/4	300-500
-16	1	500-700
-20	1-1/4	600-900

6. **Hydraulic Fluid — Contamination Check**

NOTE: Aircraft hydraulic fluid should be kept as clean as possible to ensure long life of components. Caution should be used whenever servicing the reservoirs or connecting a ground servicing cart not to introduce any contaminants. When replacing components, any open hydraulic lines should be immediately capped with protective plugs.

A. Clean, used hydraulic fluid will have a color close to the original color. If the fluid becomes dark brown in color, then it is recommended that the system be flushed and refilled with new fluid.

B. The fluid should be checked visually for solid particle contaminants.

C. The following criteria is used by hydraulic lab quality control at Gulfstream Aerospace to determine acceptable levels of contamination. A 100 ml sample is taken from hydraulic reservoir and passed through a 3.0 micron filter. The filter is then analyzed under a Millipore Binocular Microscope.

PARTICLE SIZE RANGE (MICRONS)	ALLOWABLE NUMBER (MAX. EACH SAMPLE)
5-25	3500
26-50	250
51-100	25
Over 100	0 (None)

7. **Hydraulic Power System — Operational Test**

NOTE: Check hydraulic fluid level with landing gear down and locked, main entrance door open and brake accumulator charged.

Fill reservoir to GROUND FULL level on sight gage, if needed, with approved type fluid. (see Section 12-1-6)

A. Normal System

(1) Start right engine; observe normal hydraulic pressure gage immediately indicates 1400 to 1500 psi.

(2) Cycle flaps; observe hydraulic pressure builds up to 1400 to 1500 psi after flaps are raised.

(3) Start left engine; when engine has stabilized, shut right engine down, cycle flaps; observe hydraulic pressure builds up again to 1400 to 1500 psi after flaps are raised.

(4) Shut down engine.

(5) Check hydraulic power system lines and components for leaks.

(6) Service reservoir, if needed, with approved type fluid. (see Section 12-1-6)

B. Auxiliary System

(1) Apply electrical power to aircraft.

WARNING: ENSURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR BEFORE OPERATING FLAPS AND DOORS.

(2) Using emergency hydraulic pump switch, operate following systems one at a time. Observe hydraulic pressure builds up to 1400 to 1500 psi but does not exceed 1900 psi.

- Airstair door.
- Emergency flaps.
- Landing gear door system.
- Wheel brakes.

NOTE: Pump capacity is 0.5 gpm. Systems will operate at slower rate than operated in normal system operation (flaps, landing gear doors, and brake pressure).

(3) Ensure each system functions smoothly and pump maintains continuous operation of system.

(4) After operation of emergency flaps, reset emergency flaps, reset emergency flap handle to neutral and reset emergency dump valve on Fuselage Station 169 bulkhead.

(5) Check hydraulic lines and components in auxiliary power systems for leaks.

(6) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

MAIN HYDRAULIC POWER SYSTEM — DESCRIPTION / OPERATION

1. General

The Main Hydraulic Power System components are located as follow:

UNIT	NO PER A/C	LOCATION
Hydraulic Reservoir	1	Aft side of Bulkhead, Fuselage Station 169.
Shutoff Valves (for suction lines)	2	Beneath removable section of cabin floor forward of the wing, one on left and one on right side of aircraft. FLR-6 and -9.
Vent and Relief Valve	1	Above reservoir.
Ground Test Connections.	2	Right engine nacelle below accessory compartment top panel.
Engine Driven Pumps	2	Each engine nacelle below accessory compartment top panel.
Main Filters	2	Each engine nacelle (left side) in proximity of gearbox housing.
Engine Pumps Bypass Line Filter	1	On hydraulic reservoir bulkhead.
Check Valves (in pressure line)	2	Beneath removable section of cabin FLR-9 forward of wing and forward of left shutoff valve.
Pressure Relief Valve	1	Near hydraulic reservoir.
Hydraulic Fuse (gage) (Aircraft 1-90 and 114, only)	1	Access panel on right side of fuselage, outboard of nose gear.
Pressure Snubber (gage)	1	Access panel on right side of fuselage, outboard of nose gear.
Hydraulic Pressure Gage	1	Copilots right skirt panel, lower right corner.
Auxiliary to Main Selector Valve (Aircraft having ASC 109).	1	Aft left hand corner of nose wheel well.

The main hydraulic power system develops, maintains and supplies 1500 psi hydraulic pressure. The hydraulic reservoir is divided into two sections: The main system chamber and the auxiliary system chamber. Hydraulic fluid flows from the main system chamber through two separate suction lines. One line is for the right engine driven pump and the other is for the left engine-driven pump. Each pump suction line has its own electrically operated shutoff valve installed between the reservoir and the related engine driven pump. A single suction line runs from the auxiliary chamber of the hydraulic reservoir to the electrically driven auxiliary hydraulic pump. (See Auxiliary Hydraulic Power System). Either pump is capable of supplying main system pressure and volume requirements. Hydraulic fluid, maintained at the pressure required to operate the main systems, leaves the two pumps and is directed through filters and check valves to one common supply line. The fluid under pressure passes, in turn, from the common supply line into the main pressure relief valve, the hydraulic fuse (Aircraft 1-90 and 114, only), the pressure snubber and the main hydraulic system pressure gage. The fluid then operates the main hydraulic power system. A vent and relief valve is included to maintain sea level atmospheric pressure and relieve excessive pressures in the reservoir. Two drain ports are provided for draining the reservoir. The main section is drained at the reservoir; the auxiliary section is drained at a tee fitting under the auxiliary hydraulic pump in the nose wheel well.

Two external ground test connections are installed in the right engine hydraulic pump to provide a means of hookup to an external hydraulic test rig for ground operation and tests.

2. Engine-Driven Pump

The two main hydraulic system engine-driven pump develop and maintain 1500 psi hydraulic fluid pressure for the subsystems of the main hydraulic power system. The pump are mounted on the accessory gear box in each nacelle and are accessible through the top cowling. Pump volume varies from 0 to 9.5 gpm with full flow from 0 to 1275 psi. The compensator unit in the pump starts to cut off the flow at 1275 psi and at 1500 psi, no hydraulic fluid flows. The pump has three ports: intake, discharge and bypass. A filter is incorporated in the return line into which the bypass lines from the pump are joined.

Each pump consists of three major functions enclosed by the housing and cover assemblies. The major functions are the mechanical-drive mechanism, fluid displacement and pressure control units.

A. Mechanical-Drive Mechanism

Piston motion is caused by the drive cam displacing each piston the full cam height per revolution. The drive cam also provides a means to positively return the pistons. By coupling the ring of pistons with a nutating (wobble) plate supported by a fixed center pivot, the pistons are held in constant contact with the cam face. As the drive cam depresses one side of the nutating plate, as pistons are advanced, the other side of the nutating plate is withdrawn an equal amount thereby moving the pistons with it. Creep plates are provided to decrease wear by lessening the energy absorbed in the high speed revolving cam and its supporting bearing surface. These creep plates are free to turn at some intermediate speed between the cam and the stationary bearing. Energy absorbed by each surface is reduced, as absorbed energy in a bearing is a direct function of relative surface velocity.

B. Fluid Displacement

Fluid is displaced by axial piston motion. As the piston advances in its respective cylinder block bore, pressure opens the check and a quantity of fluid is forced past. Combined back pressure and check spring pressure closes the check when piston bypass ports align, hydraulically, with the cylinder block bypass passage. The partial vacuum occurring in the cylinder during the piston return, causes fluid flow from the intake into the cylinder either under atmospheric pressure or from a pressurized reservoir.

C. Fluid Flow

When maximum flow is not required in the system, a pressure build up occurs in the discharge. Discharge pressure bleeds through a control orifice into the pressure compensator cylinder and moves the compensator piston stem against the force of a calibrated control spring. This motion is transmitted directly to piston sleeves which are free to move axially along the pistons. Sleeve position determines the effective piston stroke. When the sleeves are in their original position as in full flow, the relief holes enter the sleeve as soon as the piston starts its pressure stroke. When the sleeves are fully advanced as in no flow, the relief holes do not enter the sleeves until the piston has almost completed its pressure stroke. As long as the relief holes are uncovered, fluid escapes out of the piston into the pump body, decreasing the effective pumping stroke by the amount of fluid that escapes. When no flow is required by the system, only enough fluid is pumped to maintain the system pressure against leakage.

D. Pressure Control

The three conditions of fluid flow required in operation, full flow, partial flow and no flow, are selected by system pressure. Under all flow conditions, fluid enters the intake port and is discharged to the high pressure side through the pump by the reciprocating action of the pistons. A pressure tap (orifice) connects the discharge, or high pressure side, with the pressure compensator piston. Under full flow requirements, the discharge pressure is insufficient to move the compensator piston against the force of the compensator spring. As flow requirements are reduced, pressure is built up in the discharge sufficient to move the compensator piston stem. As the compensator piston moves, the piston sleeves uncover the relief holes, thereby reducing the effective stroke. If pressure continues to build up as under zero or no flow conditions, the relief holes are uncovered for practically the entire piston stroke. They will be covered only for the time necessary to produce adequate bypass flow and to maintain system pressure against leakage.

E. Bypass

The bypass system is provided to supply self-lubrication during full flow and no flow operation. The ring of bypass holes in the piston is aligned, hydraulically, with the bypass passage at the very end of the

pistons forward travel, under all conditions of flow. A small quantity of fluid is pumped out the bypass passage and back to the supply reservoir, thereby constantly changing the fluid in the pump. The bypass is designed to pump against a considerable back pressure for use with pressurized reservoirs. This feature is not required in the Gulfstream I installation.

3. Shutoff and Check Valves

Two electrically operated, gate type, shutoff valves, which are normally open, are located beneath cabin floor boards no. 6 and 9 forward of the wing. If a hydraulic leak or fire is discovered, the shutoff valves can be actuated by pulling the T-handle, placarded FIRE PULL. These T-handles are located at the center of the eyebrow panel. The shutoff system provides a method of controlling leakage in the nacelle region. Approved hydraulic fluids are relatively nonflammable, but if injected into a hot fire will burn. It is therefore desirable to shut off this fluid in case of fire. (See Figure 1)

Each of the valves shut off the hydraulic fluid through the suction line to the engine driven pump. Each pressure line is equipped with a check valve, located beneath the removable section of the cabin floor forward of the left shutoff valve. These check valves permit flow to the system and prevent return flow. An arrow on the body of the check valve indicates direction of free flow. Thus, closing either of shutoff valve hydraulically isolates the related nacelle.

When one of the T-handles is pulled, a limit switch is actuated, which energizes the close windings of the split field dc motor in the shutoff valve. Power is supplied by the essential dc bus through the HYD S/O 5 ampere circuit breaker, located on the pilots circuit breaker panel.

4. Filters

Hydraulic fluid leaves the two pumps under pressure and is routed to two filters. Each filter contains a replaceable micronic filter element to prevent contamination of the systems by dirt and foreign objects. A 50 psi differential pressure bypass valve also is installed. Each filter is located in the vicinity of the gearbox housing on each engine nacelle, near the pump. Normally the fluid under pressure flows through the filter element and out to the systems. However, if the element becomes so dirty that the fluid meets resistance, the fluid pressure forces the relief valve off its seat against spring pressure. The hydraulic fluid then bypasses the filter element and continues on through the system without being filtered. (See Section 29-0 Figure 1)

5. Pressure Relief Valve

The pressure relief valve is located near the reservoir and is installed to protect the main hydraulic system against excessive pressures. It is rated for a maximum full flow pressure of 16 gpm at 1900 psi. The minimum reset pressure for the relief valve is 1517 psi. Even when it is opening to relieve fluid under excessive pressures, the relief valve does not prevent fluid pressure from operating the systems. Fluid under excessive pressure enters the valve at the pressure port (marked INLET) and leaves the valve at the return port (marked OUTLET) to return to the reservoir. (See Section 29-0 Figure 1 and Figure 2

6. Ground Test Connections

Two quick disconnect ground test connections, equipped with dust caps, are located on the right hydraulic pump behind the access panel. Valves, incorporated in the connections, open when the ground test connections are connected, and close when they are disconnected. These connections permit attachment of an external hydraulic test rig to operate the subsystems of the main hydraulic power system for ground operation and tests. During such operations, the right and left engine-driven hydraulic pumps are bypassed. A 1500 psi hydraulic test rig, filled only with hydraulic fluid, identical to that used in the aircraft hydraulic system, should be used in testing. (See Section 29-0 Figure 1)

Fluid under pressure leaves the pressure connection of the rig and flows into the right engine-driven hydraulic pressure outlet. It flows directly to the filter and then to the subsystems of the main hydraulic power system. Fluid returns to the reservoir and flows out through the shutoff valve to a tee connection on the right engine hydraulic pump suction inlet, bypassing the pump and returning to the rig.

7. **Indicating System Pressure Gage**

The main hydraulic system pressure gage is installed in the pressure line after the pressure snubber. The indicator is located on the copilots outboard skirt panel. The dial is calibrated from 0 to 2000 psi. The pressure gage records hydraulic pressure of the main hydraulic system as maintained by the engine-driven pumps. The gage is protected against pressure surges by the pressure snubber and against excessive loss of hydraulic fluid, through a line break, by the hydraulic fuse (Aircraft 1-90 and 114, only).

NOTE: The output of either pump alone can maintain system pressure and flow requirements. Since the gage measures system pressure and not the pressure of individual pumps, the only check on individual pump outlet is on engine starting and shut down.

8. **Indicating System Pressure Snubber**

The pressure snubber is located in the pressure line which runs to the main hydraulic pressure gage. It protects the hydraulic pressure gage by minimizing the oscillations of the gage indicator caused by pressure surges. The pressure snubber contains two ports: an inlet port for fluid from the automatic resetting hydraulic fuse (Aircraft 1-90 and 114, only) and an outlet to the gage. (See Section 29-0 Figure 1)

9. **Indicating System Automatic Resetting Hydraulic Fuse**

NOTE: The hydraulic fuse applies only to Aircraft 1-90 and 114. On Aircraft 91-200, 322 and 323, the hydraulic fuses have been eliminated from the system.

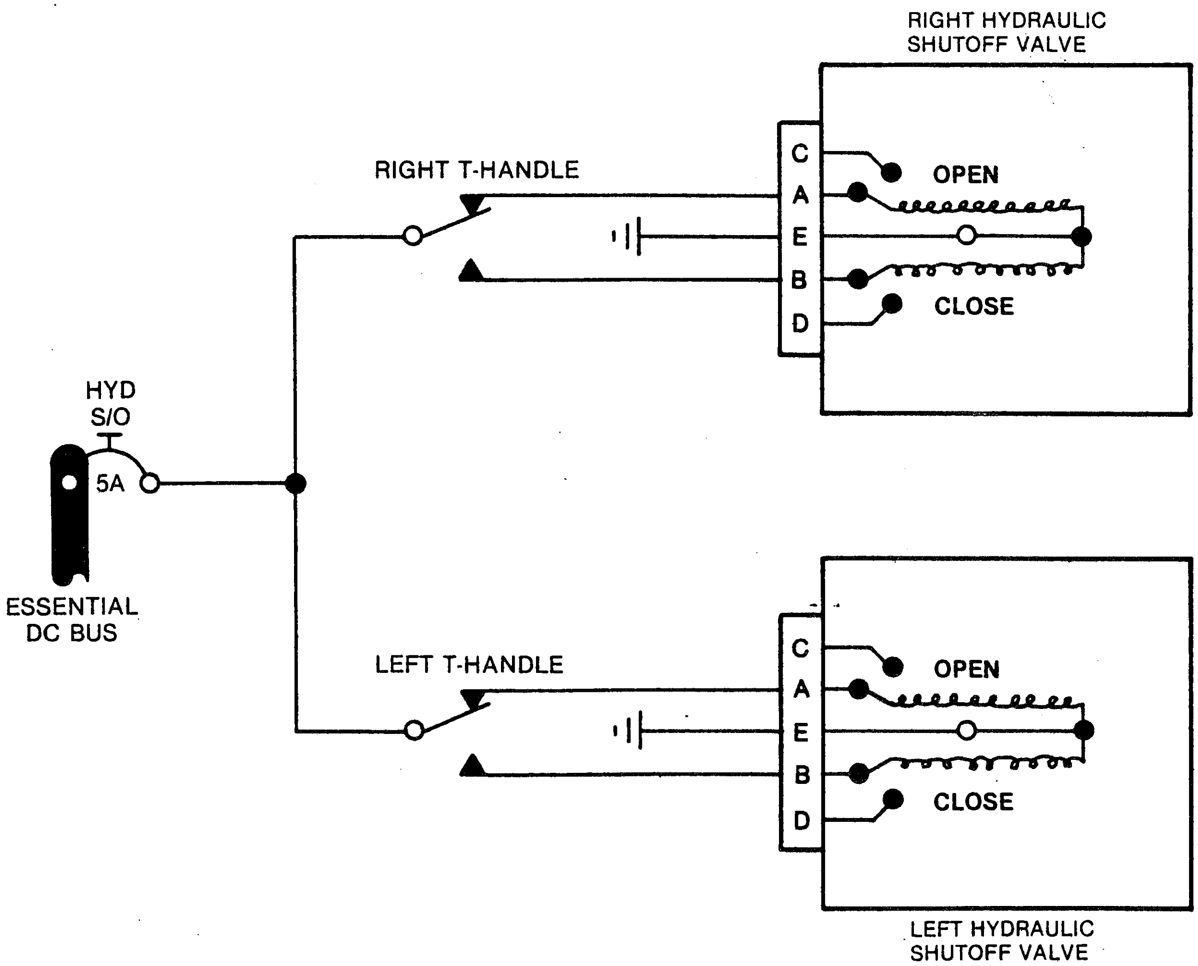
This hydraulic fuse is installed in the line which runs to the main hydraulic pressure gage; it is accessible behind the access panel on the right side of the fuselage outboard of the nose gear. Its function is to minimize fluid loss if the pressure gage or the line to the gage were to rupture. The hydraulic fuse contains two ports; an inlet port for system pressure and an outlet to the pressure snubber. The automatic resetting hydraulic fuse is designed to pass 1.8 cubic inches of fluid. If a failure occurs downstream of the fuse, only 1.8 cubic inches of fluid is lost, as a poppet valve will seat and be held shut by system pressure.

10. **Internal Ground Service Pressure Provisions**

Described in A. and B. below are two means whereby the main hydraulic system can be pressurized by operating the auxiliary pump in lieu of the main engine-driven hydraulic pumps or a hydraulic rig.

A. Aircraft not having ASC 109 are provided with jumper connections which allow the main hydraulic system to be pressurized utilizing the auxiliary hydraulic pump. These jumper connections are located in the nose wheel well, right side, just aft of the nose strut. Two capped T-fittings are provided, one in each hydraulic gage line. Instructions for use of these fittings are contained under Maintenance Practices in this section.

B. On Aircraft having ASC 109, a selector valve has been installed in lieu of the capped T-fittings. This selector valve has two positions; AUX TO MAIN "ON" and FLIGHT POS. The valve is spring loaded in the FLIGHT POS. position. It must be secured in the AUX TO MAIN "ON" position using ground safety lock P/N 159GT1007. Instructions for use of this selector valve are contained under Maintenance Practices in this section.



Main Hydraulic Shutoff — Schematic
Figure 1.

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MAIN HYDRAULIC POWER SYSTEM – FAULT ISOLATION

FAULT	POSSIBLE CAUSE	CORRECTION
Main system pressure gage fluctuates or indicates less than full system pressure.	Low fluid level or no fluid in reservoir.	Check entire hydraulic system for leaks and refill reservoir.
	Pump malfunctioning	Replace pump.
	Air leak or blocked pump suction lines.	Check tubing. Check pressure gage for excessive pressure drop when engines are idling. Check emergency shutoff valve to ensure that it is open.
	Defective pressure indicating system.	Replace defective unit.
Excessive heating of hydraulic lines or units.	Excessive internal leakage. within units.	Trace leakage to hottest part of line. Allow line to cool, then make test again.
Engine pump fails to deliver normal pressure.	Pump cavitating due to insufficient fluid in main reservoir	Fill reservoir
	Incorrect pressure indication.	Check pressure indicating system.
	Main relief valve sticking open.	Replace relief valve.
	Leak in system.	Block off system beyond relief valve and check pump pressure. If normal, find leak and repair.
	Pump cavitating due to failure to fill pump case with fluid before installation.	Remove pump and fill with fluid. Replace pump if necessary.
	Misadjusted or defective pump.	Replace pump.
Pump fails to deliver fluid.	Insufficient fluid in reservoir.	Fill reservoir.
	Air leak or obstruction in suction line.	Remove and check suction line.
	Check valves improperly installed	Check for proper installation
	Pump seizure because of contaminated fluid.	Flush system, replace filter elements and pump.
	Emergency shutoff valves closed.	Open shutoff valves.
	Pump seizure from any of the above causes resulting in sheared drive coupling.	Check above remedies and replace pump
Pump delivers excessive pressure	Incorrect pressure indication pressure.	Check pressure indicating system
	Misadjusted or defective pump.	Replace pump.
Pump overheating.	Relief valve sticking open.	Replace relief valve.
	Pump compensator sticking allowing pump to operate at full volume against excessive pressure.	Replace pump.

FAULT	POSSIBLE CAUSE	CORRECTION
External Leakage at pump.	Shaft fluid seal damaged.	Replace pump.
Air in main hydraulic system.	Suction line leaking or not properly connected.	Check suction line.
	Small air leak at pump intake fitting.	Replace pump.
Chatter in system.	Air in system.	Operate system several times to return trapped air in reservoir. Fill reservoir.

MAIN HYDRAULIC POWER SYSTEM — MAINTENANCE PRACTICES

1. **Ground Service Pressure Provisions**

A. External

Self sealing quick disconnects are provided for connecting an external hydraulic power source. The disconnects are located at the right engine hydraulic pump.

B. Internal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

CAUTION: ENSURE THAT MAIN HYDRAULIC SYSTEM IS NOT IN OPERATION WHEN USING JUMPER CONNECTIONS.

(1) Jumper connections are provided so that the main hydraulic system can be pressurized with the auxiliary hydraulic pump. These connections are located in the nose wheel well, right side, just aft of the nose strut. Two capped T-fittings are provided, one in each hydraulic gage line. To use the T-fittings proceed as follows:

(a) Uncap fittings.

(b) Attach No. 4 hydraulic flex line at least 15 inches long.

(c) Energize auxiliary hydraulic pump. Auxiliary hydraulic pump will pressurize both auxiliary and main hydraulic systems.

WARNING: JUMPER MUST BE REMOVED BEFORE FLIGHT.

(2) For aircraft having ASC 109, use of the selector valve is as follows:

CAUTION: ENSURE THAT MAIN HYDRAULIC SYSTEM IS NOT IN OPERATION WHEN SELECTOR VALVE IS IN THE AUX TO MAIN ON POSITION.

(a) Rotate selector valve handle to AUX TO MAIN "ON" position.

(b) Secure in position with ground safety lock Part No. 159GT1007.

(c) Energize auxiliary hydraulic pump to pressurize both auxiliary and main hydraulic systems.

WARNING: VALVE MUST BE RETURNED TO FLIGHT POSITION BEFORE FLIGHT.

2. **Engine-Driven Hydraulic Pump — Removal / Installation**

A. Removal

(1) Operate brakes to deplete hydraulic system pressure. Slowly loosen reservoir emergency filler cap to deplete remaining pressure and pull appropriate engine FIRE PULL T-handle to close hydraulic shutoff valve.

(2) Gain access to hydraulic pump located in engine nacelle, below accessory compartment top panel.

(3) Disconnect hydraulic lines from pump. Cap and plug lines and ports.

(4) Remove nuts and washers that secure pump to gearbox and remove pump.

(5) Disconnect fittings in hydraulic pump ports and discard seals. See Figure 201 for left pump and Figure 202 for right pump for fittings and O-rings.

(6) Remove and discard pump shaft O-ring if New York Air Brake 109A is installed.

B. Installation

NOTE: When installing pump, ensure to fill the case with clean hydraulic fluid identical to that used in the aircraft hydraulic system. (See Chapter 12 for approved fluids)

Before installation, ensure flex hydraulic pressure lines are not kinked, torque stressed or chafing.

Lubricant not required if New York Air Brake 109A is installed.

(1) Ensure shaft is secured by split ring retainer.

(2) Apply light coat of Aeroshell 5, Andok 260 or Marfak 3 HD grease to splines of shaft.

(3) Install fittings and new O-rings in hydraulic pump ports. See Figure 201 for left pump and Figure 202 for right pump for fittings and O-rings.

(4) Install O-ring on pump shaft if New York Air Brake 109A is installed. Lubricate with approved gearbox oil.

(5) Install new gasket on gearbox pad.

(6) Install pump on gearbox and secure with nuts and washers. Torque nuts to 160 inch pounds.

(7) Remove plugs from lines and connect lines to pump.

(8) Inspect and clean main hydraulic pressure filter, see this Section.

(9) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(10) Reset engine FIRE PULL T-handle.

CAUTION: IF GAGE FAILS TO INDICATE NORMAL PRESSURE IMMEDIATELY, SHUT DOWN ENGINE AND INVESTIGATE.

(11) Start engine and check that normal system pressure is obtained.

(12) Check for leaks.

(13) Shut down engine.

(14) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(15) Inspect area for foreign objects.

(16) Replace access panel.

3. **Hydraulic Pump Coupling — Inspection**

A. Remove hydraulic pump, this Section.

B. Remove splined coupling by carefully snapping out split retainer using screw driver, and slide splined coupling out of pump.

C. Clean splined coupling and face of pump with P-D-680 solvent.

D. Inspect splined coupling for wear.

E. If wear appears normal, proceed to Step G.

F. If wear appears excessive, proceed as follows: (See Figure 204)

(1) Using micrometer, measure over three teeth of 16 tooth spine (large end that fits into pump). Measure around complete spline, moving micrometer one tooth for each check (16 places). Minimum distance is 0.358 inch over three teeth.

(2) Using micrometer, measure over three teeth of 12 tooth spline (small end that fits into gearbox). Measure around complete spline, moving micrometer one tooth for each check (12 places). Minimum distance is 0.348 inch over three teeth.

(3) If either end of spline coupling fails to meet tolerances specified, replace hydraulic pump shaft:

(a) On hydraulic pumps not having New York Air Brake Modification 109A, replace with shaft (P/N 2349 or 2476).

(b) On hydraulic pumps having New York Air Brake Modification 109A, replace with shaft (P/N 6406). The use of this coupling also requires an O-ring.

(4) If gearbox end of splined coupling fails to meet tolerances, clean and inspect gearbox internal spline for wear. If worn, replace gearbox.

G. Apply a light coat of PlastiLube #3 moly grease on splines of pump end of coupling and install to pump. Secure with split retainer.

H. Install hydraulic pump, this Section.

4. Hydraulic Pump Shaft — Lubrication

NOTE: Lubrication is not applicable to Aircraft having Dowty Rotol Modification GB 2483 and NY Mod. 109A (continuous/wet spline lubrication).

See Figure 203

NOTE: Clean spline prior to lubrication and wipe spline clean of excess grease upon completion.

A. Remove hydraulic pump, see this Section.

B. Visually inspect coupling splines for evidence of wear and corrosion.

NOTE: If wear appears excessive, see Hydraulic Pump Coupling - Inspection, this Section.

C. Apply a light coat of grease to splines on gearbox end of pump coupling.

D. Install hydraulic pump, see this Section.

5. Hydraulic Pressure Flex Hose — Inspection / Maintenance

A. Gain access to top of accessory section and inboard side of right engine nacelle.

B. Inspect area forward of hydraulic filter where hose passes through frame for interference, hose to frame. If interference is found at frame forward of filter, contact Gulfstream Aerospace Corporation Technical Operations for assistance.

C. Inspect area forward of hydraulic filter where hose passes under engine truss for chafing. If interference is found at truss gain clearance by relocation of clamps.

D. Inspect area forward of hydraulic pump where hose passes over accessory gearbox drip tray (left and right installation).

NOTE: Flex hose should not touch any object between clamps. Flex hose should be free of twisting on installation.

If flex hose is damaged, replace. If chafing exists at drip tray, notch for clearance. (See Figure 205)

E. As an additional precaution it is recommended that the edges of the structure at the interference points be coated with silastic silicone rubber sealant (Dow Corning P/N 94-002 sealant and RTV-1200 primer) using the following procedure:

(1) Lightly abrade the surfaces to be coated to remove surface gloss.

(2) Clean surfaces using a clean cloth saturated with MEK. Thoroughly clean surface to be coated. Wipe off solvent film immediately with a clean dry cloth. Do not allow the solvent to dry on the surface. Perform solvent cleaning a minimum of two times before coating.

(3) Apply RTV-1200 primer using a brush or swab in a uniform film, as thin as possible, and allow to dry for 30 minutes.

(4) Apply sealant in the form of a bead directly from the tube or with a spatula. Allow to cure for 24 hours.

F. Inspect area for presents of foreign objects, security of all attachments and install access panels.

6. Hydraulic Pump Shutoff Valve — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic system pressure.

(2) Pull HYD S/O circuit breaker. (Pilots circuit breaker panel.)

(3) Drain hydraulic reservoir using one of the following methods.

(a) Using clean container, drain hydraulic fluid from reservoir at reservoir drain and at auxiliary hydraulic pump drain in nose wheel well.

(b) Connect hydraulic power unit suction line to service connection in right nacelle and ensure FIRE PULL T-handles are in. Open hydraulic power unit tank valve and allow fluid to gravity feed from reservoir to power unit.

(4) Gain access to valve by removing cabin FLR-6 for right side or FLR-9 for left side.

(5) Disconnect electrical connector from valve.

(6) Disconnect hydraulic lines from valve; cap or plug lines and ports.

(7) Remove valve.

B. Installation

(1) Remove caps and plugs from lines and ports.

(2) Install valve and connect line only to reservoir side of valve.

(3) Connect electrical connector to valve.

(4) Connect hydraulic power to aircraft.

(5) Depress HYD S/O circuit breaker.

CAUTION: DO NOT TOUCH FIRE EXTINGUISHER SWITCH WHICH IS EXPOSED WHEN T-HANDLE IS PULLED.

(6) Pull applicable FIRE PULL T-handle.

(7) Place container under open port of valve.

(8) Service reservoir with approved fluid. Tighten reservoir cap(s). (See Section 12-1-6)

(9) There should be no evidence of fluid leakage from open port of valve indicating valve is properly closed.

(10) Connect remaining hydraulic line to valve; remove container.

(11) Push in FIRE PULL T-handle. Check valve indicator flag for OPEN position.

CAUTION: IF NORMAL PRESSURE IS NOT INDICATED IMMEDIATELY, SHUT DOWN ENGINE AND CHECK THAT SHUTOFF VALVE IS OPEN.

(12) Start engine and ensure that normal system pressure is obtained immediately.

(13) Shut down engine.

(14) Inspect area for foreign objects.

(15) Install removed FLR-6 or -9.

(16) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

7. Main Hydraulic Pressure Filter Element — Removal / Installation

A. Removal

(1) Operate brakes to deplete residual pressure in hydraulic system.

(2) Slowly loosen emergency filler cap on reservoir to deplete remaining pressure.

(3) Pull engine FIRE PULL T-handle to close hydraulic shutoff valve.

(4) Remove accessory access panel on nacelle of filter to be inspected.

(5) Remove safety-wire at filter bowl; unscrew filter bowl from housing.

(6) Remove filter element.

CAUTION: IF METAL PARTICLES ARE FOUND IN FILTER ELEMENT OR FILTER HOUSING, INVESTIGATE FURTHER FOR POSSIBILITY OF SYSTEM CONTAMINATION CAUSED BY COMPONENT FAILURE.

B. Installation

(1) Discard old seals from bowl and element.

(2) Install new element O-ring in filter housing and install filter element.

(3) Install new O-ring and backup ring on filter bowl and fill filter bowl with clean, approved hydraulic fluid.

(4) Install filter bowl, torque to 150 inch pounds and safety-wire.

(5) Push in engine FIRE PULL T-handle.

(6) Check hydraulic reservoir fluid level.

(7) Start applicable engine and check for normal system pressure indication of 1400 to 1500 psi.

(8) Inspect for hydraulic leaks

(9) Shut down engine.

(10) Inspect for foreign objects.

(11) Install accessory access panel.

(12) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

8. Hydraulic Return Filter Element — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic system pressure.

(2) Slowly loosen emergency filler cap on reservoir and deplete remaining pressure.

(3) Locate filter on Fuselage Station 169 bulkhead.

(4) Remove lockwire and unscrew filter bowl from housing.

(5) Remove filter element, discard O-rings.

CAUTION: IF THERE IS ANY EVIDENCE OF METAL PARTICLES IN FILTER ELEMENT OR FILTER HOUSING, INVESTIGATE FURTHER FOR POSSIBILITY OF SYSTEM CONTAMINATION CAUSED BY A COMPONENT FAILURE.

(6) Inspect element for foreign matter.

B. Installation

(1) Install new O-ring in filter housing and install filter element.

(2) Install new O-ring and backup ring on filter bowl and fill filter bowl with clean, approved hydraulic fluid. bowl and safety-wire.

(3) Start engine and check for 1400 to 1500 psi hydraulic pressure.

(4) Check for leakage.

(5) Shut down engine.

(6) Inspect area for presents of foreign objects, security of all attachments.

(7) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

9. Main System Relief Valve — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic system pressure.

(2) Slowly open filler cap at top of reservoir to deplete remaining pressure.

(3) Locate main relief valve inboard of reservoir. See Section 29-0, Figure 1 and Figure 2).

(4) Disconnect lines from valve and remove valve from aircraft.

(5) Cap and plug all lines and ports.

B. Installation

(1) Remove caps and plugs from lines and ports.

(2) Install valve, connect lines to valve and tighten.

(3) Tighten valve cap on reservoir.

(4) Connect hydraulic power to aircraft and pressurize hydraulic system.

(5) Observe that valve maintains system pressure.

(6) Inspect valve for leaks.

(7) Inspect area for foreign objects and security of all attachments.

(8) Service reservoir, if needed, with approved type hydraulic fluid. (See Section 12-0)

10. Engine Pressure Line Check Valve — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic pressure.

(2) Slowly open filter cap at top of reservoir to deplete reservoir pressure.

(3) Pull engine FIRE PULL T-Handle to close hydraulic shutoff valve.

(4) Remove FLR-9, left forward side of cabin.

(5) Disconnect hydraulic lines and remove check valve.

(6) Cap or plug all lines and ports.

B. Installation

(1) Remove caps and plugs from lines and ports.

(2) Install valve, with FLOW arrow pointing away from pump, connect lines to valve and tighten.

(3) Push in FIRE PULL T-handle.

(4) Start appropriate engine; observe engine pump gage indicates normal system pressure.

(5) Bleed gage line if pressure fluctuates or is below normal system pressure.

(6) Check for hydraulic fluid leakage.

(7) Shut down engine.

(8) Inspect area for presents of foreign objects, security of all attachments.

(9) Install FLR-9.

(10) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

11. Normal Hydraulic Pressure Gage — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic system pressure.

(2) Slowly open filler cap at top of reservoir to deplete remaining pressure.

(3) Disconnect lines from gage, located on copilots skirt panel and remove gage.

(4) Cap and plug hydraulic line and gage port.

B. Installation

(1) Remove cap and plug from hydraulic line and gage port.

(2) Install gage, connect and tighten line.

(3) Apply hydraulic power to aircraft and check for 1400 to 1500 psi, normal hydraulic pressure.

(4) Check for leaks.

(5) Inspect area for foreign objects and security of all attachments.

(6) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

12. Main Pressure Gage Snubber — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic pressure.

(2) Slowly open filler cap at top of reservoir to deplete pressure.

(3) Open access panel on right side of fuselage, outboard of nose gear.

NOTE: On Aircraft 1-90 and 114, remove snubber together with fuse, then separate snubber from fuse.

(4) Disconnect hydraulic lines and remove snubber.

(5) Cap and plug all lines and ports.

B. Installation

NOTE: On Aircraft 1-90 and 114, assemble fuse to snubber. Use O-ring between fuse and snubber.

(1) Remove caps and plugs from hydraulic lines and ports.

(2) Connect snubber to hydraulic lines with FLOW arrow pointing toward gage and tighten line.

(3) Start either engine and check for 1400 to 1500 psi normal hydraulic pressure.

(4) Check for leaks.

(5) Shut down engine.

(6) Inspect area for foreign objects and security of all attachments.

(7) Install access panel on right side of fuselage.

(8) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

13. Main Pressure Gage Fuse — Removal / Installation

(Aircraft 1-90 and 114, only)

A. Removal

(1) Operate brakes to deplete hydraulic pressure.

(2) Slowly open filler cap at top of reservoir to deplete remaining pressure.

(3) Open access panel on right side of fuselage, outboard of nose gear.

(4) Disconnect hydraulic lines and remove fuse and snubber as a unit.

(5) Cap and plug all lines and ports.

(6) Separate fuse from snubber.

B. Installation

(1) Assemble fuse with snubber, using new O-ring between fuse and snubber.

(2) Remove caps and plugs from hydraulic lines and ports.

(3) Connect assembly to hydraulic lines.

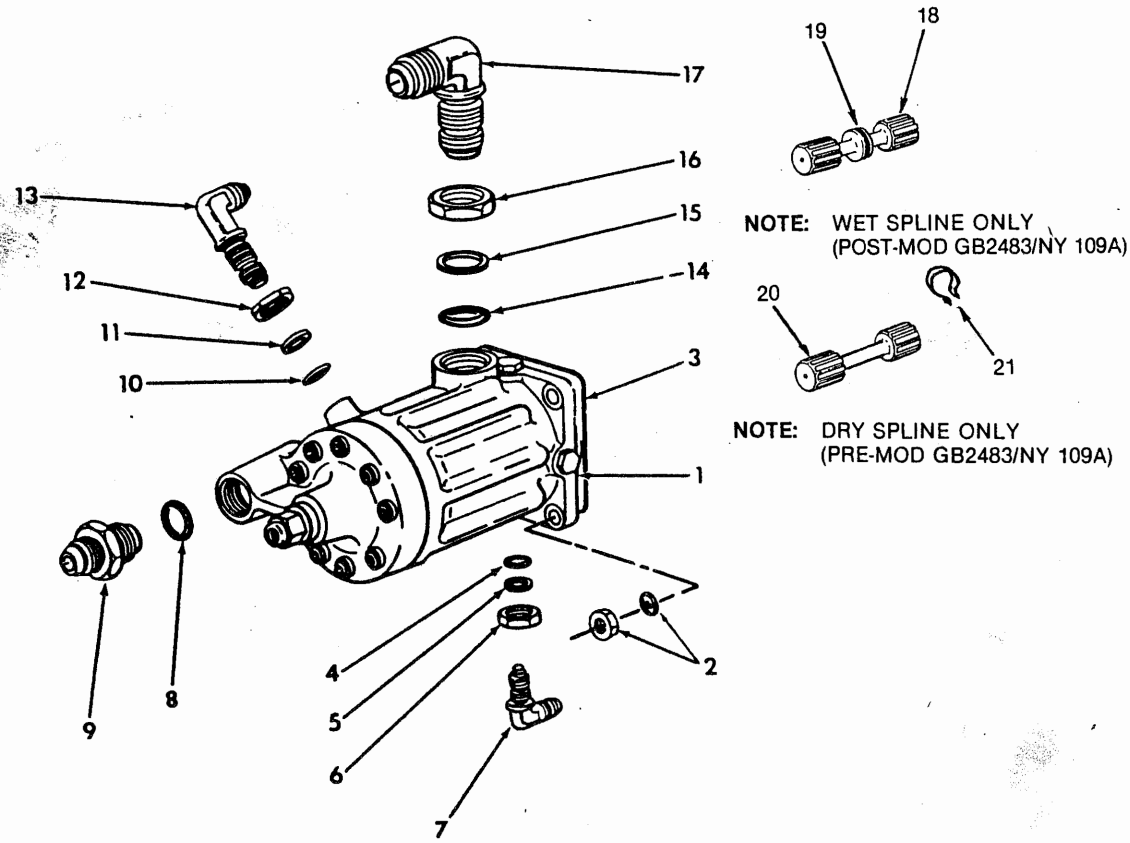
(4) Apply hydraulic power to aircraft and check for 1400 to 1500 psi, normal hydraulic pressure gage.

(5) Check for leaks.

(6) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(7) Inspect area for foreign objects and security of all attachments.

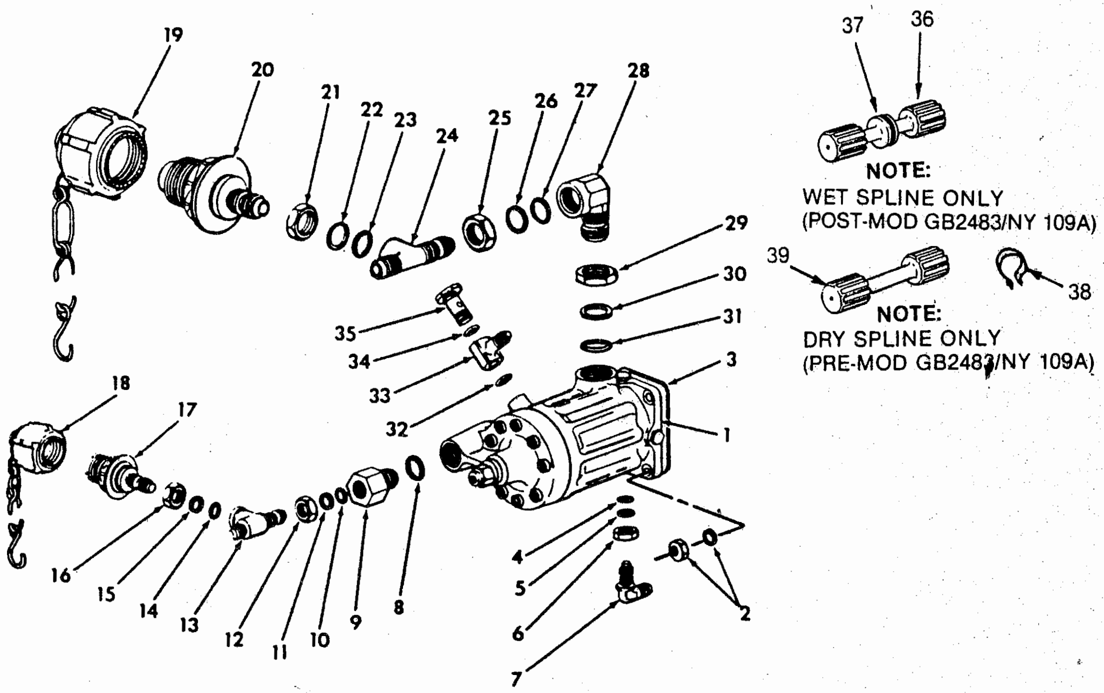
(8) Install access panel.



1. Hydraulic Pump	159SCH100-3/-5 (See * below)	10. O-ring	1159H20011-6
2. Nut (4)/Washer (4)	A27GP	11. Backup Ring	MS28773-6
3. Gasket	G28080	12. Universal Nut	AN6289D6
4. O-ring	1159H20011-4	13. Elbow	AN833-6D
5. Backup Ring	MS28773-4	14. O-ring	1159H20011-20
6. Universal Nut	AN6289D4	15. Backup Ring	MS28773-20
7. Elbow	AN833-4D	16. Universal Nut	AN6289D20
8. O-ring	1159H20011-10	17. Elbow	AN833-20D
9. Reducer	AN919-15D	18. Coupling	6406
		19. O-ring	6270-112
		20. Coupling	2476/2349
		21. Coupling retainer (Used on both couplings)	67A933

*159SCH100-3 (Pre-mod GB2483/NY109A)
*159SCH100-5 (Post-mod GB2483/NY109A)

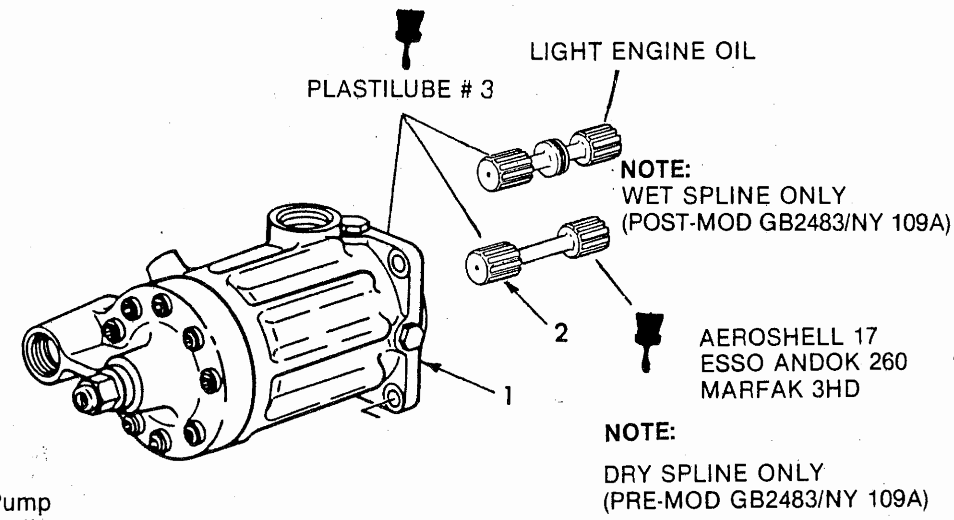
Left Hydraulic Pump
Figure 201.



1. Hydraulic Pump	159SCH100-3 / -5 (See * below)	19. Dust Cap Assy	015598-S7-16D
2. Nut (4)Washer (4)	A27GP	20. Coupling Half	015598-S4-14D
3. Gasket	G28080	21. Universal Nut	AN6289D16
4. O-ring	1159H20011-4	22. Back-up Ring	MS28773-16
5. Backup Ring	MS28773-4	23. O-ring	1159H20011-16
6. Universal Nut	AN6289D4	24. Tee	159HM10091-1
7. Elbow	AN833-4D	25. Universal Nut	AN6289D20
8. O-ring	1159H20011-10	26. Backup Ring	MS28773-20
9. Reducer Bushing	AN893-151D	27. O-ring	1159H20011-20
10. O-ring	1159H20011-8	28. Elbow	159HM10090-1
11. Backup Ring	MS28773-8	29. Universal Nut	AN6289D20
12. Universal Nut	AN6289D8	30. Backup Ring	MS28773-20
13. Tee	AN783C8	31. O-ring	1159H20011-20
14. O-ring	1159H20011-8	32. O-ring	1159H20011-6
15. Backup Ring	MS28773-8	33. Universal Elbow	NAS553-6D
16. Universal Nut	AN6289D8	34. O-ring	1159H20010-110
17. Coupling Half	015598-S4-8D	35. Universal Bolt	NAS551-6
18. Dust Cap Assy	015598-S7-8D	36. Coupling	6406
		37. O-ring	6270-112
		38. Coupling retainer (Used on both Couplings)	67A933
		39. Coupling	2476/2349

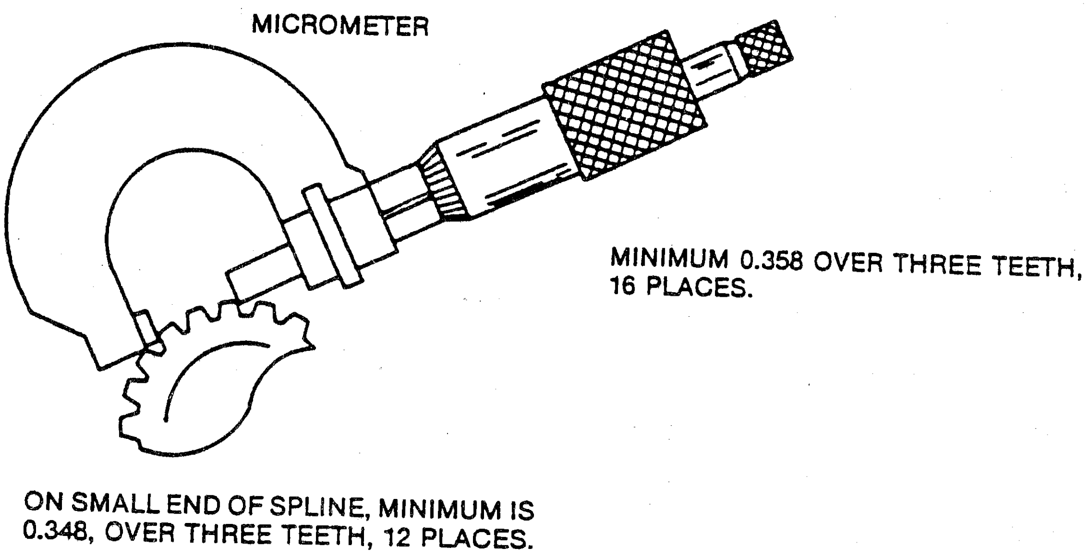
*159SCH100-3 (Premod GB2483/NY109A)
*159SCH100-5 (Postmod GB2483/NY109A)

Right Hydraulic Pump
Figure 202.

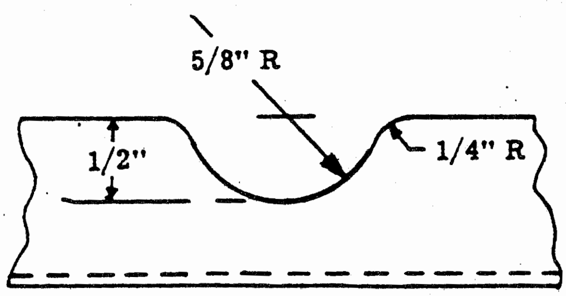


1. Hydraulic Pump
2. Splined Coupling

Hydraulic Pump Coupling Lubrication
Figure 203.



Hydraulic Pump Coupling Spline Inspection
Figure 204.



Drip Tray Modification
Figure 205.

AUXILIARY HYDRAULIC POWER SYSTEM — DESCRIPTION / OPERATION

1. General

The Auxiliary Hydraulic Power System components are located as follows:

UNIT	NO PER A/C	LOCATION
Electrically Driven Pump	1	Forward portion of nose wheel well on right side.
Auxiliary Filter	1	Center portion of nose wheel well on right side
Pressure Relief Valve	1	Near hydraulic reservoir above main pressure relief valve.
Hydraulic Fuse (gage) (Aircraft 1-90 and 114, only)	1	Access panel on right side of fuselage outboard of nose gear.
Pressure Snubber (gage)	1	Access panel on right side of fuselage outboard of nose gear.
Hydraulic Pressure Gage	1	Copilots right skirt panel.
DOOR CONT Circuit Breaker	1	Pilots circuit breaker panel.
AUX PUMP Current Limiter	1	Aft right side Fuselage Station 193.
Auxiliary Hydraulic Pump Relay	1	Mid relay box under floorboard No. 8

The auxiliary hydraulic power system develops, maintains and supplies 1500 psi hydraulic pressure when the pilots or the copilots AUX HYD PUMP switch is set to ON. Hydraulic fluid leaves the lowest point of a separate emergency compartment of the reservoir, from the suction drain port to the electrically driven pump. Fluid is directed from the reservoir to the electrically driven auxiliary hydraulic pump. The fluid, under auxiliary pump pressure, is routed through the filter, pressure relief valve, hydraulic fuse (Aircraft 1-90 and 114 only), pressure snubber and hydraulic pressure gage of the auxiliary system and operates the auxiliary hydraulic power system.

2. Electrically-Driven Pump

This 28V dc electrically driven, continuous duty, gear type, auxiliary hydraulic pump is rated at 1500 psi hydraulic pressure with an output of 0.5 gpm for the subsystems of the auxiliary hydraulic system. (See Section 29-0 Figure 1 and Figure 2) It is located in the forward portion of the nose wheel well on the right side. The pump is controlled by the pilots AUX HYD PUMP switch, located on the left console forward of the nose wheel steering control, or the copilots AUX HYD PUMP switch, located between the two hydraulic pressure gages. The pump has two ports, the suction inlet from the bottom of the reservoir and the pressure outlet to the filter. The pump receives 28V dc power from the main dc bus through a 50 ampere fuse (current limiter) located in the inverter fuse panel (See Figure 1). The pump is controlled by the hydraulic pump relay which is energized when either the pilots or copilots AUX HYD PUMP switch is ON. The main dc bus provides power to energize the relay through the DOOR CONT 5 ampere circuit breaker located on the circuit breaker panel in back of the pilots seat. The auxiliary hydraulic pump may also be operated for airstair door control with the pilots or copilots AUX HYD PUMP switch set to OFF. This is accomplished when the OUTSIDE DOOR SW is in the OPEN or CLOSE position, or when the AIRSTAIR SW is in the UP or DOWN position, through the DOOR CONT circuit breaker, located on the pilots circuit breaker panel, through the UNSAFE position of the DOOR SAFETY switch and through the unlock positions of the three main door lock limit switches. The inside door control switch is located on the outboard side of the pilots circuit breaker panel; the outside door control switch is located on the nose wheel well junction box; the DOOR SAFETY SW is located on the lower overhead panel and the three main door lock limit switches are located in the fuselage structure around the door frame. For more detailed information about the door and stairs system, See Chapter 52.

3. Pressure Relief Valve

The pressure relief valve is located downstream of the auxiliary hydraulic system filter near the hydraulic reservoir. (See Section 29-0 Figure 1 and Figure 2) This valve protects the auxiliary hydraulic system against excessive pressures. It is rated for a maximum full flow pressure of 1.2 gpm at 1900 psi. The minimum reseat pressure for the relief valve is 1517 psi. Fluid under excessive pressures leaves the valve at the return port (marked OUTLET) to return to the reservoir.

4. Filter, Automatic Resetting Hydraulic Fuse, Pressure Snubber and Pressure Gage

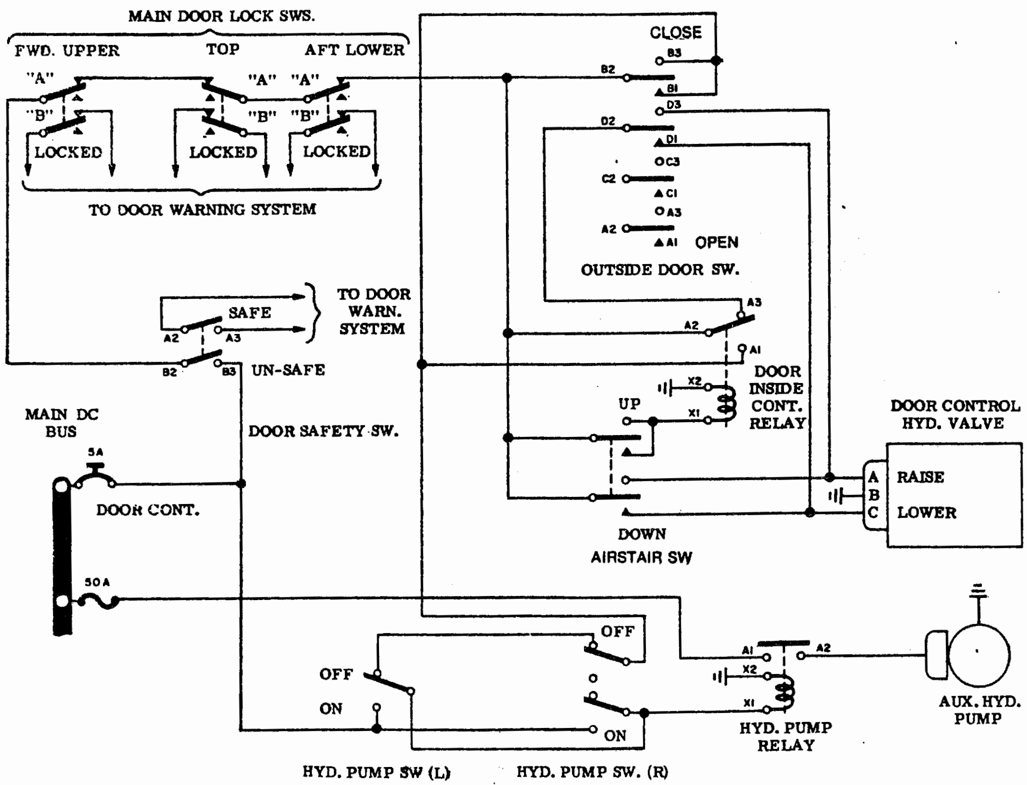
These components are similar to those used in the main hydraulic power system. For description and operation refer to appropriate paragraphs in this chapter. (See Section 29-0 Figure 1)

These units are located in the same vicinity as their counterparts in the main hydraulic power system, with the exception that the filter is located in the center portion of the nose wheel well on the right side.

5. Brake Accumulator

A brake accumulator is installed as an emergency source of pressure for the hand operated emergency and propeller brakes. The accumulator is located in the nose wheel well. A gage is connected to the air side of the accumulator and is located on the copilots instrument panel. When properly precharged with air (800 $\pm$ 25 psi), the accumulator will hold 22.5 cu. in. of fluid at 1500 psi. This is sufficient for five full applications of the PARK/EMER brake. The accumulator is connected to the auxiliary hydraulic power system, any time the auxiliary hydraulic pump operates it charges the accumulator. A check valve is provided to prevent the pressure from bleeding out when the auxiliary hydraulic pump is shut off. A preflight check of the precharge in the accumulator is required. Because of the check valve, it is necessary to evacuate the hydraulic fluid in the accumulator manually. With the auxiliary hydraulic pump off, actuate the PARK/EMER brake handle while observing the HYDRAULIC BRAKE ACCUMULATOR PRESSURE INDICATOR. When the gage stabilizes and does not drop further with application of the brake, it is indicating the air pressure. This must be within the limitation of 800 $\pm$ 25 psi.

NOTE: The accumulator will be recharged with hydraulic fluid under pressure when the main entrance door is closed electro-hydraulically, or by turning on the pilots or copilots AUX HYD PUMP switch.



Auxiliary Hydraulic Pump Control Circuit — Schematic
Figure 1.

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AUXILIARY HYDRAULIC POWER SYSTEM — FAULT ISOLATION

NOTE: In many respects, the fault isolation procedure for this system duplicates the main hydraulic power system trouble shooting procedure.

FAULT	POSSIBLE CAUSE	CORRECTION
Auxiliary pump fails to selected operation.	Low fluid level or no fluid in reservoir.	Check entire hydraulic system for leaks and refill reservoir.
	Defective auxiliary pump.	Replace auxiliary pump.
	Air leak or obstruction in suction line.	Remove and check suction line.
	Relief valve sticking open.	Replace relief valve.
	Leak in system.	Check system and repair leak.
	Filters clogged..	Replace filter elements.
	Check valve improperly installed.	Check for proper installation
	Defective selector valve.	Replace selector valve.
Auxiliary pump not operating.	Pilots or copilots auxiliary hydraulic pump switch defective.	Replace switch.
	Defective hydraulic pump relay.	Replace relay.
	Blown current limiter.	Replace current limiter.

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AUXILIARY HYDRAULIC POWER SYSTEM — MAINTENANCE PRACTICES

1. Hydraulic Pump Pressure Filter Assembly — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic system pressure.

(2) Slowly loosen filler cap on reservoir to deplete remaining pressure.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(3) Disconnect hydraulic lines from filter assembly in nose wheel well.

(4) Cap or plug exposed hydraulic ports and fittings.

(5) Remove filter assembly from structure. Note position of ports to facilitate proper installations.

B. Installation

(1) Remove caps and plugs from hydraulic ports and fittings.

(2) Install filter assembly on structure with ports properly positioned.

(3) Connect hydraulic lines and tighten.

(4) Connect electrical power to aircraft.

(5) Operate auxiliary pump and observe auxiliary pressure gage indicates 1500 psi, if gage fluctuates or indicates below normal pressure gage line, bleed gage line.

(6) Check filter assembly and lines for leakage.

(7) Inspect area for presents of foreign objects, security of all attachments.

(8) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

2. Hydraulic Pump Pressure Filter Element — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic system pressure.

(2) Slowly loosen filler cap on reservoir to deplete remaining pressure.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(3) Locate filters in nose wheel well, right hand side, aft of auxiliary hydraulic pump.

(4) Remove lockwire from bowl and unscrew bowl from housing.

(5) Remove filter element and discard O-ring.

CAUTION: WHEN PERFORMING THE FOLLOWING INSPECTION IF THERE IS ANY EVIDENCE OF METAL PARTICLES IN FILTER ELEMENT OR FILTER HOUSING INVESTIGATE FURTHER FOR POSSIBILITY OF SYSTEM CONTAMINATION CAUSED BY COMPONENT FAILURE.

(6) Inspect element for foreign matter.

B. Installation

(1) Install new element O-ring in filter housing and install filter element.

(2) Install new O-ring and backup ring on filter bowl and fill filter bowl with clean, approved hydraulic fluid.

(3) Install filter bowl to housing and lockwire.

(4) Connect electrical power to aircraft.

(5) Operate auxiliary pump. Observe auxiliary pump pressure gages indicates 1500 psi. If gage fluctuates or indicates below normal pressure, bleed gage line.

(6) Check filter for leakage.

(7) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(8) Inspect area for foreign objects, security of all attachments.

(9) Close up area.

3. Hydraulic Pump — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic pressure.

(2) Slowly open filler cap on reservoir to deplete remaining pressure.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(3) Drain emergency compartment of reservoir at bottom of auxiliary pump (pump is located in nose wheel well, right side, forward).

(4) Disconnect electrical leads at pump motor.

(5) Disconnect hydraulic lines from pump.

(6) Remove pump, motor and mount bracket as one unit. Cap and plug all lines and ports.

(7) Remove pump and motor as one unit from mounting bracket.

B. Installation

(1) Install pump and motor onto mounting bracket.

(2) Install pump, motor and mounting bracket in nose wheel well, right side, forward.

(3) Remove caps and plugs from all lines and ports.

(4) Connect hydraulic lines to pump and tighten.

(5) Connect electrical leads to pump motor.

(6) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(7) Apply electrical power to aircraft.

(8) Operate pump and observe auxiliary pump pressure gage 1500 psi. If gage fluctuates or indicates below normal pressure bleed gage line.

(9) Remove door safety struts and operate landing gear door system to purge air from auxiliary system.

(10) Inspect area for presents of foreign objects, security of all attachments.

(11) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

4. Hydraulic Pump Pressure Relief Valve — Removal / Installation

A. Removal

(1) Operate brakes to deplete hydraulic pressure.

(2) Slowly open filler cap on reservoir to deplete remaining pressure.

(3) Disconnect hydraulic lines from valve.

(4) Remove attaching hardware and remove valve from bracket.

NOTE: This valve is located inboard of the hydraulic reservoir above the main pressure relief valve.

(5) Cap and plug all lines and ports.

B. Installation

(1) Install valve onto mounting bracket and secure with attaching hardware.

(2) Remove caps and plugs from hydraulic lines and ports.

(3) Connect hydraulic lines to valve and tighten.

(4) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(5) Apply electrical power to aircraft.

(6) Operate auxiliary pump and observe pressure gage. If gage fluctuates or indicates below normal pressure bleed gage line.

(7) Disconnect electrical power from aircraft.

(8) Inspect area for foreign objects, security of all attachments.

(9) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(10) Inspect area for presents of foreign objects, security of all attachments.

(11) Close up area.

5. Auxiliary to Main Selector Valve — Removal / Installation

(Aircraft having ASC 109)

A. Removal

(1) Operate brakes to deplete hydraulic pressure.

(2) Slowly open filler cap on reservoir to deplete remaining pressure.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(3) Gain access to valve located in nose wheel well, left hand side.

(4) Remove handle assembly and torsion spring.

(5) Disconnect hydraulic lines and cap or plug lines or ports.

(6) Remove attaching hardware securing valve to mounting bracket.

(7) Remove valve.

B. Installation

(1) Install valve onto mounting bracket and secure with attaching hardware.

(2) Remove caps or plugs from hydraulic lines and ports.

(3) Connect lines to valve and tighten.

(4) Install torsion spring and handle assembly.

(5) Perform Auxiliary to Main Selector Valve - Operational Test, this Section. this Section

NOTE: Bleed lines while performing operational test by operating brakes.

(6) Check for leaks.

(7) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(8) Inspect area for foreign objects, security of all attachments then close area panels.

6. **Auxiliary to Main Selector Valve — Operational Test**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

CAUTION: ENSURE THAT MAIN HYDRAULIC SYSTEM IS NOT IN OPERATION WHEN PERFORMING OPERATIONAL TEST ON THE AUXILIARY TO MAIN SELECTOR VALVE.

A. Place battery switch to NORM.

B. Place AUX TO MAIN SELECTOR lever to FLIGHT.

C. Energize auxiliary pump. Observe both main and auxiliary system pressure gages. The auxiliary system pressure gage should indicate 1500 psi, the main system gage should indicate zero psi.

D. De-energize auxiliary pump.

E. Place auxiliary to main selector lever to AUX TO MAIN and pin lever.

F. Energize auxiliary pump. Observe both auxiliary and main system pressure gages. Both should indicate 1500 psi.

G. De-energize auxiliary pump.

WARNING: VALVE MUST BE RETURNED TO FLIGHT POSITION BEFORE FLIGHT.

H. Return auxiliary to main selector lever to FLIGHT by removing pin from lever.

I. If battery power is no longer required, place battery switch to OFF.

7. **Hydraulic Pump Fuse — Removal / Installation**

A. Removal

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT.

(1) Gain access to inverter fuse panel on aft side of Fuselage Station 193 bulkhead, right side.

(2) Locate auxiliary hydraulic pump fuse (current limiters - bolt down type).

(3) Loosen hold down nuts at each end of fuse and slide fuse out from under nut and bus bar, do not remove nuts.

(4) Note value of fuse.

B. Installation

(1) Install fuse; same value as one removed.

(2) Ensure forked ends of fuse are securely under bus bar stud and stud at wire connection.

(3) Torque nuts as follows:

(a) Using torque wrench, tighten stop nut until just short of making contact with terminal or bus bar; note torque to move nut.

(b) Add this torque to value stud requires (65-95 inch pounds) and tighten to final torque.

(4) Inspect area for foreign objects, security of all attachments.

(5) Close access panel.

8. **Pressure Gage Snubber — Removal / Installation**

A. Removal

(1) Operate brakes to deplete hydraulic pressure.

(2) Slowly open filler cap at top of reservoir to deplete remaining pressure.

(3) Open access panel on right side of fuselage, outboard of nose gear.

NOTE: On Aircraft 1-90 and 114, remove snubber together with fuse, then separate snubber from fuse.

(4) Disconnect hydraulic lines and remove snubber.

(5) Cap and plug all lines and ports.

B. Installation

NOTE: On Aircraft 1-90 and 114, assemble fuse to snubber. Use O-ring between fuse and snubber.

(1) Remove caps and plugs from hydraulic lines and ports.

(2) Connect snubber to hydraulic lines with FLOW arrow pointing toward gage and tighten lines.

(3) Start either engine and check for 1400 to 1500 psi normal hydraulic pressure indicated.

(4) Check for leaks.

(5) Shut down engine.

(6) Inspect area for foreign objects, security of all attachments.

(7) Install access panel on right side of fuselage.

(8) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

9. Pressure Gage Fuse — Removal / Installation

(Aircraft 1-90 and 114, only)

A. Removal

(1) Operate brakes to deplete hydraulic pressure.

(2) Slowly open filler cap at top of reservoir to deplete remaining pressure.

(3) Open access panel on right side of fuselage, outboard of nose gear.

(4) Disconnect hydraulic lines and remove fuse and snubber as a unit.

(5) Separate fuse from snubber.

(6) Cap and plug all lines and ports.

B. Installation

(1) Assemble fuse with snubber. Use O-ring between fuse and snubber.

(2) Remove caps and plugs from hydraulic lines and ports.

(3) Connect assembly to hydraulic lines and tighten.

(4) Apply hydraulic power to aircraft and check for 1400 to 1500 psi, normal hydraulic pressure.

(5) Check for leaks.

(6) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (see Section 12-1-6)

(7) Inspect area for foreign objects, security of all attachments.

(8) Install access panel.

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EMERGENCY AIR SYSTEM — DESCRIPTION / OPERATION

1. Description

In the event of main hydraulic power system failure, the emergency air system supplies 1950 psi pneumatic pressure to extend the landing gear only. A single control handle located on the copilots side console when pulled upward releases the dry compressed air or nitrogen into the landing gear system. A gage for the pneumatic bottle is located in the nose wheel well and indicates, the amount of charge within the bottle. A preflight check is required and the limitations are 1950 psi $\pm$ 50 psi. For more detailed information about emergency extension of the landing gear, See Chapter 32, Landing Gear.

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MAIN AND AUXILIARY HYDRAULIC POWER SYSTEMS RESERVOIR — DESCRIPTION / OPERATION

1. Description and Operation

The dual-chambered hydraulic reservoir has a total capacity of 4.9 US gallons (4.08 Imperial gallons; 18.55 liters) of hydraulic fluid which is not readily flammable. The working capacity of the main system chamber is 3.1 gallons; the capacity of the auxiliary system chamber is 1.8 gallons. The reservoir is located inside the left side of the fuselage, just aft of the doorway on the bulkhead. (See Section 29-0 Figure 1 and Figure 2) A vent and relief valve is provided to protect the reservoir against excessive positive and negative pressures. The reservoir is a welded aluminum unit with an emergency filler neck and filler plug, and a normal filler neck and filler plug. The normal filler is used at regular periods or when needed during ground servicing. The emergency filler may be used during flight. Screens are incorporated in the filler neck to filter out dirt and foreign objects. A glass sight gage is installed in the reservoir for use by ground and flight personnel in determining fluid level. The lowest marking on the sight gage is the REFILL LEVEL, the middle marking is the GROUND FULL LEVEL, and the highest marking is the FLIGHT FULL LEVEL. The filler plugs are attached to the filler necks with chains to prevent their loss. Hydraulic fluid is supplied to the main hydraulic system through two outlet port fittings at the right and left side near the bottom of the reservoir. Hydraulic fluid is supplied to the electrically-driven hydraulic pump through one auxiliary outlet fitting at the bottom center of the reservoir. Two separate drain fittings, one for the main and one for the auxiliary chamber, are provided for draining hydraulic fluid from the reservoir. The drain fitting for the main chamber only is on the reservoir. The drain fitting for the emergency system is on the suction line tee at the auxiliary hydraulic pump. Hydraulic fluid from all the systems is returned to the reservoir through five inlet ports. The reservoir should be filled when the aircraft is on the ground or on jacks and with the landing gear fully extended, the airstair door open and the brake accumulator charged. The reservoir should be kept filled to the GROUND FULL LEVEL shown on the sight gage. The FLIGHT FULL LEVEL should be checked with the aircraft in a clean configuration; landing gear up and locked, airstair door closed, and brake accumulator charged.

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MAIN AND AUXILIARY HYDRAULIC POWER SYSTEMS RESERVOIR — MAINTENANCE PRACTICES

1. **Hydraulic Fluid Maintenance**

   A. Aircraft hydraulic fluid should be kept as clean as possible to ensure long life of components. Caution should be used whenever servicing the reservoirs or connecting a ground cart not to introduce any contaminants. When replacing components, any open hydraulic lines should be immediately capped with protective plugs.

   B. Clean, used hydraulic fluid will have a color close to the original new color. If the fluid becomes dark brown in color, it is recommended that the system be flushed and refilled with new fluid.

   C. The fluid should be checked visually for solid particle contaminates at the frequency specified in the inspection schedule. In service aircraft fluid contamination should not exceed standard NAS1638 Class 8. Excessive fluid contamination will reduce the life of individual components.

2. **Reservoir Relief Valve — Removal / Installation**

   A. Removal

      (1) Remove hydraulic power from aircraft.

      (2) Operate brakes to dissipate residual hydraulic system pressure.

      (3) Slowly open filler cap at top of reservoir to relieve head pressure.

      (4) Locate valve above reservoir and disconnect hydraulic lines.

      (5) Remove clamp that secures valve to bulkhead.

      (6) Remove valve.

      (7) Cap and plug all open lines and ports.

   B. Installation

      (1) Position valve on bulkhead and secure with clamp.

      (2) Remove caps and plugs.

      (3) Connect hydraulic lines to valve.

      (4) Tighten valve cap on reservoir.

      (5) Connect hydraulic power to aircraft and pressurize hydraulic system.

      (6) Check valve and overboard drain for leakage.

      (7) Inspect area for presents of foreign objects, security of all attachments.

      (8) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (See Section 12-1-6)

3. **Reservoir Relief Valve Filter Element — Removal / Installation**

   A. Removal

      (1) Operate brakes to deplete hydraulic pressure.

      (2) Slowly open filler cap at top of reservoir to deplete remaining pressure.

      (3) Locate valve above reservoir and disconnect hydraulic lines.

      (4) Remove clamp that secures valve to bulkhead, remove valve, cap and plug all open lines and ports.

      (5) Remove safety wire and unscrew cap from valve, remove filter element.

   B. Installation

      (1) Install new filter element and O-ring.

      (2) Install cap and safety wire.

      (3) Remove caps and plugs from all open lines and ports.

(4) Install relief valve on bulkhead.

(5) Connect all lines to check valve and tighten to correct torque.

(6) Apply hydraulic power to aircraft and pressurize hydraulic system.

(7) Check valve and overboard drain for leaks.

(8) Inspect area for presents of foreign objects, security of all attachments.

(9) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (See Section 12-1-6)

4. **Hydraulic Reservoir — Removal / Installation**

A. Removal

(1) Remove hydraulic power from aircraft.

(2) Operate brakes to dissipate residual hydraulic system pressure.

(3) Open both reservoir filler caps and remove scupper clamp at outboard filler neck.

(4) Drain hydraulic reservoir by using one of the following methods:

(a) Drain hydraulic fluid from reservoir at reservoir drain or at auxiliary pump drain, in nose wheel well, into a clean container.

(b) Connect hydraulic power unit suction line to service connection in right nacelle and ensure fire pull T-handles are in. Open hydraulic power unit tank valve and allow fluid to gravity feed from reservoir to power unit. When draining is completed, remove hydraulic power unit suction line from engine nacelle.

(5) Disconnect, cap, plug and tag all hydraulic lines at reservoir.

(6) Remove turnbuckles from straps that secure reservoir to structure.

(7) Remove reservoir.

B. Installation

(1) Install reservoir on structure and secure with clamps and turnbuckles. Do not tighten turnbuckles until all hydraulic lines are connected.

(2) Identify hydraulic lines, remove caps and plugs. Connect lines to reservoir fittings.

(3) Tighten turnbuckles on clamps and lockwire. Install clamp at outside filler scupper.

(4) Install clean filter elements in reservoir.

(5) Ensure reservoir drain fittings are closed (one under reservoir and other in nose wheel well).

(6) Service reservoir, with approved type fluid. Tighten reservoir cap(s). (See Section 12-1-6)

(7) Start both engines to ensure pumps are properly primed. Engine pump gage should indicate normal system pressure without fluctuating.

(8) Operate flaps both in normal and emergency modes to purge any air that may be in system.

(9) Inspect area for presents of foreign objects, security of all attachments.

(10) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). See Section 12-1-6)

5. **Hydraulic Reservoir / Fluid / Filters — Inspection**

A. Remove filler caps from reservoir

B. Drain hydraulic reservoir as follows:

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED

(1) Use thoroughly clean, open container and drain hydraulic fluid from reservoir at reservoir drain at auxiliary hydraulic pump drain in nose wheel well

C. Examine drained fluid for contamination, discoloration and presence of metal particles

NOTE: If metal particles are found in drained fluid, inspect hydraulic filters and replace as necessary, see this Section.

D. Remove retainer ring at each filler neck and remove screens and inspect.

E. Clean screens thoroughly with approved solvent; dry thoroughly.

F. Inspect reservoir for corrosion and condition.

G. Install screens and retainer rings.

H. Install drain plugs at reservoir and auxiliary pump

I. Replace O-rings in filler cap.

J. Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (See Section 12-1-6)

K. Inspect reservoir for leaks and security

L. Inspect area for foreign objects.

M. Close-up area.

6. Hydraulic Reservoir Return Filter Assembly — Removal / Installation

A. Removal

(1) Operate brakes to dissipate hydraulic system pressure.

(2) Slowly loosen reservoir filler cap to dissipate any remaining reservoir pressure.

(3) Remove electrical power from aircraft.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(4) Disconnect hydraulic lines from filter on bulkhead inboard of hydraulic reservoir.

(5) Cap or plug exposed hydraulic ports and fittings.

(6) Remove filter from bulkhead. Note position of ports to facilitate proper installation.

B. Installation

(1) Locate replacement filter on bulkhead with ports in proper position.

(2) Secure filter to bulkhead.

(3) Remove caps and plugs from ports and fittings.

(4) Connect hydraulic lines to filter.

(5) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (See Section 12-1-6)

(6) Tighten cap on reservoir.

(7) Start either engine. Check filter and lines for fluid leakage.

(8) Shut down engine.

(9) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (See Section 12-1-6)

(10) Close-up area.

7. Hydraulic Reservoir Return Filter Element — Removal / Installation

CAUTION: FACET AND PUROLATOR FILTER ELEMENTS ARE NOT INTERCHANGEABLE.

A. Removal

(1) Operate brakes to dissipate hydraulic system pressure.

(2) Slowly loosen reservoir filler cap to dissipate any remaining reservoir pressure.

(3) Remove electrical power from aircraft.

(4) Locate filter on bulkhead inboard of hydraulic reservoir.

(5) Remove safetywire and unscrew filter bowl from housing.

(6) Remove filter element and discard O-rings.

(7) Inspect element for foreign matter before discarding.

CAUTION: IF THERE IS ANY EVIDENCE OF METAL PARTICLES IN FILTER ELEMENT OR FILTER HOUSING, INVESITGATE FURTHER FOR POSSIBILITY OF SYSTEM CONTAMINATION CAUSED BY A COMPONENT FAILURE.

B. Installation

(1) Install new O-ring in filter housing and install new filter element.

(2) Install new O-ring and backup ring on filter bowl.

(3) Install filter bowl in housing and safetywire.

(4) Start engine and check for 1400 - 1500 psi hydraulic pressure.

(5) Check for leaks.

(6) Shutdown engine.

(7) Service reservoir, if needed, with approved type fluid. Tighten reservoir cap(s). (See Section 12-1-6)

(8) Inspect area for foreign objects and security of all attachments.

(9) Close-up area.

MAIN AND AUXILIARY HYDRAULIC POWER SYSTEMS VENT AND RELIEF VALVE — DESCRIPTION / OPERATION

1. Description and Operation

The vent and relief valve functions in the reservoir to maintain barometric pressure, or relieve excessive air pressure resulting from the return of large volumes of fluid during the operation of large capacity units. It is located in the hydraulic compartment above the reservoir. (See Section 29-0 Figure 1 and Figure 2) By maintaining a pressure on the hydraulic fluid in the reservoir, the valve minimizes cavitation in the engine-driven hydraulic pumps and the electrically-driven auxiliary hydraulic pump. The relief valve is designed to crack when air pressure trapped in the reservoir is 16 $\pm$ 1 psi above outside barometric pressure. The vent valve is designed to open when cabin pressurization is 0.5 psi above the trapped air in the reservoir. A filter is installed in the air inlet to eliminate dirt and foreign objects. The relief and vent ports are open to cabin and atmosphere respectively. The reservoir port is connected to the top center fitting of the hydraulic reservoir.

MAIN AND AUXILIARY HYDRAULIC POWER SYSTEMS VENT AND RELIEF VALVE — MAINTENANCE PRACTICES

1. **Main Relief Valve — Removal / Installation**

   A. Removal

   (1) Ensure hydraulic power is off.

   (2) Operate brakes to dissipate hydraulic system pressure.

   (3) Slowly open filler cap of reservoir to dissipate any remaining reservoir pressure.

   (4) Locate main relief valve inboard of reservoir.

   (5) Disconnect lines from valve.

   (6) Remove bolts that secure valve to bulkhead and remove valve.

   (7) Cap and plug all lines and ports.

   B. Installation

   (1) Install valve on bulkhead and secure with bolts.

   (2) Remove cap and plugs from lines and ports.

   (3) Connect hydraulic lines to valve.

   (4) Tighten valve cap on reservoir.

   (5) Connect hydraulic power to aircraft and pressurize hydraulic system.

   (6) Observe valve maintains system pressure.

   (7) Check valve for leakage.

   (8) Check hydraulic reservoir fluid level.

2. **Main Pressure Filter Assembly — Removal / Installation**

   A. Removal

   (1) Operate brakes to dissipate hydraulic system pressure.

   (2) Slowly loosen reservoir filler cap to dissipate any remaining reservoir pressure.

   (3) Pull engine FIRE PULL T-handle to close hydraulic shutoff valve of side involved.

   (4) Remove inboard accessory access cover from engine where filter will be removed.

   (5) Disconnect hydraulic lines from filter.

   NOTE: Observe position of filter ports during removal to facilitate installation.

   (6) Remove filter.

   B. Installation

   (1) Install filter on structure in same position noted during removal.

   (2) Secure filter to structure with hardware.

   (3) Connect hydraulic lines to filter.

   (4) Check hydraulic reservoir fluid level.

   (5) Push in engine FIRE PULL T-handle to open shutoff valve.

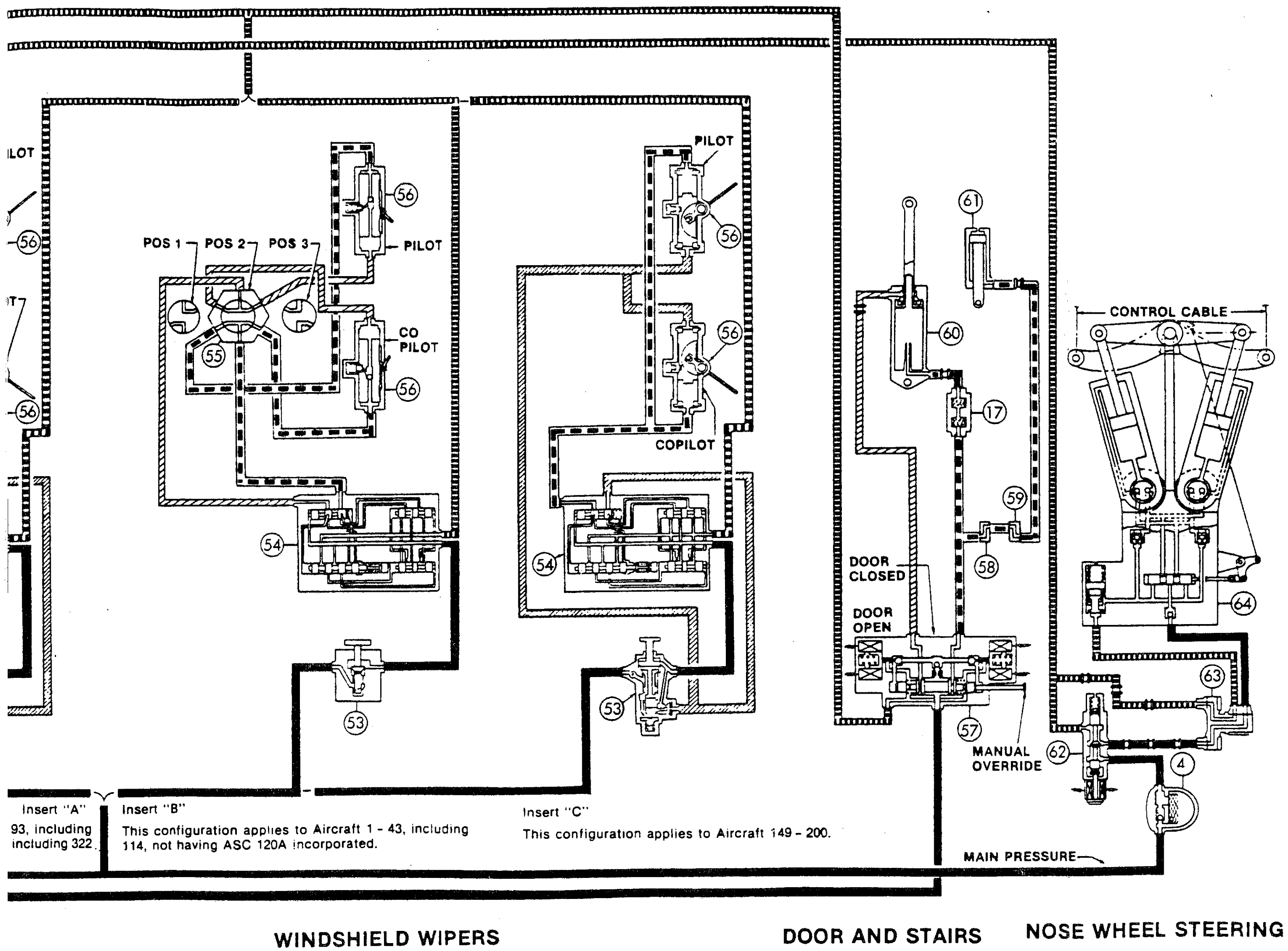
   (6) Start applicable engine and check that normal system pressure is obtained.

   (7) Check for hydraulic fluid leakage.

   (8) After engine is shut down install cowling removed.

CODE

- PRESSURE
- RETURN
- UP OR OPEN
- DOWN OR CLOSE
- EMERGENCY AIR OR FLUID



PARTS NOMENCLATURE

1. Vent and Relief Valve
2. Reservoir
3. Shutoff Valve
4. Filter
5. Ground Test Connection
6. Engine Driven Hydraulic Pump
7. Check Valve
8. Electrically Driven Auxiliary Hydraulic Pump
9. Main Pressure Relief Valve
10. Auxiliary Pressure Relief Valve
11. Hydraulic Fuse
12. Pressure Snubber
13. Main Hydraulic Pressure Gage
14. Auxiliary Hydraulic Pressure Gage
15. Flap Shutoff Valve
16. Emergency Flap Selector Valve
17. Restrictor
18. Flap Control Valve
19. Flap Dump Valve (With manual reset)
20. Flap Shuttle Valve
21. Flap Hydraulic Motor
22. Flap Gearbox
23. Wheel Doors Selector Valve (To open wheel doors for ground servicing)
24. Thermal Relief Valve
25. Shuttle Valve
26. Nose Gear Uplock Cylinder
27. Nose Gear Actuating Cylinder
28. Nose Gear Door Cylinder
29. Timer Valve
30. Nose Gear Door Control Valve
31. Nose Gear Selector Valve
32. Landing Gear Dump Valve (With manual reset)
33. Main Gear and Speedbrake Selector Valve
34. LG Emergency Dump Valve
35. LG Emergency Air Release Valve
36. Air Bottle
37. Air Bottle Filler Valve
38. Air Pressure Gage
39. Main Gear Actuating Cylinder
40. Main Gear Door Cylinder
41. Main Gear Up Lock Cylinder
42. Main Gear Door Control Valve
43. Main Gear Down Lock Cylinder
44. Main Gear Drag Brace
45. Propeller Brake Selector Valve
46. Accumulator
47. Accumulator Air Filler Valve
48. Accumulator Air Pressure Gage (Cockpit)
49. Parking and Emergency Brake Selector Valve
50. Power Brake Valve
51. Wheel Brakes
52. Decelostat Unit
53. Speed Control Valve
54. Control Unit
55. Windshield Wiper Selector Valve
56. Window Unit
57. Door and Stair Selector Valve (with manual override)
58. Floor Swivel
59. Door Swivel
60. Door and Stair Cylinder
61. Down Latch Release Cylinder
62. Nose Wheel Steering Selector Valve
63. Steer Swivel
64. Steer Damper Unit
65. Auxiliary to Main Pressure System Jumper Connections
66. Filter
67. Pressure Relief Valve
68. Pressure Relief Valve
69. Hydraulic Shutoff Valve
70. Filter (See Note 4, Page 3/4)
71. Auxiliary to Main Selector Valve
72. Accumulator Air Pressure Gage Nose Gear Wheelwell (See Note 4)
73. Dump Valve, Manually Operated Nose Gear Wheelwell (See Note 4)

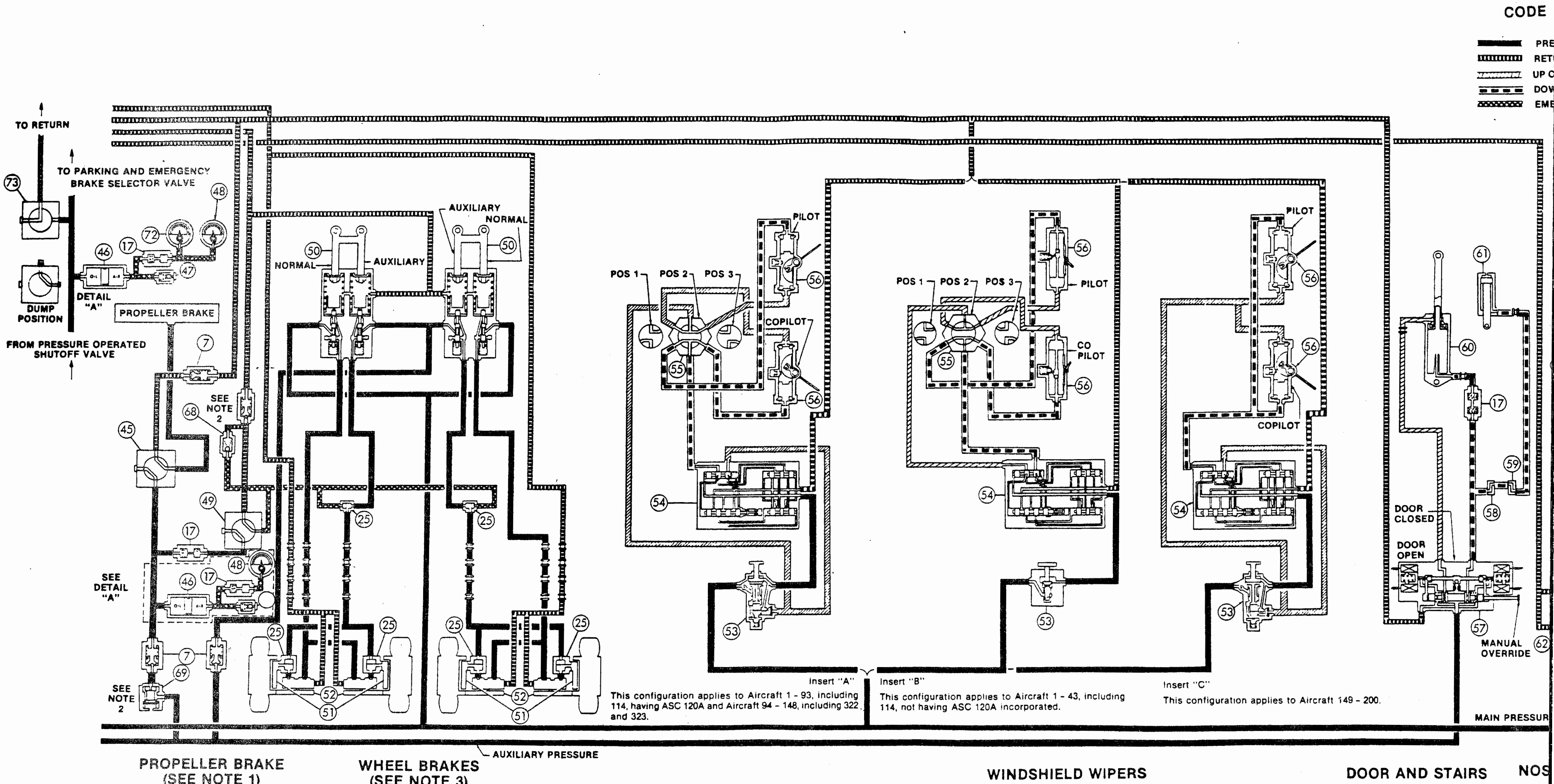
DIAGRAM SHOWS

BRAKES	RELAXED
PROPELLER BRAKE	OFF
WINDSHIELD WIPERS	OFF
DOOR AND STAIRS	CLOSED AND UP
NOSE WHEEL STEERING	DAMPER POSITION

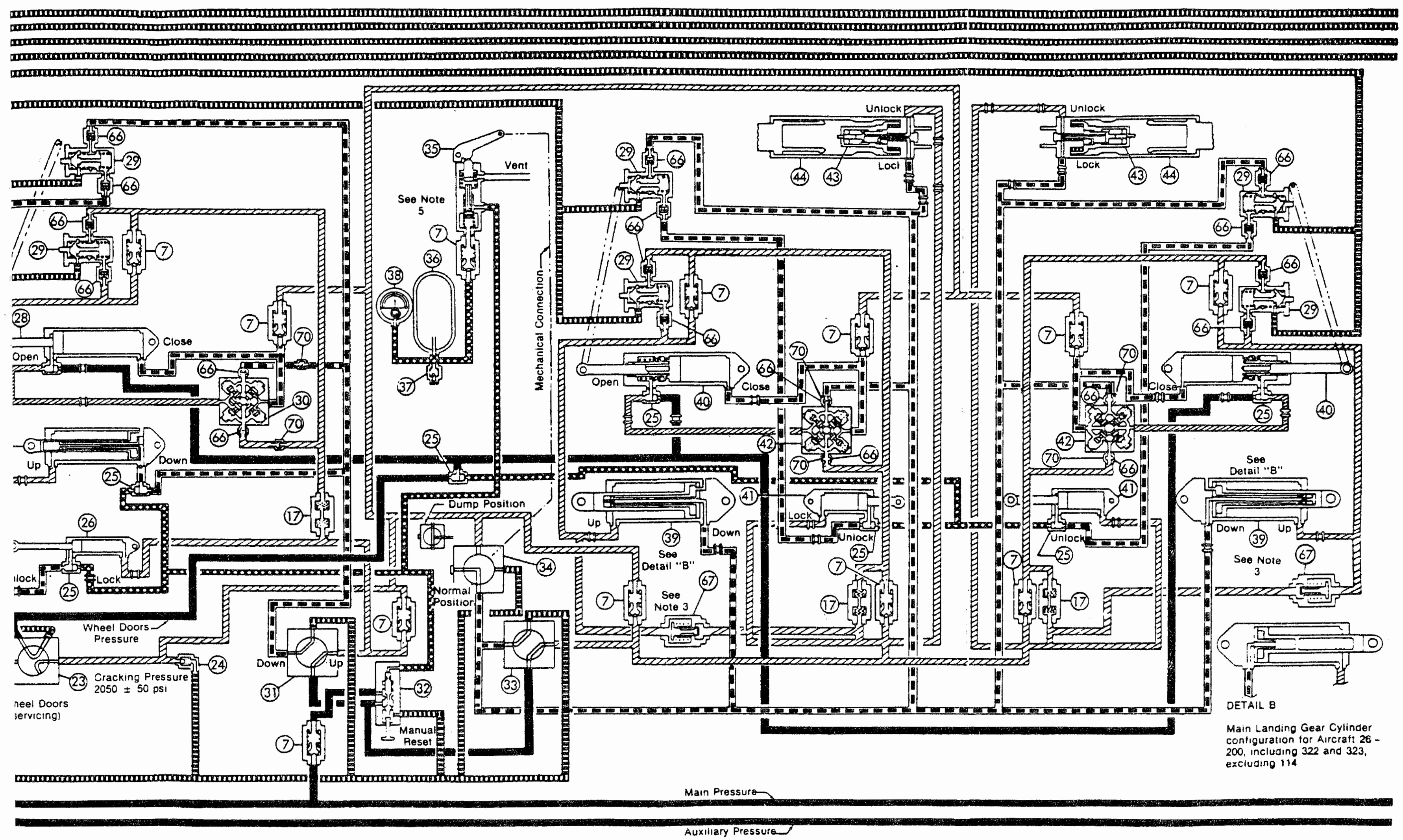
NOTES:

1. Aircraft 1 - 148, including 322 and 323 only.
2. Aircraft 1 - 87, including 114, having ASC 137 and Aircraft 88 - 200, including 322 and 323.
3. This configuration applies to Aircraft 1 - 200, including 322 and 323, not having ASC 206 for Aircraft having ASC 206, see Figure 3.
4. Aircraft 1 - 200, including 322 and 323, having ASC 214.

Hydraulic Power System — Schematic
Figure 1 (Sheet 1 of 2).



CODE
PRE
RET
UP C
DOW
EME
TO RETURN
TO PARKING AND EMERGENCY BRAKE SELECTOR VALVE
DUMP POSITION
DETAIL "A"
PROPELLER BRAKE
FROM PRESSURE OPERATED SHUTOFF VALVE
SEE NOTE 2
SEE DETAIL "A"
SEE NOTE 2
AUXILIARY
NORMAL
NORMAL
AUXILIARY
POS 1
POS 2
POS 3
PILOT
COPILOT
PILOT
CO PILOT
PILOT
COPILOT
DOOR CLOSED
DOOR OPEN
MANUAL OVERRIDE
Insert "A"
This configuration applies to Aircraft 1 - 93, including 114, having ASC 120A and Aircraft 94 - 148, including 322 and 323.
Insert "B"
This configuration applies to Aircraft 1 - 43, including 114, not having ASC 120A incorporated.
Insert "C"
This configuration applies to Aircraft 149 - 200.
MAIN PRESSUR
AUXILIARY PRESSURE
PROPELLER BRAKE (SEE NOTE 1)
WHEEL BRAKES (SEE NOTE 3)
WINDSHIELD WIPERS
DOOR AND STAIRS
NOS



NOSE GEAR **LANDING GEAR** **MAIN GEAR**

CODE

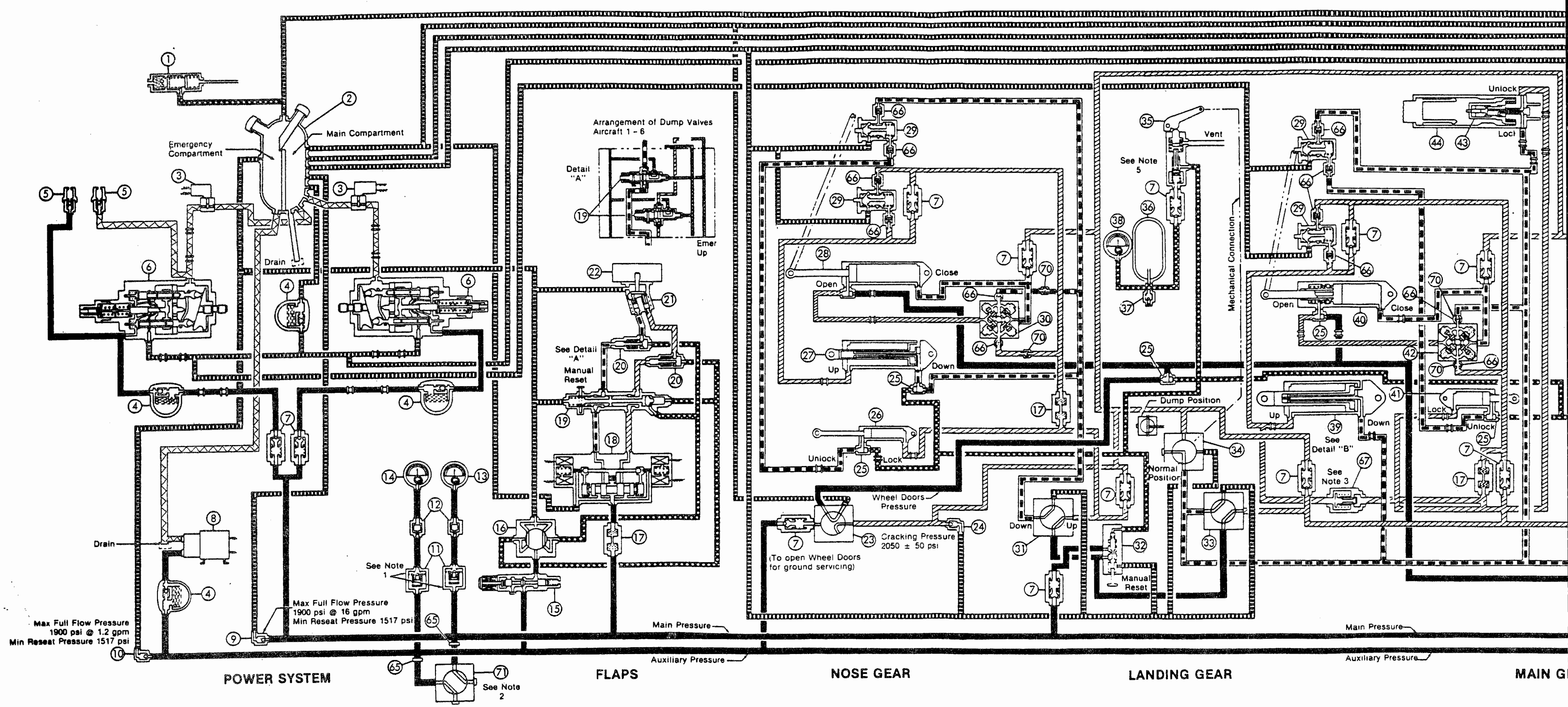
- PRESSURE
- RETURN
- PUMP SUCTION
- UP OR OPEN
- DOWN OR CLOSE
- EMERGENCY AIR OR FLUID

NOTES:

1. Aircraft 1 - 90, including 114 only.
2. Aircraft 1 - 200 including 322 and 323, having ASC 109.
3. Aircraft 1 - 87 including 114 having ASC 140 and Aircraft 88 - 200, including 322 and 323.
4. Aircraft 1 - 69, including 114, having ASC 94 and Aircraft 70 - 200 including 322 and 323.
5. This check valve is removed on Aircraft 1 - 200, including 322 and 323, having ASC 215

DIAGRAM SHOWS

FLAPS	UP
LANDING GEAR	UP



Emergency Compartment
Main Compartment
Drain
Arrangement of Dump Valves Aircraft 1 - 6
Detail "A"
Emer Up
See Detail "A"
Manual Reset
See Note 1
Max Full Flow Pressure 1900 psi @ 16 gpm Min Reseat Pressure 1517 psi
Max Full Flow Pressure 1900 psi @ 1.2 gpm Min Reseat Pressure 1517 psi
See Note 2
Main Pressure
Auxiliary Pressure
Close
Open
Up
Down
Unlock
Lock
Wheel Doors Pressure
Cracking Pressure 2050 ± 50 psi
(To open Wheel Doors for ground servicing)
See Note 5
Vent
Mechanical Connection
Dump Position
Normal Position
Manual Reset
See Detail "B"
See Note 3
POWER SYSTEM
FLAPS
NOSE GEAR
LANDING GEAR
MAIN GE

LIST OF EFFECTIVE PAGES

CHAPTER 30

ICE AND RAIN PROTECTION

* Pages revised by Revision 47
** Pages added by Revision 47
*** Pages deleted by Revision 47

TABLE OF CONTENTS

CHAPTER 30

ICE AND RAIN PROTECTION

ICE AND RAIN PROTECTION — DESCRIPTION / OPERATION

1. **General**

   A. Ice and rain protection for this aircraft is provided by:

   - Pneumatically operated wing and empennage deicer boots. Pneumatic power is supplied by bleed air from engines by various components of deicer system to the boots.
   - Electrically heated cockpit window panes. Electrical power is supplied by engine-driven alternators for windshield heating.
   - Pitot heads and stall warning lift transducer. Electrical power is supplied by the main dc bus, and essential dc bus (See Pitot Heat Systems this chapter).
   - Windshield wipers hydraulically operated from the main hydraulic system.

WING AND EMPENNAGE DE-ICING SYSTEM — DESCRIPTION / OPERATION

1. General

The de-icing of wings, vertical and horizontal stabilizers is accomplished by 13 rubber boots installed over the leading edges. (See Figure 1) installed on the wings have cordwise inflatable tubes. Horizontal and vertical stabilizer boot inflatable tubes are spanwise. Boots are cemented to the leading edges without the use of cap strips or screws. Boots are inflated and deflated by pressure and vacuum routed from solenoid operated distributor valves in a sequence controlled by an electronic timer. Pressure is supplied by bleed air from the second stage compressor of either or both engines. (See Figure 1). Suction is supplied by a bleed air operated ejector located in the inboard side of the right engine nacelle. Pressure and suction gages located on the upper left hand side of the upper overhead panel in the cockpit indicate operating pressure and suction in manifolds. (See Figure 3. Controls for operation of the system are located on the upper overhead panel, as shown in Figure 4. There are two switches, one for OFF and ON, the other for HEAVY ICE or LIGHT ICE.

A. Components (See Figure 1)

Wing and empennage de-icing system is comprised of the following components:

Unit	No Per A/C	Location
Electronic Timer	1	Under floorboard No. 7, rear of compartment.
Check Valves	4	One in each nacelle, two under floorboard panels No. 11 and No. 12.
Filters	2	Upstream of pressure regulator in each nacelle.
Pressure Regulators	2	One in each nacelle aft of filter.
Safety Valves	2	Downstream of regulator.
Solenoid Valves	7	Two in each nacelle, one in each outer wing panel and one in tail.
Ejector	1	In right nacelle.
Vacuum Relief Valve	1	In right nacelle outboard fillet.
Pressure Switches	2	In each boot supply line in tail.
Pressure Gage	1	Overhead panel, top left corner.
Vacuum Gage	1	Overhead panel, top left corner.
Minimum Flow Orifice	1	Under floorboard panels No. 11 and No. 12.
De-icer Boots, Pneumatic	13	Leading edge of the following areas. Five on each wing, one on each horizontal stabilizers and one on the vertical stabilizer.

(1) Electronic Timer (See Figure 2)

The electronic timer is a stepping switch with eight output poles, one for each A and B solenoids of the wing distributor valves and tail distributor valve. The switch timing is electronically controlled. When energized by moving the system ON/OFF switch to ON, after a 20 second tube warm-up period, the timer moves the stepping switch to the first contact. This contact energizes the A solenoid of the two distributor valves serving the boots inboard of each nacelle. After an interval determined by the timing circuit, the switch moves to the second contact which energizes the B solenoid of these valves in turn. Successive switching energizes all of the distributor valve solenoids. After energizing the last solenoid, the stepping switch returns to the start position and does not begin the next cycle until a dwell time, selected by the LIGHT ICE/HEAVY ICE switch has elapsed. If at any time during the cycle the system ON/OFF switch is moved to the OFF position, the stepping switch trips through the remaining position without energizing any of the remaining solenoids and returns to the starting position. The timing of the system operation is given in the

operational checkout procedure. Automatic boot cycling is not available during emergency dc power operation.

(2) Check Valves

The check valves are line mounted, spring-loaded flapper valves. One check valve is installed in each pressure supply line at the system pressure manifold to prevent loss of pressure from the manifold in the event of an engine failure, pressure regulator safety valve failure, or a broken supply line. Two other check valves are located in the fuselage to prevent loss of pressure for door seals, pressure control vacuum and blower control pressure if a line failure should occur.

(3) Filters

Each filter consists of a stainless steel housing with an inlet, outlet and drain port. The housing is made of two halves held together by a V-band clamp to permit disassembly for inspection or replacement of the filter cartridge. The filters are installed in the supply lines to the system pressure manifold upstream of the pressure regulators, safety valves and check valves to prevent any dirt in the engine bleed air from entering these parts or the distributor valve and boots. The drain fitting permits any condensation to drain overboard.

(4) Pressure Regulators

Installed in the housing of each pressure regulator is a diaphragm actuated regulating valve. On the main housing is another smaller housing containing an adjustable pilot valve. A screw for adjusting the outlet pressure of the regulator protrudes from this smaller housing.

The pressure regulator receives bleed air from the second stage compressor and supplies it to the system pressure manifold at 18 psi above ambient pressure. The regulator has no provision for shutoff and is continually supplying air to the ejector and other pneumatic systems using the pressure manifold as a source.

(5) Safety Valves

Each safety valve consists of a housing containing an inlet port, vent port and adjustment screw. The safety valve function is to protect the boots against excessive pressure in the event of a pressure regulator malfunction. The valves are located in the supply lines from the pressure regulators to the system pressure manifold. They vent air from these lines if the pressure in the lines exceed 22.5 psi above the ambient pressure.

(6) Solenoid Distributor Valves

Each distributor valve incorporates a pressure supply port, vacuum supply port, two solenoids A and B, two corresponding A and B outlet ports and a vent port. The pressure supply port is connected to the system pressure manifold and the vacuum port to the system vacuum manifold. The A and B outlet ports are connected to the A and B manifold supply ports of the boot. The vent port is open to ambient at the valve location. When energized the valve applies the vacuum pressure to both the A and B outlet ports for the in flight "hold down". When energized by the electronic timer, the A solenoid of the valve is first actuated, changing the supply for the A port from vacuum to pressure, thereby inflating all the A tubes in the boot. After an inflation period determined by the timer, the A solenoid is de-energized, cutting off the pressure supply to the A tubes of the boot and allowing the air in these tubes to escape through an integral check valve to the vent port. When the pressure in the tubes decreases to 0.5 psig, the valve once again applies vacuum to the A outlet port to evacuate the remaining air in the boot. At the same time that the A solenoid is de-energized by the timer, the B solenoid is energized and the B portion of the valve follows the same sequence.

(7) Ejector

The ejector consists of a venturi section with an inlet and outlet port at each end and a vacuum port at the venturi throat. The inlet port of the venturi is connected to the pressure manifold while the outlet port is vented to ambient. Whenever pressure is built up in the pressure manifold, air flows through the venturi section, reducing the static pressure in the throat below ambient. The port at the throat thus becomes a vacuum source and is connected to the system vacuum manifold.

(8) Vacuum Relief Valve

The vacuum relief valve is a line-mounted, spring-loaded negative pressure relief valve with a screened relief port and a safety-wired adjustment knob. The function of the vacuum relief valve is to limit the level of vacuum produced by the ejector. The valve is mounted in the system vacuum manifold and opens vent air to the manifold whenever the vacuum in the manifold is more than 6 inches of mercury. The screened relief port prevents foreign objects from jamming the valve.

(9) Pressure Switches

The pressure switches are three-wire, externally grounded, differential pressure switches. The pressure switches function is to provide cockpit indication that the tail boots are receiving pressure. The switches sense pressure in each of the lines from the A and B outlet ports of the tail boot distributor valve and compare it to the ambient pressure at their location. The pressure switches, which are normally open, close the circuit to the light whenever the differential pressure sensed by them is 10 psi or greater.

(10) Minimum Flow Orifice (See Figure 4)

The minimum orifice and check valves were installed by ASC 102 (Aircraft 1 - 80 and 114) and a similar change in production Aircraft 81 - 200, 322 and 323. The purpose of this installation is to provide a protected manifold pressure in the event of the loss of one engine or a break in the pressure lines. The minimum flow orifice installation ensures pressure will be maintained for the door seals, ejector tail distributor valves pressurization gage pneumatic air conditioning pressurization valves and full de-icer boot operation through the use of check valves or by the remaining engine.

(11) Pneumatic Boots

The pneumatic boots are of laminated rubber construction, the center lamination consisting of a series of inflatable tubes laid side by side. Pressure or vacuum is supplied to the inflatable tubes from a manifold or manifolds (depending upon the type of boot) which run perpendicular to the direction of the tubes. The manifolds are molded to the underside of the boot and have a 3/8 inch OD aluminum tube molded into them, through which air is supplied or evacuated. These supply tubes pass through holes in the leading edge skins. The skins are provided with a recess to receive the manifolds and are also recessed to receive the trailing edges of the boot itself. The boots are cemented into these recesses, the result being a flush installation in which the outer surface of the boot forms the leading edge of the airfoil, thus producing negligible drag.

When icing conditions are not present, the de-icing system is turned off. When the system is off, the distributor valves are so positioned that they port all boot manifolds to the vacuum source. This maintains a vacuum in the inflatable tubes of approximately 6 inches of mercury below ambient, thus preventing inflation of these tubes in the low presure areas of the airfoils which could occur if the tubes contained ambient pressure. When icing is encountered and the de-icing system is turned on, operation of the boots is as follows:

(a) Vertical Fin Boot

The inflatable tubes on the vertical fin boot run spanwise and are supplied by two manifolds (A and B) running cordwise. The center tube on the boot is attached to manifold A, those on either side are attached to manifold B and progress outward alternating from A to B.

During operation, pressure is supplied to manifold A while vacuum pressure is applied to manifold B and alternates from A to B in predetermined time intervals.

(b) Wing Panel Boots

The inflatable tubes in the wing panel boots run chordwise while the manifolds run span-wise.There are two manifolds (A and B), inflatable tubes are connected to opposite manifolds. The inflation cycle is identical to that described for vertical fin boots.

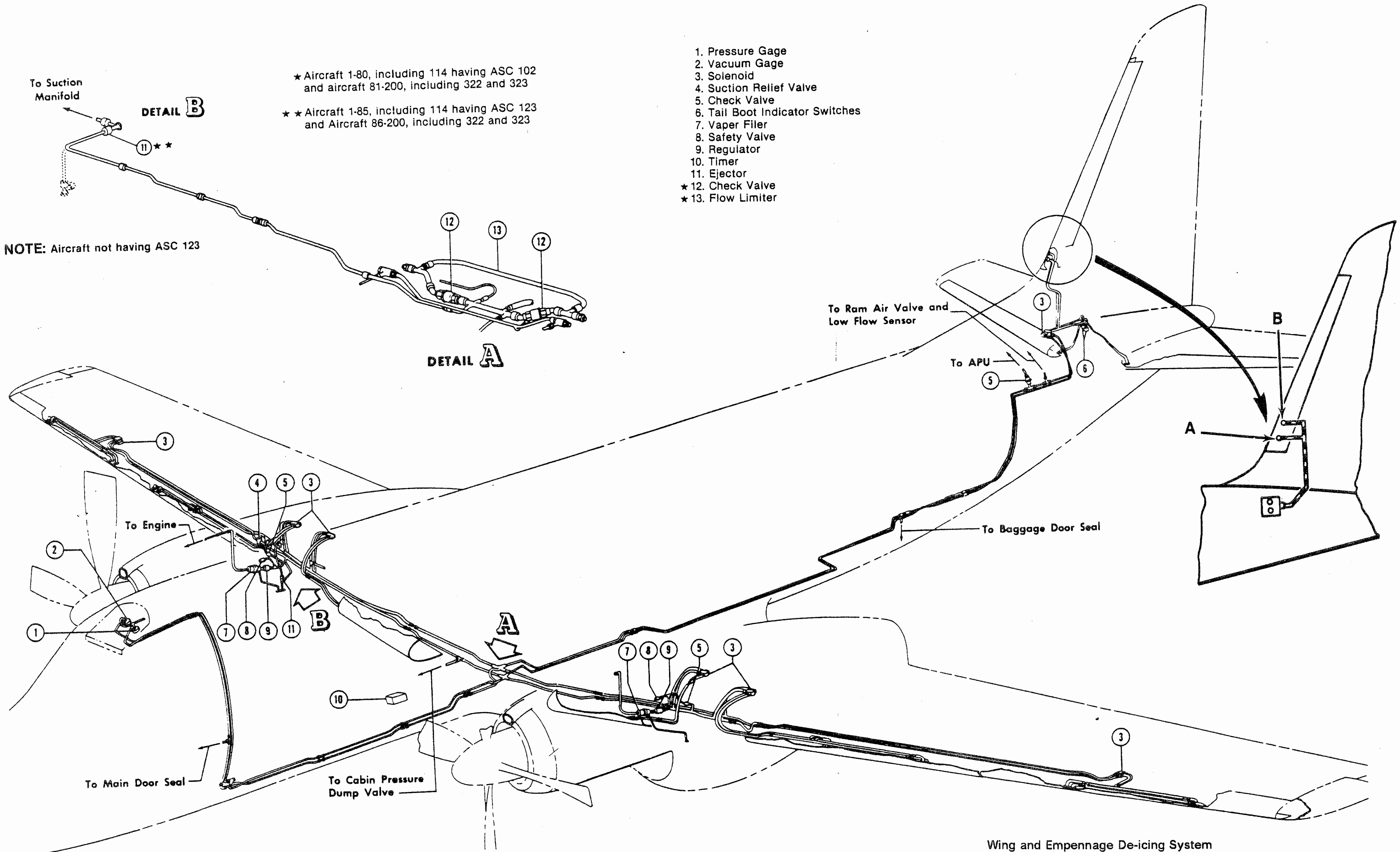
(c) Stabilizer Boots

The stabilizer boots have spanwise inflatable tubes like those in the vertical fin boots. However, all the tubes are connected to a single manifold supplied by a single port. This

port is connected to only one of the outer ports of a distributor valve so that when the valve operates, all the tubes in the boot are simultaneously supplied with either vacuum or pressure. The A port of the distributor valve is connected to the right horizontal stabilizer, and the B port is connected to the left horizontal stabilizer.

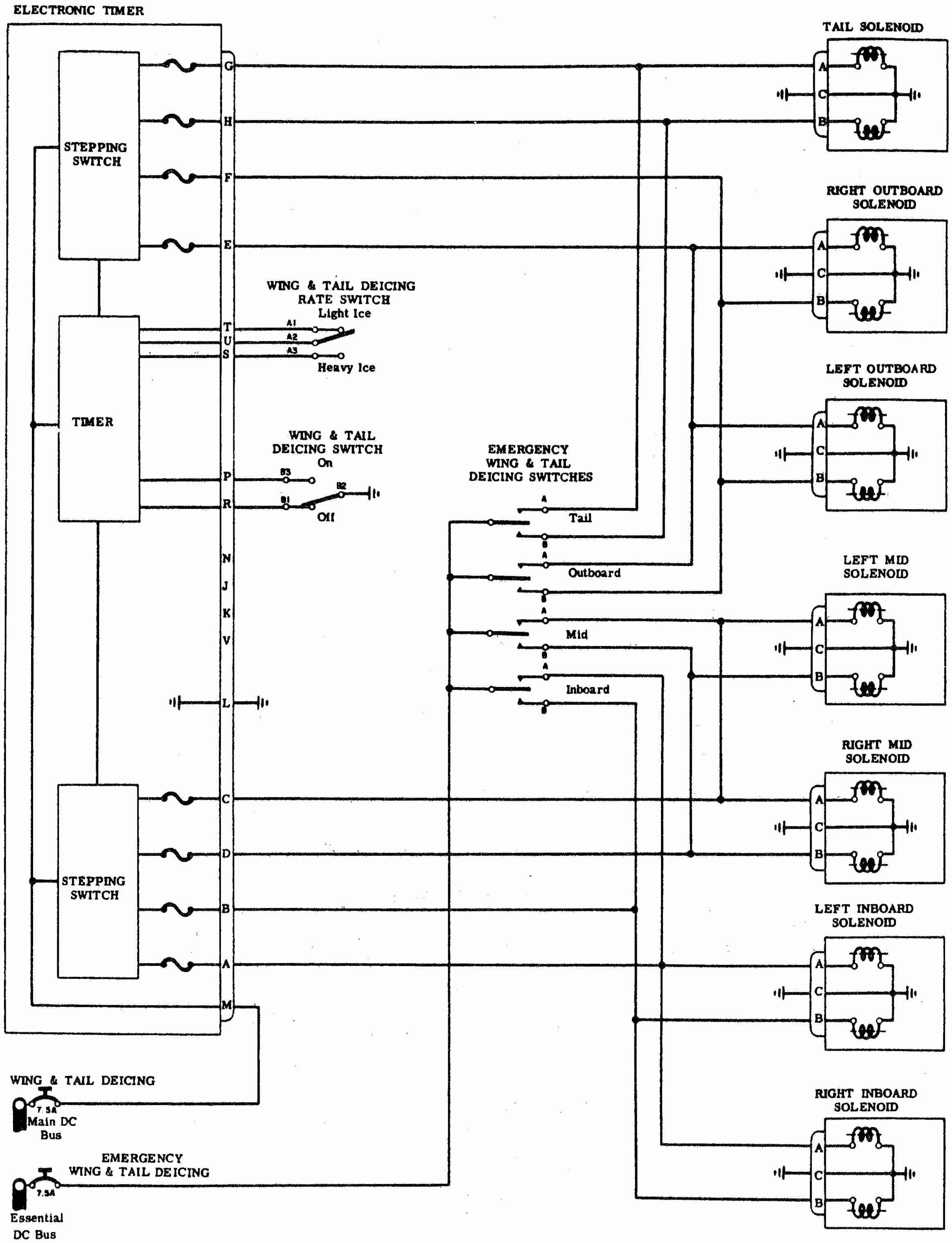
2. Operation

It is mandatory that before actuation of de-ice system, one fourth inch layer ice be allowed to build up. This will aid in preventing any bridging of ice which might render de-icer tubes inefficient. De-icer boot efficiency can also be increased by regular applications of Icex or its equivalent, to prevent excessive ice adhesion. Application of Icex or an equivalent should be accomplished every 100 hrs or when icing conditions are anticipated.

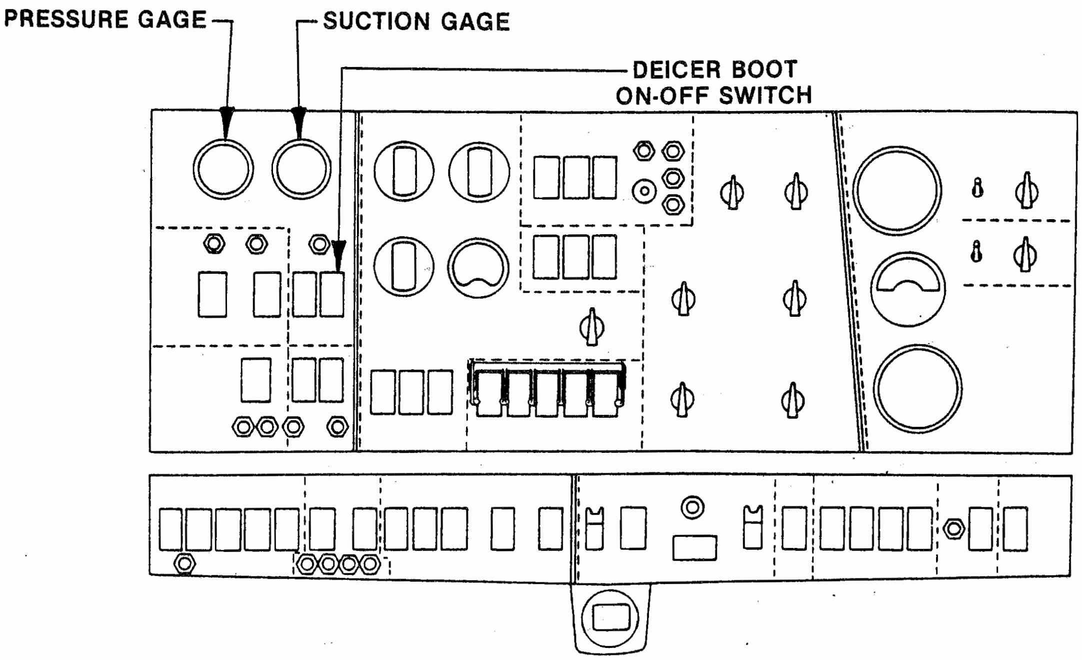


Wing and Empennage De-icing System
Figure 1.

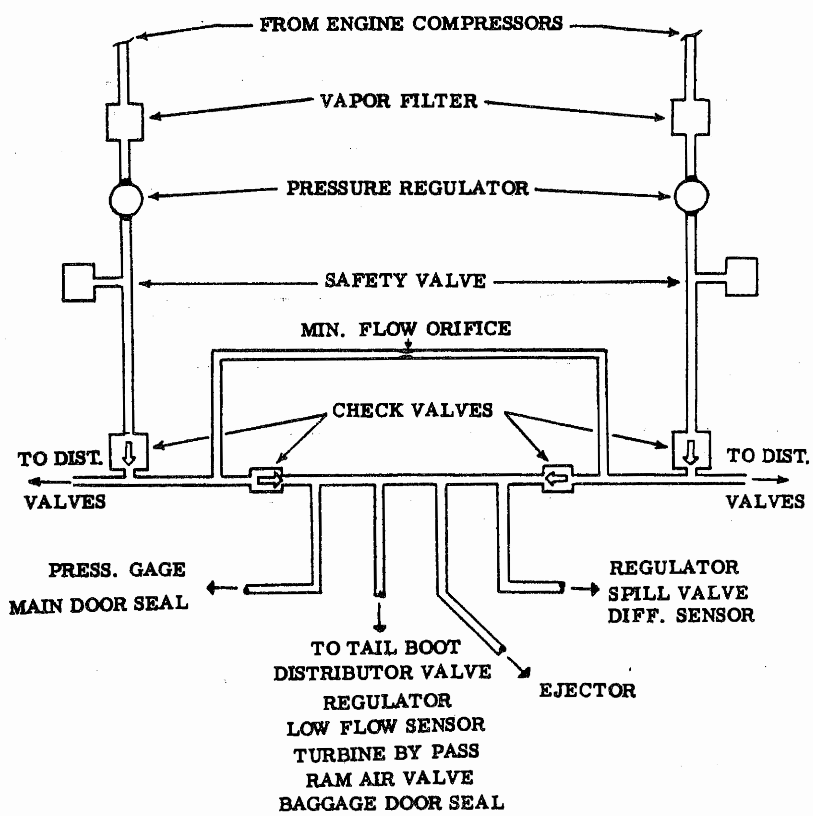
"Reference to manufacturers in this Tech. Pub. are for identification purposes only and are neither specified nor furnished as a source for obtaining such parts."



Wing and Empennage De-icing System — Schematic
Figure 2.



Wing and Empennage De-icing System — Controls and Indicators
Figure 3.



Minimum Flow Orifice — Flow Diagram
Figure 4.

WING AND EMPENNAGE DE-ICING SYSTEM — FAULT ISOLATION

FAULT	POSSIBLE CAUSE	CORRECTION
Pressure gage at normal reading, no de-icer boot inflates.	Shorted, open or grounded wiring to timer.	Locate faulty wiring and repair.
Electronic timer inoperative.	Shorted, open or grounded wiring in timer.	Locate faulty wiring and repair.
	De-icer system wiring not properly connected.	Check wiring for proper installation.
	Faulty de-icer system wiring or connections.	Check all wiring and connections for continuity. Repair or replace, as required.
	One or more timer fuses burned-out.	Remove and replace burned-out timer fuses.
Pressure gage at normal reading, one or more de-icer tubes fail to inflate.	Shorted, open or grounded wiring in timer. De-icer system wiring not properly connected.	Locate faulty wiring and repair. Check wiring for proper installation.
	Faulty de-icer system wiring or connections.	Check all wiring and connections for continuity. Repair or replace, as required.
	One or more timer fuses burned-out.	Remove and replace burned-out timer fuses.
	Faulty distributor valve solenoid.	Remove and replace distributor valve.
Timer on, pressure does not buildup in De-icer system.	Leak in lines or components.	Locate leak and repair.
	Faulty regulator.	Remove and replace regulator.
Timing cycle too short or too long.	Improper timing element resistors.	Check valves of resistors in electronic timer unit.
	Faulty component in timing circuit.	Remove and replace electronic timer.
Stepping switch fails to operate.	Defective tube.	Remove and replace tube.
	Faulty component in circuit.	Remove and replace electronic timer.

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WING AND EMPENNAGE DE-ICING SYSTEM — MAINTENANCE PRACTICES

1. **Pneumatic De-Ice Pressure Regulator Valve — Removal / Installation**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Removal

(1) Gain access to pressure regulator valve in main gear wheel wells.

(2) Disconnect pneumatic air line to valve.

(3) Remove hardware, then remove valve.

B. Installation

(1) Position valve and secure with hardware.

(2) If new or replacement regulator valve, P/N 38E20 is installed, perform Pneumatic De-ice Pressure Regulator Valve — Functional Test, see this section. If same regulator valve or P/N 38E32138E35 valve is installed proceed to next step.

(3) Connect inlet and outlet air lines.

(4) Perform Pneumatic De-ice System Operational Test, See this Section.

2. **Pneumatic De-Ice Pressure Regulator Valve — Functional Test**

(P/N 38E20)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Gain access to pressure regulator valve in main wheel well.

B. Disconnect inlet and outlet lines.

C. Connect pressure gage (0 to 50 psi) to outlet side of regulator.

D. Apply at least 35 psi air pressure to inlet side of regulator.

E. Pressure gage should Indicate 18 psig.

F. If Indication Is low adjust to 18 psig with adjustment screw on pilot valve part of regulator.

G. Reduce pressure and disconnect air supply and pressure gage.

H. Connect inlet and outlet lines to regulator.

3. **Pneumatic De-Icer Ejector — Removal / Installation**

A. Removal

(1) Remove access cover aft inboard side of right nacelle.

(2) Disconnect pressure and vacuum lines.

(3) Disconnect and remove ejector.

B. Installation

(1) Install ejector.

(2) Connect pressure and vacuum lines.

(3) Connect regulated air supply to pneumatic de-ice system manifold in nacelle.

(4) Apply 15 to 18 psi as indicated on de-ice pressure gage in cockpit.

(5) Vacuum gage in cockpit should indicate 5 to 6 inches Hg.

(6) Check all connections for leaks.

(7) Inspect area for presents of foreign objects, security of all attachments.

(8) Secure access covers.

4. Pneumatic De-ice Solenoid Distributor Valve — Removal / Installation

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Removal

(1) Gain access to solenoid distributor valves as follows:

(a) Wing, inboard distributor valve (nacelle).

(b) Wing, mid distributor valve (nacelle).

(c) Wing, outboard distributor valve (outer wing access covers).

(d) Tail distributor valve (tail compartment).

(2) De-energize electrical system.

(3) Disconnect electrical plug.

(4) Disconnect pressure and vacuum hoses.

(5) Disconnect valve from mounting.

(6) Remove valve.

B. Installation

(1) Install valve in mounting.

(2) Connect pressure and vacuum hoses.

(3) Connect electrical plug.

(4) Perform Pneumatic De-ice System — Operational Test, see this Section.

(5) Inspect area for presents of foreign objects, security of all attachments.

(6) Secure all access covers.

5. Pneumatic De-ice Distributor Valve Hoses — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Gain access to flexible de-icer hoses in main wheel well. Total of 19 hoses in left nacelle and a total of 22 hoses in right nacelle.

B. Inspect flexible hoses off distributor valves for deterioration, cracks, wear and signs of flaking of protective polysulfide coating.

NOTE: If protective coating on a hose is found to be damaged and the hose has not come into contact with fuel, oil or hydraulic fluid, application of a sealer conforming to Specification MIL-S-8802, Class B, is recommended. (Pro-Seal 890 and EC-1675 for example).

If damaged hose has come into contact with fuel. oil or hydraulic fluid. Immediate replacement of hose is recommended.

6. Nacelle/Fuselage Pneumatic De-ice Pressure Check Valve — Removal / Installation

A. Removal

(1) For fuselage check valves remove floorboards 11 and 12 (Fuselage station 232 to 288). For nacelle check valves remove nacelle fillet.

(2) Disconnect inlet and outlet lines on either side of check valve.

(3) Remove check valve.

B. Installation

(1) Install check valve with arrow pointing in direction of flow.

(2) Connect inlet and outlet lines.

(3) Perform Pneumatic De-ice System — Operational Test. See this Section.

7. **Nacelle/Fuselage Pneumatic Pressure Check Valve — Inspection**

A. Remove check valve. See procedure this section.

B. Visually inspect valve for evidence of corrosion.

C. Check for freedom of movement of flapper by gently shaking valve.

D. If visual inspection shows no defects, reinstall check valve.

8. **Tail Pneumatic De-Ice Pressure Check Valve — Removal / Installation**

A. Removal

(1) Gain access to valve in tail compartment (left side of forward part of compartment above and to left of APU enclosure).

(2) Disconnect inlet and outlet lines on either side of check valve and remove valve.

B. Installation

(1) Install check valve with arrow pointing in direction of flow (away from APU).

(2) Connect inlet and outlet lines.

(3) Start and run APU.

(4) Turn APU air on.

(5) Cycle temperature control bypass valves to full decrease.

(6) Pressure indication on pneumatic de-ice pressure should be 10 psi or more.

(7) While pressure is on check for air leaks. Shut down APU.

9. **Tail Pneumatic De-Ice Pressure Check Valve — Inspection**

A. Remove check valve, see this section.

B. Visually inspect valve for evidence of corrosion.

C. Check for freedom of movement of flapper by gently shaking valve.

D. If visual inspection shows no defects, reinstall check valve, see this section.

10. **Pneumatic De-icer Boot — Removal / Installation**

A. Removal

NOTE: For ease of removal and installation of wing de-icer boots, it is recommended that leading edges be removed and de-icer boot replacement be accomplished on the bench.

(1) Starting at one corner of upper trailing edge of de-icer boot, apply a minimum amount of solvent to soften adhesion line while tension is applied to peel back corner of boot.

(2) Using a pressure applicator filled with solvent, separate boot from wing for a distance of 4 inches along upper trailings edge.

(3) Area between boot and airfoil section which has now been separated will act as a reservoir for solvent. De-icer boot can now be pulled down toward leading edge with uniform tension.

(4) Remove remainder of boot in same manner as described above.

B. Installation

NOTE: Before installation ensure that exposed area on airfoil is thoroughly cleaned with solvent and is free of dried cement.

(1) Mask area to eliminate the need of cleaning excess adhesive.

CAUTION: DE-ICE BOOTS MAY BE DAMAGED BY MEK OR ACETONE. USE ONLY TOLUENE WHEN CLEANING AND INSTALLING THESE DE-ICE BOOTS.

DO NOT SATURATE THE BACK SURFACE TOO HEAVILY WITH SOLVENT OR SCRUB REPEATEDLY. ALLOW TO DRY THOROUGHLY BEFORE CEMENTING.

(2) Rough back surface of boot shall be cleaned using a clean cloth dampened with toluol. Cloths must be changed frequently to avoid contamination of cleaned areas. Clean entire boot in this manner not less than twice.

(3) Thoroughly mix 1300L adhesive. If necessary, adhesive may be thinned with Toluene or MEK (up to 5% by volume).

(4) Apply one even coat to back surface of boot and corresponding airfoil surface. Allow adhesive to dry a minimum of 1 hour at 50°F (10°C) or above when relative humidity is less than 75%. If relative humidity is 75% to 90%, allow additional drying time. Do not apply adhesive if relative humidity is above 90% or if temperature is below 50°F (10°C).

(5) Apply a second coat of adhesive to both surfaces and allow to dry 1 hour.

NOTE: Adhesive may be allowed to sit on surfaces up to 48 hours providing surfaces are covered and kept clean.

(6) Snap a chalk line along leading edge of the airfoil section.

(7) Pull end of inflation hose through connection hole in the airfoil surface. When there is more than one inflation hose, ensure that each hose is brought out through its proper hole to avoid improper inflation of de-ice boots.

(8) Replace any cracked or deteriorated sections of hose.

(9) Before installing de-ice boots onto airfoil, ensure that sufficient number of personnel are available to hold boot steady.

(10) Roll up de-ice boot to approximately 18 inches each side of inflation hose. Ensure that when rolling boot up that side having adhesive is out so centerline is freely visible.

(11) Position de-ice boot so centerline is directly over chalkline on leading edge of airfoil and so inflation tube connection will match the opening in the airfoil.

(12) Securely attach inflation hose to de-ice boot connection with Tinnerman clamp or suitable substitute. Tighten clamps using caution not to over tighten and damage hose or connector.

CAUTION: IF DE-ICER BOOT SHOULD ATTACH OFF COURSE (CENTERLINE ON BOOT NOT COINCIDING WITH LEADING EDGE CHALKLINE). APPLY MEK WITH SMALL BRUSH OR SQUIRT CAN TO SOFTEN THE BOND LINE. APPLY ONLY A MINIMUM OF SOLVENT TO BOND LINE WHILE APPLYING SUFFICIENT TENSION TO PEEL BACK DE-ICE BOOT. TO AVOID DAMAGE OF DE-ICE BOOT DO NOT JERK, TWIST, BEND SHARPLY OR APPLY EXCESSIVE AMOUNTS OF SOLVENT. ALLOW TO DRY THOROUGHLY BEFORE CONTINUING.

NOTE: Use self-tightening spring clamps when installing plastic tubing.

(13) Install de-ice boot ensuring centerline matches chalkline.

NOTE: Installation may best be accomplished using two people, one to hold and guide de-ice boot during installation, the other to reactivate adhesive and roll de-ice boot into place.

(14) Using a clean lint-free cloth dampened with Toluene reactivate (Tackefy) a 3 inch wide 18 inch long area of adhesive on airfoil leading edge, centered at the de-icer inflation hose connection. If connection is on or near leading edge, reactivate adhesive for about 3 inches around leading edge/air connection hole so that it may be installed first. Reactivate a matching section of adhesive on de-ice boot. If inflation hose connection is above or below leading edge, reactivate area around hole and down leading edge. De-ice boot will adhere only where adhesive is reactivated.

(15) Press de-ice boot to airfoil surface, ensuring centerlines coincide and rubber roll de-ice boot firmly against airfoil surface in tackified area.

(16) Reactivate adhesive on de-icer boot and airfoil surface along leading edge, 3 inches wide for an additional 2 to 3 feet.

(17) When adhesive becomes tacky, unroll de-ice boot (while holding a slight tension to avoid wrinkles) against airfoil leading edge. Continue this method until entire de-ice boot is installed along leading edge centerline. Roll installed area down firmly with a rubber-roller.

(18) After de-ice boot is securely attached along centerline, begin to reactivate adhesive on upper or lower sides by (beginning at inboard side) wiping with Toluene moistened cloth in one direction the entire length of boot. Return to starting place by wiping the corresponding adhesive on airfoil surface. Repeat until area is tacky.

NOTE: Too much wiping will remove adhesive.

(19) While reactivating adhesive ensure that de-ice boot is held back to reveal bondline and ensure no inadvertent contact with activated adhesive.

(20) Being careful not to trap any air under boot, roll de-ice boot down to airfoil surface.

(21) If de-ice boot is to be installed in a recess, install de-ice boot up to edge of recess and using a razor knife trim edges and/or ends to fit recessed leading edge skin, against adjacent de-ice boot or against aircraft structure.

(22) Rubber roll spanwise over entire surface of de-ice boot applying pressure to ensure a good bond. Roll trailing edges outside the inflatable area with a narrow stitcher roller.

(23) Fair in surfaces with suitable filler. Allow to dry.

(24) Mask area around boot 1/8 inch from edge of de-ice boot and 1/4 inch from edge of filler.

(25) Apply heavy coat of appropriate conductive edge sealer to surfaces between tape.

NOTE: Neoprene De-ice Boots require A56B and Estane™ De-ice Boots require 74-451-K sealer to be applied.

(26) Ensure sealer installation is continuous with no voids.

(27) Remove masking tape before sealer sets up.

(28) Using excess material trimmed from ends of boots, prepare one test specimen for each boot installed. This specimen should be a 1 X 8 inch full thickness strip of boot material cemented to a piece of sheet metal and following identical procedure used for installation. Leave 1 inch of strip uncemented in order to attach clamp. Four hours or more after specimen installation, attach a spring scale to uncemented end of each strip and measure force required to remove strip at the rate of 1 inch per minute. The pull shall be applied 180° to surface. A minimum of 5 pounds tension shall be required to remove test strip. If less than 5 pounds is required, acceptability of boot shall be based on following test:

(a) Carefully lift one corner of boot in question to attach spring clamp.

(b) Attach spring scale to clamp and pull with force 180° to surface, and in a direction that boot tends to be removed on diagonal.

(c) If a force of 5 pounds per inch of width can be exerted under these conditions installation is satisfactory.

(d) Cement corner following procedure previously outlined.

(e) Failure to meet this requirement shall result in removal and installation of boot.

(29) Supply a regulated air supply to pneumatic de-ice system pressure manifold 15-18 psig.

(30) Inflate boot which was replaced by selecting appropriate manual de-icing switch, copilots console. Check A and B tubes for proper inflation and leakage.

(31) Return system to normal.

11. Manifold Leakage Test

A. Disconnect pressure and vacuum lines at pressure and suction gages in cockpit and cap lines.

B. Disconnect line from bulkhead fitting in floor leading to outflow valves (Fuselage Station 133 right side of fuselage) and cap bulkhead fitting.

C. Disconnect line from restrictor on de-icer manifold leading to low flow sensor at Fuselage Station 620 and cap restrictor.

D. Disconnect pressure and suction supply hoses at all distributor valves and plug hoses.

E. Disconnect line at blower control pressure regulator in right nacelle and cap line (Nacelle Station 301).

F. Remove suction relief valve in right wing fillet (outboard side of right engine nacelle) and replace with a short length of flexible hose to complete connection in line.

G. Open main and baggage doors so that seals will not be inflated.

H. Disconnect pressure and suction hoses at ejector in right nacelle and plug hoses.

I. Connect external air supply to pressure manifold at any wing distributor valve.

NOTE: External air supply line must incorporate a pressure regulator, shutoff valve and pressure gage.

J. Apply 36 psig to pressurize de-icer manifold and close shutoff valve in external air supply line.

K. Observe pressure gage in air supply line and check elapsed time for pressure to drop from 30 to 20 psig. If time is less than 100 seconds, check line and repair leaks.

L. Disconnect external air supply line from pressure manifold.

M. Connect external air supply to suction manifold at any wing distributor valve.

N. Apply 36 psig to pressurize de-icer suction manifold and close shutoff valve in external air supply line.

O. Observe pressure gage in air supply line. Pressure should not drop more than 4 psig in 5 minutes. If leakage is excessive, check lines and repair leaks.

P. Restore de-icer system to normal configuration.

12. Pneumatic De-ice System — Operational Test

A. Create 15-18 psig pressure and 5-6 inches Hg suction in pneumatic de-icing manifolds by supplying pressurized air to the inlet fitting on the vapor filter in either nacelle.

B. Apply external 28V dc power and energize main and essential dc bus.

C. Depress WING/TAIL DE-ICE circuit breakers.

D. Place rate switch to HEAVY ICE.

E. Place ON/OFF switch to ON.

F. After approximately 15 seconds, boots should inflate in sequence as follows: (Allow boots to cycle 3 times before timing operation)

BOOT SECTION	PORT	TIME INFLATED
Wing - Inboard	A	4 sec ± 10%
	B	4 sec ± 10%
Wing - Mid	A	5 sec ± 10%
	B	5 sec ± 10%
Wing-Outboard	A	5 sec ± 10%
	B	5 sec ± 10%
Fin and Right Stabilizer	A	5 sec ± 10%
Fin and Left Stabilizer	B	5 sec ± 10%
		Total 38 sec ± 10%

BOOT SECTION	PORT	TIME INFLATED
		Dwell Period 22 sec ± 10%
		Cycle Time 60 sec ± 10%

NOTE: When ON-OFF switch is turned to OFF during either inflation or dwell period and then turned to ON after a short delay, boot inflation should begin in approximately 15 seconds and should follow above sequence.

G. Turn ON/OFF switch to OFF.

H. Place rate switch to LIGHT ICE.

I. Turn ON/OFF switch to ON.

J. After approximately 15 seconds, boots should inflate in sequence as follows:

BOOT SECTION	PORT	TIME INFLATED
Wing-Inboard	A	4 sec ± 10%
	B	4 sec ± 10%
Wing-Mid	A	5 sec ± 10%
	B	5 sec ± 10%
Wing-Outboard	A	5 sec ± 10%
	B	5 sec ± 10%
Fin and Right Stabilizer	A	5 sec ± 10%
Fin and Left Stabilizer	B	5 sec ± 10%
		Total 38 sec ± 10%
		Dwell Period 202 sec ± 15%
		Cycle Time 240 sec ± 15%

K. When ON/OFF switch is turned to OFF during either inflation or dwell period and then turned to ON after a short delay, boot inflation should begin in approximately 15 seconds and should follow above sequence.

L. With ON/OFF switch turned to OFF, inflate boots manually by means of wing and tail de-ice emergency switches in copilots side console.

M. Using wing and tail de-ice emergency switches, operate A and B ports of tail boots and observe operation of the tail boots indicator light.

N. Repeat Steps A - E to ensure either engine is capable of supplying air to de-ice system.

O. Restore right nacelle to normal configuration.

13. De-icer Repair Procedure — Cold Patch Method

A. Materials Required

The B.F. Goodrich repair kit number 74-451-C, for neoprene surface de-icers, contains the majority of the necessary parts required for a de-icer boot repair. The materials contained in repair kit number 74A51-C are as follows:

PART NO.	QUANTITY	DESCRIPTION
74-451-11	1/2 pt.	A-56-B Conductive Cement
74-451-16	30	1 1/4 in. x 2 1/2 in. Neoprene Patch

PART NO.	QUANTITY	DESCRIPTION
74-451-17	30	2 1/2 in. x 5 in. Neoprene Patch
74-451-18	10	5 in. x 10 in. Neoprene Patch
74-451-19	3	5 in. x 19 In. Neoprene Fabric Patch
74-451-20	1/2 pt.	No. 4 Air Curing Cement
74-451-70	2	1/2 in. Flat Brush
74-451-73	1	Steel Roller Stitcher 1/8 in. x 1-1/8 in. dia.
74-451-75	6	Buffer Stick
74-451-87	1	Buffer Shield

B. In addition, other materials required for repair are:

- Minnesota Mining and Mfg. Co., EC1300L Cement
- Toluol or equivalent
- Clean, lint free cleaning cloths
- 2 inch rubber hand rollers
- 1 inch masking tape
- Measuring tape
- Razor knives
- Small brushes
- Steel wool pads
- Hypodermic needles

NOTE: MEK (Methyl Ethyl Ketone) can be used instead of Toluol, MEK will cause very rapid drying.

C. Repair

WARNING: ADHESIVE AND SOLVENT VAPORS ARE TOXIC AND EXTREMELY FLAMMABLE. USE ONLY IN A WELL VENTILATED AREA AWAY FROM ALL SPARKS AND FLAMES. AVOID PROLONGED BREATHING OF FUMES AND CONTACT WITH SKIN OR EYES.

(1) Cuts, tears or ruptures to tube area shall be repaired with fabric reinforced neoprene patches.

(2) Area around damage shall be cleaned with a cloth dampened with solvent.

(3) A patch of ample size to cover damage and to extend at least 5/8 inch beyond edges of cut or tear should be selected. If none of the patches are of proper size, one may be cut to desired size from one of the larger molded patches and the edges should be beveled by cutting at an angle.

NOTE: Patches are manufactured to stretch in one direction only. Cut and apply patch selected so that stretch is in widthwise direction of inflatable tubes.

(4) Area around damage shall be buffed with steel wool, so surface is moderately roughened.

(5) Buffed area shall be wiped with clean cloth dampened in solvent.

(6) Remove backing from selected patch and apply one even coat of cement to patch and corresponding damaged area of de-icer.

(7) To keep patch flat while applying cement, apply masking tape to breeze surface of patch.

(8) Cement shall be allowed to dry completely, (5 to 10 minutes depending on drying conditions). Remove masking tape from patch.

(9) Apply patch to de-icer with stretch in the widthwise direction of inflatable tubes, sticking edge of patch in place, working remainder down with slight pulling action so that damage is covered. Do not trap air between patch and de-icer surface.

(10) Roll down patch with stitcher-roller to ensure good adhesion.

(11) Allow repair to set 15 minutes. Wipe patch and surrounding area with solvent to remove excess cement. Wipe from center of the patch outward.

(12) Apply one light coat of A-56-B cement to restore conductivity.

(13) Allow repair to dry a minimum of 4 hours before inflating de-icer.

14. De-ice Vapor Filter — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. In main wheel well, remove inlet, outlet, and drain lines from filter body located on vapor filter.

B. Loosen mounting clamp and remove filter assembly.

C. To disassemble filter halves, remove V-band clamp around filter assembly.

D. Remove filter element.

E. Inspect filter element and replace as required. Use new seal as required.

F. Reassemble filter body and tighten clamp.

G. Install filter assembly in mounting clamp.

H. Connect inlet, outlet, and drain lines.

15. De-ice Vapor Filter Assembly — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL. ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) In main wheel well, remove inlet, outlet, and drain lines from filter body.

(2) Loosen mounting clamp and remove filter assembly.

B. Installation

(1) Inspect filter body for condition.

(2) Install filter assembly in mounting clamp.

(3) Connect inlet, outlet, and drain lines.

16. De-ice Boot Protective Coating — Preparation / Application

A. Materials Required

NOTE: 74-451-L De-icer Protective Coating kit should not be applied over other commercial protective coatings except at installers risk. Recoating of a de-ice boot previously coated with a 74-451-L kit is not recommended without removing old application.

The materials contained in BF Goodrich De-icer Protective Coating Repair kit 74-451-L are as follows:

PART NO.	QUANTITY	DESCRIPTION
74-451-122 (021045)	1 qt.	Primer
74-451-123 (0165CP-9)	1 qt.	Protective Coating
74-451-120 (KE-7005)	4 oz.	Accelerator
75-23-062		Instructions

In addition, other materials required for repair are:

- Cleaning cloth (lint free)
- Cleaning solvent (MEK)
- Cleaning agent (laundry detergent)
- 2 inch paint brush

- Scotch Brite Abrasive, very fine Type A
- 1/2 inch masking tape
- Nylon scrub brush
- Rubber gloves

B. Procedure

WARNING: USE ONLY IN WELL VENTILATED AREA. AVOID PROLONGED BREATHING OF FUMES AND SOLVENT VAPOR. AVOID CONTACT WITH SKIN AND EYES.

(1) Wash de-icing surface to be coated thoroughly in hot soapy water using nylon scrub brush.

(2) Lightly sand surface with Scotch Brite fine grade, to ensure no silicone based materials remain imbedded in de-ice boot.

(3) Clean and dry surface with clear water and clean rags. Ensure to remove all detergent and loose particles from sanding.

(4) Wipe surface thoroughly with lint free rag and MEK, turning rag often to remove all contamination.

(5) Mask off de-ice boot.

(6) Apply one even coat of mixed primer to de-icing boot surface. Use short strokes in same direction to ensure smooth finish.

NOTE: For greatest durability, brush chordwise over spanwise de-icer tubes or spanwise over chordwise de-icer tubes.

Primer and Protective Coatings may be thinned with Methyl Isobutyl Ketone (MIBK) solvent to achieve a smoother brushing consistency. Never thin more than 15% by volume.

(7) Allow 30 minutes minimum for primer to cure.

(8) Apply one smooth coat of mixed protective coating over entire primed area. A second coat may also be applied within 15 minutes and 1 hour following the first coat of protective coating.

NOTE: Manufacturer suggests removing masking tape each 3 feet of final protection coating application.

(9) Do not fly aircraft for a minimum of 4 hours following this procedure to ensure curing time.

WINDSHIELD WIPER SYSTEM — DESCRIPTION / OPERATION

1. **Description**

NOTE: Three types of windshield wiper systems are used in the Gulfstream I aircraft. All three systems are covered separately in this Chapter.

A. Windshield wiper system controls (Aircraft 1 - 148, 322 and 323)

The windshield wiper system is hydraulically operated and manually controlled. The system is made up of a speed control valve, a control unit, a three-position selector valve and two windshield wiper units. The selector valve control handle is in the cockpit, at the forward right side of the center pedestal console. The speed control knob is at the left side of the center pedestal console. The speed control valve, control unit, and selector valve are in the nose wheel well, beside the nose gear uplock bracket. The windshield wiper units are below the pilot and copilot windshields.

B. Windshield wiper system controls (Aircraft 149 - 200)

The windshield wiper system is hydraulically operated and manually controlled. The system consists of a speed control valve, a control unit and two windshield wiper units.

The speed control knob is at the left forward side of the center pedestal console. The speed control valve and the control unit are located in the nose wheel well beside the nose gear uplock bracket. The windshield wiper units are located below the pilot and copilot windshields just forward of the flight instrument panels.

2. **Operation**

(See Figure 1

A. Windshield wiper system equipment

Main hydraulic system pressure (1500 psi) is fed to the speed control valve, which is manually operated and can be adjusted from the fully closed to the fully open position with the control knob on the pedestal. From the speed control valve, metered fluid flows to the control unit, in which spring loaded slide valves change the direction of flow to operate the windshield units. On Aircraft 1 - 86 and 114 not having ASC 120A and Aircraft 87 - 148, 322 and 323 having ASC 12A.

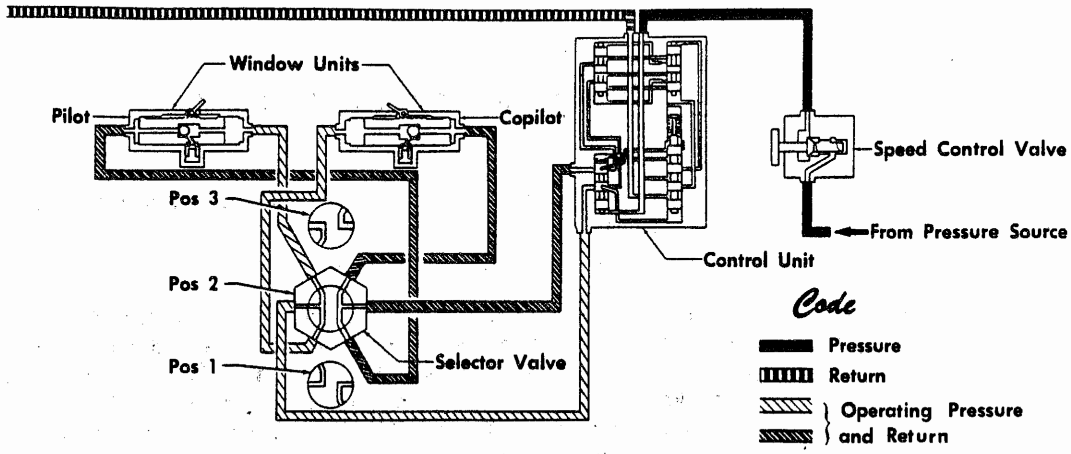
The fluid leaves the control unit and is fed to a three-position selector valve which can be set for pilot, copilot, or both. The No. 1 position feeds the pilots windshield unit; the No. 2 position feeds both windshield units and the No. 3 position feeds the copilots windshield unit. The windshield wiper units, which are piston actuators, receive fluid from the selector valve to drive the wiper arms.

On Aircraft 149 - 200 the fluid leaves the control unit and is fed to the two windshield wiper units, which are simple piston actuators, to drive the wiper arms.

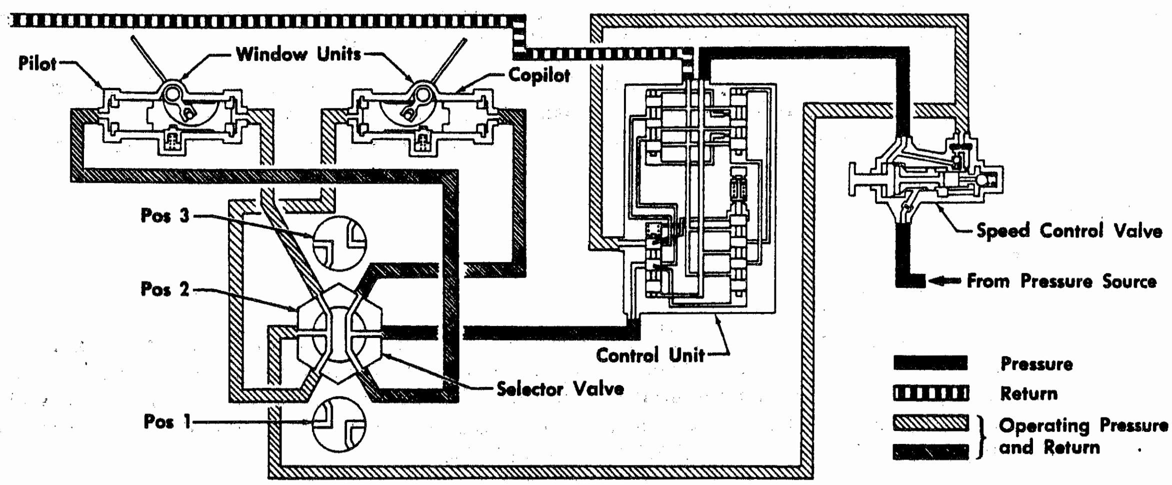
NOTE: On Aircraft 2 - 148, 114, 322 and 323 the speed control valve must be fully closed before selecting the left or right position.

On Aircraft 1 - 86 and 114 the normal park position for the wiper blades is in the inboard position.

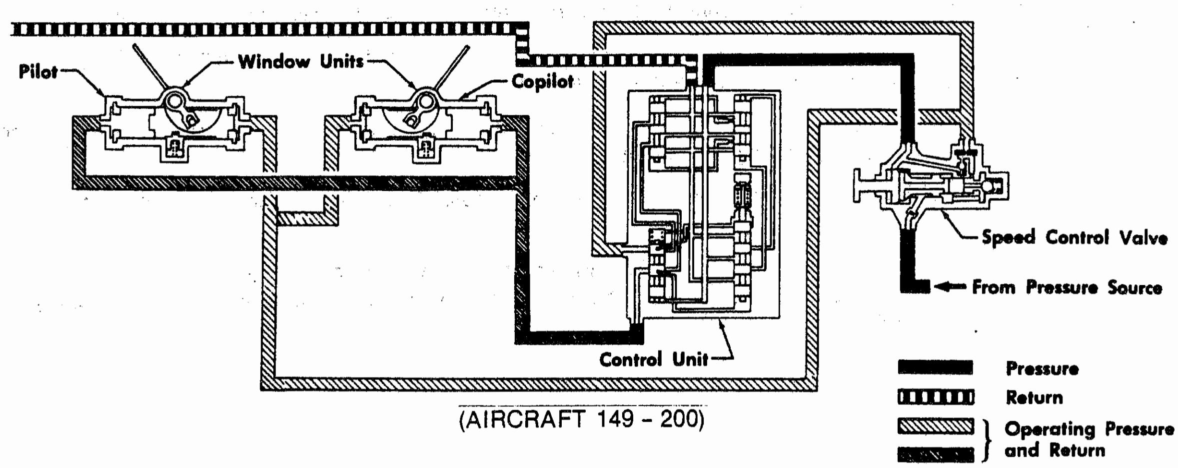
On Aircraft 87 - 200, 322 and 323 and Aircraft 1 - 86 having ASC 120A the windshield wiper arms will automatically park in the outboard position keeping the blades within the heated section of the windshields.



(AIRCRAFT 1 - 86 INCLUDING 114 NOT HAVING ASC 120A)



(AIRCRAFT 1 - 148 INCLUDING 114 , 322, 323 HAVING ASC 120A)



(AIRCRAFT 149 - 200)

Windshield Wiper Hydraulic Flow — Schematic
Figure 1.

WINDSHIELD WIPER SYSTEM — MAINTENANCE PRACTICES

1. **Windshield Wiper Speed Control Valve — Removal / Installation**

   A. Removal

   WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

   (1) Gain access to speed control valve in nose wheel well and disconnect hydraulic lines at valve.

   (2) In cockpit remove cotter pins that connect shaft and shaft extension to valve. Then rotate shaft to screw out of valve stem.

   (3) Remove screws and stat-o-seals that secure valve to mounting and remove valve.

   B. Installation

   (1) Install new O-ring in inner groove of mounting plate.

   (2) Install valve using new stat-o-seals and secure with screws.

   (3) Rotate shaft into valve stem guide. Engage shaft and shaft extension and secure with cotter pins.

   (4) Install hydraulic lines.

   (5) Service hydraulic reservoir.

2. **Windshield Wiper Motor — Removal / Installation**

   A. Removal

   (1) Dissipate hydraulic system pressure and loosen reservoir cap.

   (2) Disconnect windshield wiper arms outside of aircraft. Loosen nut on wiper blade to reduce pressure on blade.

   (3) Remove safety wire and bolt, pull wiper arm off shaft.

   (4) Remove copilots and pilots instrument panel to gain access to wiper motor.

   (5) Disconnect hydraulic lines and remove screws and washers that mount unit to structure and remove motor.

   B. Installation

   (1) Install new O-ring in shaft seal mounting plate.

   (2) Install motor on structure with screws, washers and connect hydraulic lines.

   CAUTION: ENSURE NOT TO DAMAGE O-RING ON SEAL MOUNTING PLATE WHEN INSERTING MOTOR SHAFT.

   (3) Check wiper motor for leaks. Also operate wiper motor. Observe that it stops in position required for this aircraft effectivity. (See Windshield Wiper System — Operational Test, see this Section)

   (4) Install wiper arm in its normally stopped position.

   (5) Tighten adjustment nut until nine pounds barely lifts the wiper off glass.

   (6) Spray oscillating shaft with anticorrosive solution.

   (7) Install instrument panel.

3. **Windshield Wiper Arm, Link and Blade — Removal / Installation**

A. Removal

(1) Disconnect windshield wiper arm. Loosen nut on wiper blade to reduce pressure on blade.

(2) Remove safety wire and bolt, pull wiper arm off shaft.

(3) Disconnect and remove wiper link and wiper blade from arm as required.

B. Installation

(1) Reassemble wiper link and wiper blade to arm as required.

(2) Install wiper arm in its normally stopped position.

(3) Adjust wiper arm to set pressure of 9 + 0, − 1 pounds against windshield.

4. **Windshield Wiper Control Unit — Removal / Installation**

A. Removal

(1) Operate wing flaps to dissipate remaining hydraulic system pressure.

(2) Slowly loosen reservoir emergency filler cap to dissipate remaining reservoir pressure.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(3) Gain access to control unit through nose wheel well and disconnect hydraulic lines.

(4) Remove bolts, washers and stat-o-seals, then remove unit.

(5) Cap or plug all hydraulic lines and fittings.

B. Installation

(1) Install control unit on structure. Use stat-o-seals between unit and structure.

(2) Install hydraulic lines.

(3) Service hydraulic reservoir.

5. **Windshield Wiper Rotary Selector Valve — Removal / Installation**

A. Removal

(1) Remove right side cover from pedestal.

(2) Remove hardware from sleeve which connects tube to fitting on valve stem and remove fitting.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: Observe position of selector knob on pedestal and position of fitting on valve before removal to facilitate installation.

(3) In nose wheel well disconnect and remove hydraulic lines.

(4) With man in wheel well supporting valve remove screws from valve in cockpit. Discard stat-o-seals.

(5) Remove valve.

NOTE: Record position of valve shaft with reference to housing to facilitate installation.

B. Installation

(1) Install new O-ring in mounting plate.

(2) Position shaft of replacement valve to same position noted during removal. Install valve using screws and stat-o-seal washers.

(3) Install fitting on valve shaft, then sleeve to tube. Secure with hardware.

NOTE: Ensure that fitting and control knob are in same position noted during removal.

(4) Install hydraulic lines.

(5) Install pedestal cover.

6. Windshield Wiper — Tension

A. Using a pull type spring scale, hook scale under outside edge of wiper arm clip.

B. Tighten adjustment nut until 9 pounds barely lifts the wiper off glass.

7. Oscillating Shaft — Application of Anticorrosive Solution

A. The shaft of the windshield wiper unit should be sprayed with an anticorrosive per MIL-C-2341 1 to keep the oscillating shaft free within the sleeve of the windshield unit. Treatment of these shafts should be applied periodically in accordance with the Chapter 5.

B. The following anticorrosive solutions, which conform to MIL-C-23411, may be used: LPS #1, LPS #2, LPS #3, CRC3-36, and CRC2-36. LPS #3 is the preferred solution.

NOTE: These materials are available at local sources within the United States, or they may be obtained from:

LPS Research Laboratories, Inc.
2050 Cotner Avenue
Los Angeles, California 90025
(for LPS #1, LPS #2, and LPS #3)

CRC Chemicals
Limekiln Pike
Dresher, Pennsylvania 19025
(for CRC3-36 and CRC2-36)

8. Windshield Wiper System — Operational Test

NOTE: Two separate procedures are given for checking operation of windshield wiper system. Aircraft effectivities are given with procedure to be used.

A. Perform following on Aircraft 1-148, 322 and 323:

(1) Apply hydraulic power.

(2) Thoroughly wet windshield or place a nonabrasive sheet of paper under wiper blades.

CAUTION: DO NOT ALLOW WINDSHIELD TO DRY WHILE OPERATING WIPERS, OR PERMIT PAPER (IF USED) TO BE THROWN FROM BLADES.

(3) Ensure speed control knob is off (turned fully clockwise).

(4) Using rotary control valve selector, rotate knob to No. 1.

(5) Rotate speed control valve from OFF to ON and pilots wiper should operate, regulate valve through all speed ranges, then shut off.

(6) Ensure that wiper blade stops in following positions:

(a) Inboard position on Aircraft 1 - 86 and 114, not having ASC 120.

(b) Outboard position (in heated area of windshield) on Aircraft 87 - 148, 322 and 323, and Aircraft 1 - 86 and 114, having ASC 120.

(7) Rotate control valve selector to No. 2.

(8) Rotate speed control valve from OFF to ON and both wipers should operate, regulate valve through all speed ranges, then shut off.

(9) Ensure that both wiper blades stop in positions required by aircraft effectivity (See Step 6 above).

(10) Rotate control valve selector to No. 3.

(11) Open speed control and observe that only copilots wiper operates, shut off speed control valve.

(12) Ensure that wiper blade stops in position required by aircraft effectivity (See Step 6 above).

(13) Remove hydraulic power from aircraft.

B. Perform following on Aircraft 149 - 200.

(1) Apply hydraulic power.

(2) Thoroughly wet windshield or place a nonabrasive sheet of paper under wiper blades.

CAUTION: DO NOT ALLOW WINDSHIELDS TO DRY WHILE OPERATING WIPERS, OR PERMIT PAPER (IF USED) TO BE THROWN FROM BLADES.

(3) Rotate speed control valve from OFF to ON through all speed ranges, wipers should operates smoothly through all ranges.

(4) Rotate speed control valve to OFF; both wipers should stop in outer position in heated area of windshield.

(5) Remove hydraulic power from aircraft.

9. **Windshield Wiper Motor — Service / Inspection**

A. Inspect windshield wiper motor actuator shaft for condition and security.

NOTE: Following anticorrosive solutions may be used: LPS # 1, LPS # 2, LPS # 3, CRC3-36 and CRC2-36.

B. Spray oscillating shaft of windshield wiper motor with an anti-corrosive solution to keep shaft free within sleeve of unit.

WINDSHIELD / DV WINDOW DE-ICING SYSTEM — DESCRIPTION / OPERATION

1. Description

(See Figure 1).

A. The windshield de-icing system is designed to accomplish automatic de-icing or defogging to cockpit windows. Once activated, the system will automatically maintain glass temperature.

B. For the windshield to be fully birdproof, the glass must be maintained at or above a specific temperature. This temperature is met or exceeded when the heating system is active in either de-ice all or defog modes. To ensure the ability to produce this heat, this system is so designed that no single failure can cause total heat loss. It is a function of the operating crew to activate the system when required, and to observe the indicating system which denotes normal or abnormal operation.

C. Power for window heat is three phase, 212 volt, variable frequency, supplied by engine driven alternators. DC control power is supplied from the main DC bus. Control circuit bridge power is supplied from the 115 volt, 400 cycle, inverter system main AC bus.

D. Each window pane, except side windows, is powered by an individual transformer-rectifier power unit. The purpose of these units is to develop power of proper voltage to match the window characteristics, while at the same time drawing balanced three phase power from the alternators. These units are controlled by means of an internal power relay.

E. Application of heat to side cockpit windows is considered as a customer request option. Equipment required to produce and control this heat is not part of the standard aircraft. However, the side window are those capable of being heated. Wiring is not installed, however mounting facilities are or installation of a controller and power unit. Side windows are supplied from a single power unit. The amount of power involved is quite small, since these windows are only heated to the defog level. Both side windows are fitted with control elements, but only the left is connected to a controller unit.

2. Cockpit Windshield / DV Heat Controls

(See Figure 2).

A. Two switches, mounted in the lower overhead panel in the cockpit, control this system. One of the switches has positions DE-ICE/OFF/DEFOG. This allows the crew to deactivate the system, or to select low glass temperature or high glass temperature. The other switch has two positions; NORM/EMER. This permits selection of the alternator which powers the front windows. When set to NORM, the left alternator powers the two front windows, while the right alternator powers the left and right direct vision windows and the side windows. When the switch is set to EMER, the right alternator supplies power to the front windows, and the direct vision and side windows are cut off.

B. A sensor thermostat, imbedded in each window, is part of a temperature control system, which includes the sensor, a device for changing cycling temperatures from DE-ICE to DEFOG when selected, and the controller. Each window has its own separate temperature control system, except the side windows, who depend on one single sensor in the left window and a single controller to supply heat control to both side windows. The sensor takes continuous glass temperature information, which the controller interprets. If the glass gets too cold, the sensor signals the controller. The controller amplifies this signal and signals the power unit for that window to give the window heat. The power unit relay closes and power flows through the window, producing the heat. When the sensor is satisfied, it signals the controller and the controller cuts out the power unit. This takes place continuously, and is independent for each window. The crew has to make sure that the alternators are on the line and that the system heat range selection, DE-ICE or DEFOG, is made.

C. Windshield / DV Heat Indication Lights (See Figure 3).

To provide the crew with information that the system is functioning properly, four green indicator lights are installed just below the windshield heat control switches on the left side of the lower overhead panel. These are absolute indicators that the windows are receiving power and therefore heat.

NOTE: The side windows have no indicator lights. Each light reports for one window pane, left direct vision, left front, right front and right direct vision. Presence of a light shows that power is flowing at that

instant to that particular window. Depending on operating conditions, these indicators will pulse on and off at varying rates. Each indicator light can be mechanically dimmed. All respond when the warning lights test switch is actuated.

3. Major Components and Locations

A. The following units are mounted under the cabin floor between Fuselage Station 133 and Fuselage Station 169 (under FLR No. 2).

- Controllers are for the two main front panels, two direct vision panels and one (optional) serves both side windows.
- Power units (transformer-rectifier) supply power for right front panel and left front panel.
- Power units (transformer-rectifier) supply for right direct vision panel and left direct vision panel.
- Power unit (transformer-rectifier) supplies power for side panels. (optional)
- De-ice switch box causes resistance to be placed in the controller circuit in defogging conditions.

B. The following units are located in the shock mounted windshield heat relay box beneath the cabin floor between Fuselage Station 169 and Fuselage Station 193 (under FLR No. 2).

- Power sensing relays are sensitive units with a coil resistance of 50 ohms. Pick up is at 10 milliamperes, drop out is 4 milliamperes, coils can dissipate 1 1/2 watts. Contacts will handle 1.0 amperes.
- Power sensing resistors are rated at 0.4 ohms, 20 watts.

C. The following units are mounted on the windshield heat junction box, under the cabin floor between Fuselage Station 169 and Fuselage Station 193 (under FLR No. 4)

- Windshield heat indicator light test relay in the warning lights test system, is used to test the windshield power indicator lights.
- Left and right de-ice relays actuate when the DE-ICE/DEFOG switch is placed in the de-ice position. Controls system 26V dc and 115 volts 400 cycle power for this mode.
- Left and right defog relays actuate when the DE-ICE/DEFOG switch is placed in the DEFOG position. Controls system 28V dc and 115 volts 400 cycle power for this mode.
- Direct vision cutout relay cuts out the left and right direct vision windows when the NORM/EMER switch is placed in the EMER position.

D. Windshield power transfer relay, is a double coil, three pole, 25 ampere mechanically interlocked contactor is arranged so both sets of contacts cannot close simultaneously. This unit is mounted within the inverter relay box.

4. Operation

(See Figure 4).

A. When the control switch is set to NORM, the left alternator powers the two front windows, while the right alternator powers the left and right direct vision windows, and the side windows. With the switch set to EMER, the right alternator is connected to all the windows, but the direct vision cutout relay prevents the direct vision and side window controllers from signaling for heat. This position of the switch is intended for use only when some type of failure prevents the left alternator from supplying power. Actual switching of three phase power is done by the windshield heat power transfer relay, which is a double armature, three pole contactor of the type generally used in motor reversing applications. (See Figure 5). A mechanical interlock prevents both sets of contacts from closing simultaneously, a necessary precaution since the two alternators are not paralleled.

B. A sensor element imbedded in the glass pane is a part of a bridge circuit in controller unit. (See Figure 6). Unbalances of this bridge are measured by an amplifier circuit and cause actuation of a control relay which either signals for glass heat, or does not. The set points are as follows:

DE-ICING - Power OFF at 113°F ±5°F.
Power ON at 8°F ±2°F below OFF point.

DE-FOGGING - Power OFF at 100°F ±5°F.
Power ON at 6°F ±2°F below OFF point.

The following is an example of when electrical power for de-icing and defogging would be applied and turned off:

DE-ICING	DEFOGGING
Power OFF 113°F ± 5°F	Power OFF 100°F ± 5°F
Power ON 8°F ±2°F below OFF point	Power ON 8°F ± 2°F below OFF point
Power OFF no later than 118°F	Power OFF no later than 105°F
Power OFF no sooner than 108°F	Power OFF no sooner than 95°F
Power ON no sooner than 112°F	Power ON no sooner than 99°F
Power ON no later than 108°F	Power ON no later than 95°F

The de-icing switch box is a sealed switching device consisting of a multipole relay and resistors. Sensor information from all cockpit windows is routed through this unit. With the switching unit coil energized, the sensor data is passed directly to the bridge, causing the glass temperature to be regulated nominally between the temperatures specified above for the de-ice condition. If the switching unit coil is relaxed, resistance is introduced into the sensor circuit so the glass temperature is regulated nominally between the temperatures specified for the defog range. In the case of the side windows, if heated, the resistance in the switching unit is always inserted in the sensor circuit, giving the defog heat range at all times the system is active. Each direct vision and front window has its own controller unit, operating its own power unit.

(1) For each of the four panes which have an indicator, a resistor is inserted in the negative leg of the power feed. Note that no part of the power feed circuit is grounded and this resistor develops a small voltage drop across itself. This is sensed by a power detecting relay which controls operation of the indicator light.

(2) All electrically heated windshield panels are inoperative in emergency.

(3) Specific panel data.

(a) Front windows (per window)

(1) Area heated - 376 square inch

(2) Minimum heat density - 4.76 watts/square inch

(b) Direct vision windows (per window)

(1) Area heated - 115 square inch

(2) Minimum heat density - 4.50 watts/square inch

(c) Side windows (per window)

(1) Area heated - 249 square inch

(2) Minimum heat density - 0.6 watts/square inch

5. **Operational Practices**

A. The windshield heat must be on at all times during flight in order to make the windows structurally strong to prevent bird penetration. DE-ICE and DEFOG settings are both designed to make the window birdproof. DEFOG should be used at all times unless imminent icing conditions prevail, then select DE-ICE before entering icing conditions if possible.

B. The following conditions should be checked for during use of the system:

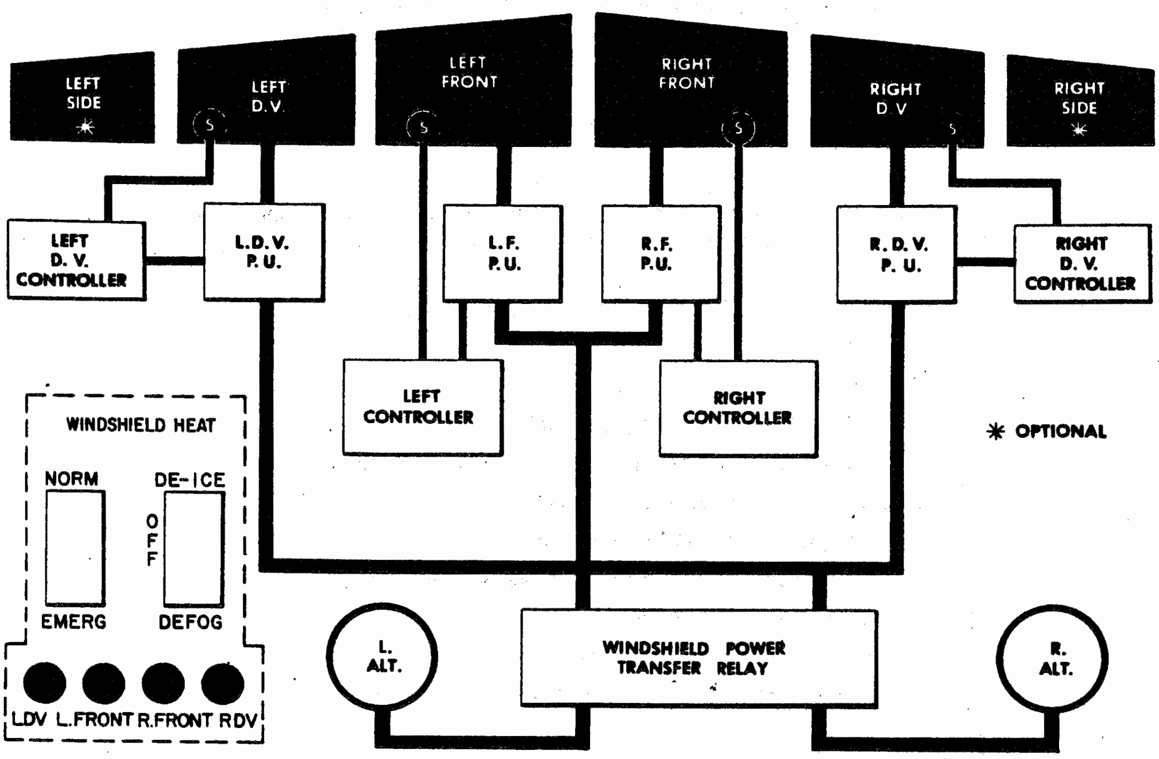
(1) Underheating, as denoted by a cool feeling of the glass and no indicator lights. If this exists, it will allow icing, fogging and possible bird penetration.

(2) Overheating, as denoted by abnormally long or continuously lighted indicator light with others cycling normally. This will, over a period of time, cause the adhesive layer to dry out. Glass will no longer be shatterproof. Dried out material is distinctly yellow. Normal glass has a faint iridescent appearance.

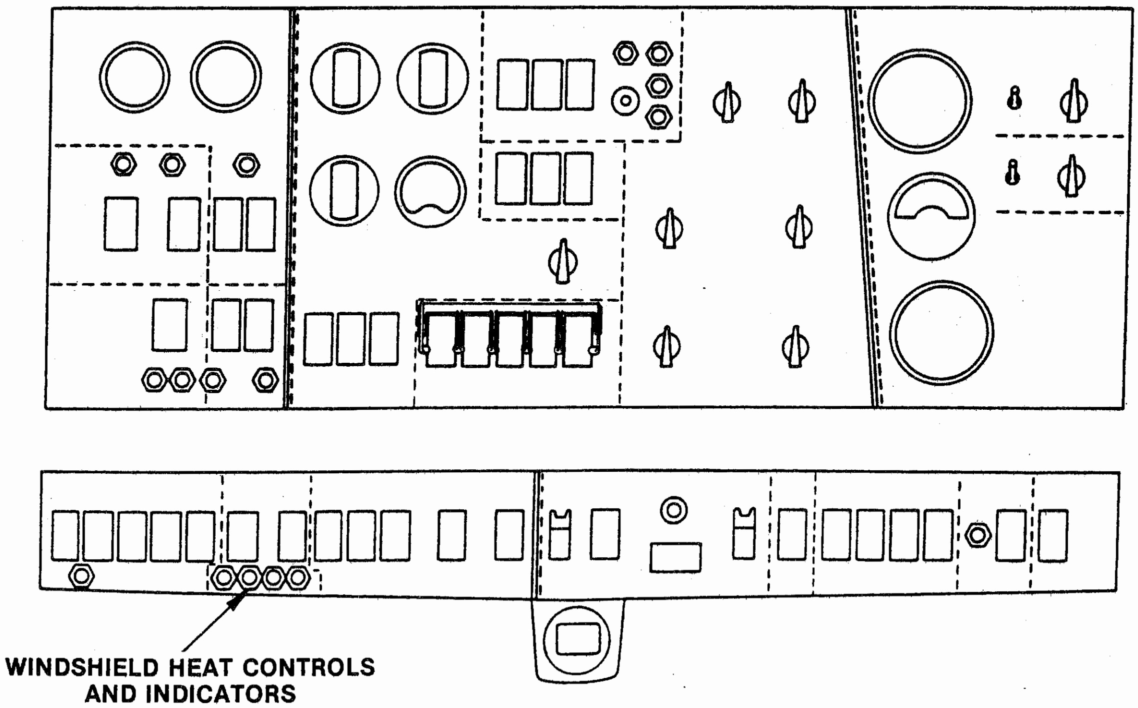
C. In the case of trouble with any window, each front and each direct vision window can be completely isolated from the system by pulling three windshield power unit circuit breakers located on the aisle side forward end of the inverter relay panel.

CAUTION: ENSURE AND PULL ALL THREE BREAKERS FOR THAT ONE WINDOW. SHUT OFF WINDSHIELD HEAT BY PLACING SWITCH TO OFF BEFORE PULLING BREAKERS. IF NOT, PHASE UNBALANCE MAY RESULT AND THE ALTERNATOR EFFECTED MAY DROP OFF THE LINE.

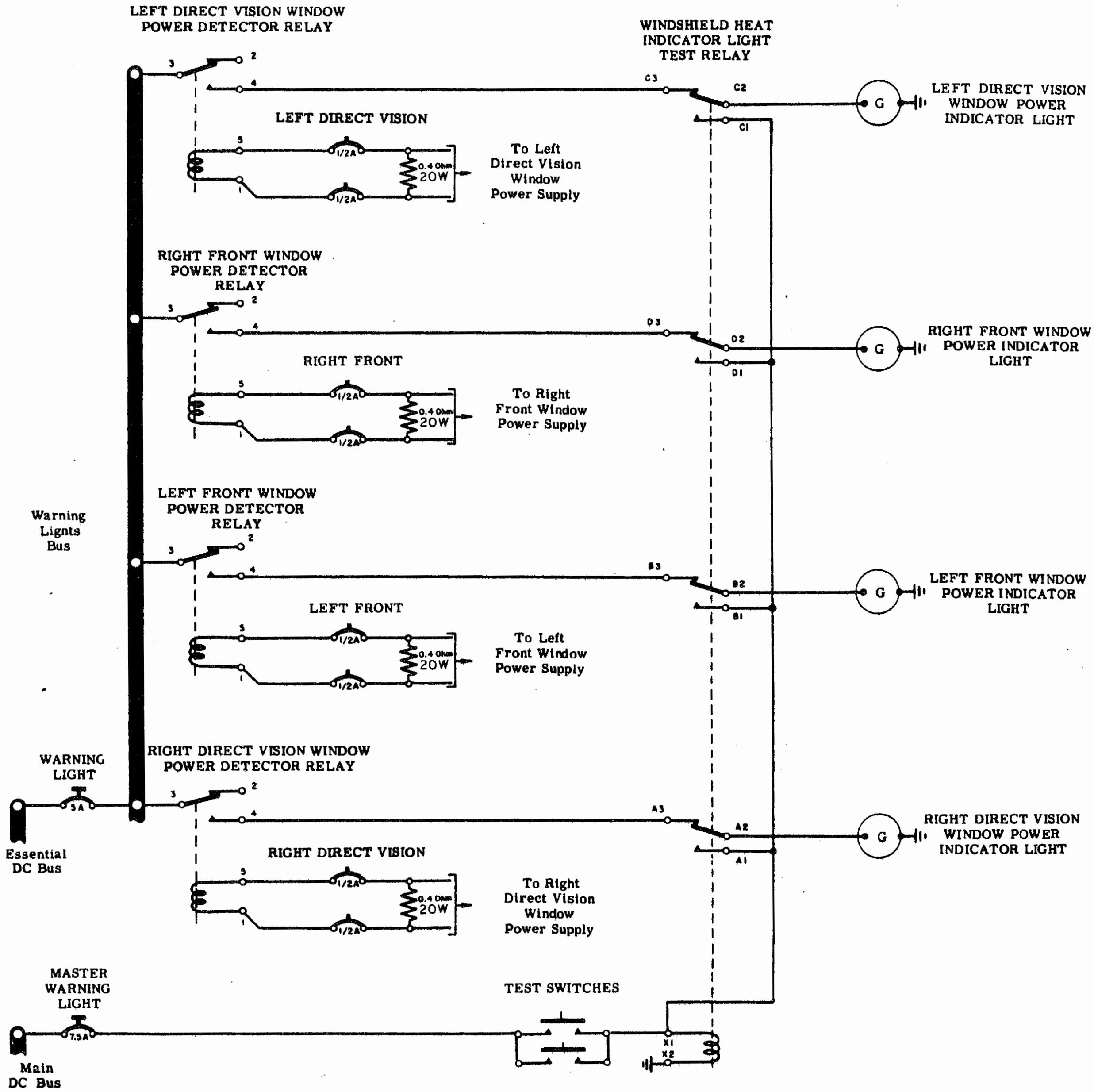
D. In case of trouble with the side windows, both windows are isolated, since there is only one bank of three circuit breakers controlling both side window power. The circuit breakers mentioned above are well placarded as to the window which they power.



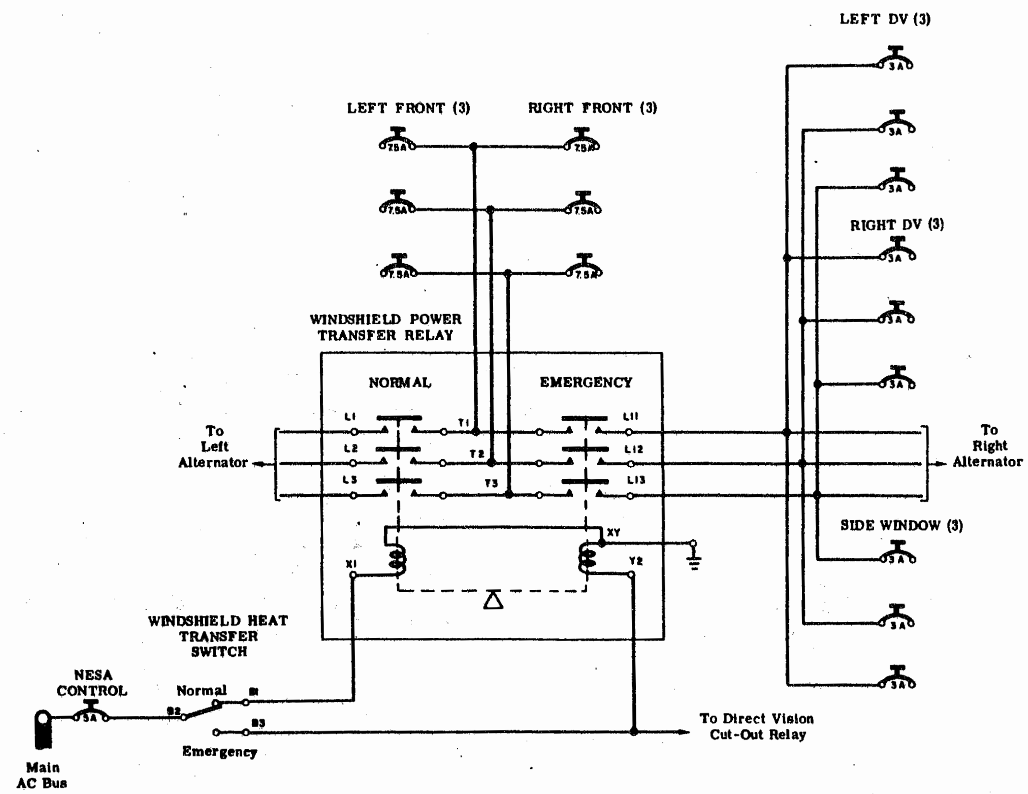
Windsheild Heat System — Block Diagram
Figure 1.



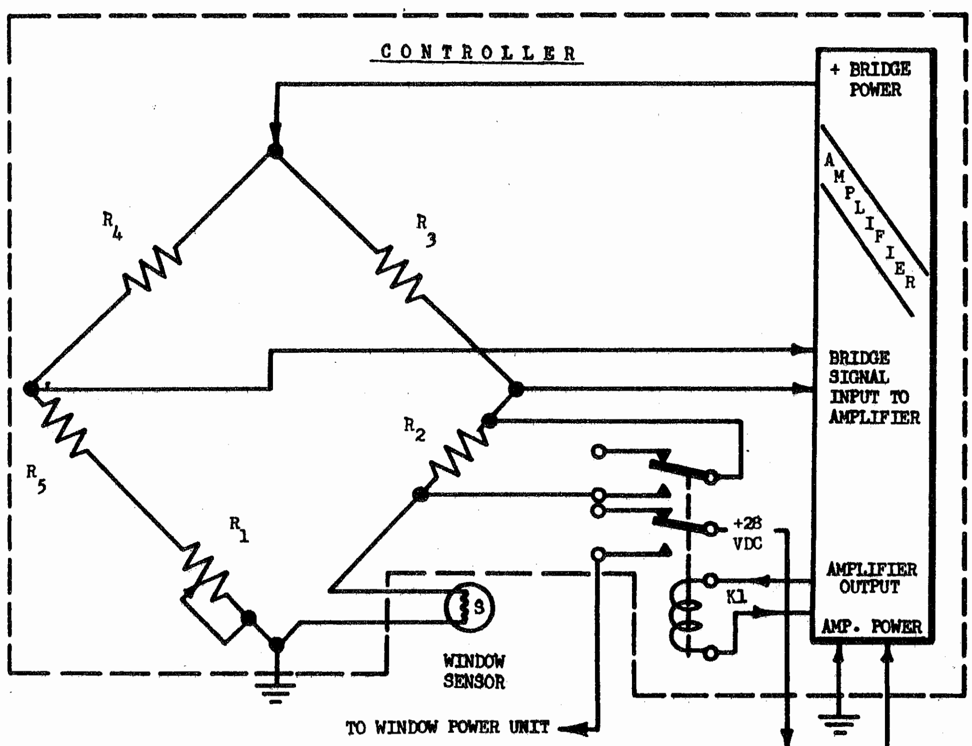
Windshield Heat Controls Location in Cockpit
Figure 2.



Windshield Heat Indicator Lights — Schematic
Figure 3.



Windshield Heat Power Transfer — Schematic
Figure 5.



NOTE: ALL COMPONENTS WITHIN THE DOTTED LINES ARE IN THE CONTROLLER UNIT

Windshield Heat Controller — Schematic
Figure 6.

WINDSHIELD / DV WINDOW DE-ICING SYSTEM — MAINTENANCE PRACTICES

1. **Windshield / DV Heat Power Unit — Removal / Installation**

A. Removal

(1) Remove electrical power from aircraft.

(2) Remove FLR No. 2 to gain access to windshield components, Fuselage Station 133 to Fuselage Station 169.

(3) Remove electrical connector.

(4) Remove power unit.

B. Installation

NOTE: Replacement power unit must be burnished at mounting holes to ensure a good electrical contact with aircraft structure.

On front and DV windshield power units the tap setting must be checked on replacement power unit to ensure power unit and window resistance code is identical. Side windows do not have tap settings.

Reference window tap setting code. It should have R_1 or Z, R_2 or Y, or R_3 or X. If window does not have a code marking a resistance check must be performed to determine tap setting. See chart in adjustment test section to determine tap setting from window resistance.

When the code is determined for the window, set the power unit electrical leads to identical tap. This is accomplished by removing top cover from power unit and changing three electrical leads.

(1) Install power unit.

(2) Inspect and connect electrical connector. Safety wire connector for security.

(3) Temporarily install FLR No. 2.

(4) Perform Windshield Heat System — Operational Test, see this Section.

(5) Return aircraft to flight status.

2. **Windshield / DV Heat Controllers — Removal / Installation**

A. Removal

(1) Remove electrical power from aircraft.

(2) Remove FLR No. 2 to gain access to windshield heat controllers (Fuselage Station 133 to Fuselage Station 169).

(3) Remove electrical connector from controller.

(4) Remove controller.

B. Installation

NOTE: Replacement unit must be burnished at mounting holes to ensure good electrical contact with aircraft structure.

(1) Install controller.

(2) Inspect and connect electrical connectors, safety wire connectors.

(3) Temporarily install FLR No. 2.

(4) Perform Windshield Heat System — Operational Test, see this Section.

(5) Return aircraft to flight status.

3. Windshield Heat De-icing Switch Box — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Remove FLR No. 2 (Fuselage Station 133 to Fuselage Station 169)

(3) Remove clamp from backshell, remove connector.

(4) Remove switch box.

B. Installation

NOTE: Replacement unit must be burnished at mounting holes to ensure a good electrical contact with aircraft structure.

(1) Install switch box.

(2) Inspect and connect electrical connectors, reinstall clamp, safety wire connectors.

(3) Temporarily install FLR No. 2.

(4) Perform Windshield Heat System — Operational Test, see this Section.

(5) Return aircraft to flight status.

4. Windshield / DV Window — Resistance Check

A. A considerable variation of resistance between identical windshield panels is possible due to production tolerances. The manufacturer checks these values and marks each panel to record findings. The permissible resistance range is split into three parts and is marked as follows:

(1) R_1 or Z - High resistance

(2) R_2 or Y - Medium resistance

(3) R_3 or X - Low resistance

To cope with the variations in glass resistance, the transformer-rectifier units for these windows are made with taps. These taps are accessible through a removable cover plate on the top of each unit. Taps are labeled in groups of X, Y and Z; with three taps (phases) in each group. Settings should be made so that all three phases use the same letter tap, X1, X2 and X3; in accordance with the resistance of that window to which the transformer-rectifier is connected. The X taps produce the lowest voltage, the Y taps the medium and the Z taps the highest voltage. If side windows are heated, it will be noted that no code marking is provided on the glass, and no taps are supplied in the side window power unit.

WARNING: DO NOT ATTEMPT TO MEASURE TRANSFORMER-RECTIFIER OUTPUT VOLTAGES WITH POWER LEADS DISCONNECTED FROM WIN DOW.

When replacing a window or a transformer-rectifier unit, tap settings of the transformer-rectifiers must be matched to the window resistance to ensure optimum performance. Check the resistance of the window and adjust the tap settings as follows, if necessary:

Panel	Panel Resistance Range (Ohms)	Panel Code Marking	Trans-Rect Setting	Trans-Rect Loaded Voltage ±20.6V
Front	94.4-102.5	R_1or Z	ZI, Z2, Z3	430
	85.5-94.5	R_2 or Y	YI,Y2,Y3	411
	76.5-85.5	R_3 or X	XI,X2, X3	391
*Direct Vision	62-69	R_1 or Z	Z1, Z2, Z3	186
	56-63	R_2 or Y	Y1,Y2,Y3	178
	51-57	R_3 or X	X1,X2,X3	169

Panel	Panel Resistance Range (Ohms)	Panel Code Marking	Trans-Rect Setting	Trans-Rect Loaded Voltage ±20.6V
Side (OPTIONAL)	62.9-85.1	NONE	NONE	115
*: *When determining voltage tap by means of resistance measurement of window and resistance reading falls into two tap ranges due to overlap use the higher voltage tap.				

NOTE: Each window panel must be treated separately. Should it be necessary to change a window panel, the setting of that panel unit must be rechecked in accordance with the new panel resistance.

B. Windshield/DV Window — Resistance Check

NOTE: The following procedure applies only to aircraft having windshields P/N 1595CCE100-11/-12 and DV windows P/N 1595CCE105-5/-6 manufactured in 1979 and 1980.

(1) Check windshield/DV window part number and serial number.

NOTE: Part number and serial number of windshield are located along bottom lower portion of window, approximately 2 to 3 inches from the center post. Part numbers must be read from the outside. Use of a mirror is recommended.

Part number and serial number of DV windows are located center of the top edge, where the sensor wires enter window.

Windshield P/N 1595CCE100-11/-12 and DV windows P/N 1595CCE105-5/-6 bearing manufacture serial numbers beginning with 9-H-1 through 0-H-12 are the affected windows. No further action is required if part numbers/serial numbers are not same as above.

(2) If affected serial numbered windows are installed, proceed as follows:

(a) Remove electrical power from aircraft.

(b) Disconnect leads and check windshield/DV window to ensure resistance is within tolerance of table below.

NOTE: Resistance code is located adjacent to serial number of front window and DV window.

NOTE: If resistances are within tolerances, windows may resume normal service until next recurring inspection.

If resistance does not exceed 10% of specified range maximums, transformer tap adjustments may be made and windows may resume normal service until next recurring inspection.

If resistance exceeds 10% of specified range maximums, flight operations must be conducted observing limitations in the Flight Manual applicable to window heat inoperative until affected windshield/DV window can be replaced.

(c) Reconnect leads.

5. **Windshield Heat System — Operation Test**

A. Perform engine run and maintain an engine speed of approximately 11,000 rpm.

B. If access to components is required, remove FLR No. 2.

C. Energize main ac and dc buses.

NOTE: If temperatures are high, an ice bag should be placed over sensor while performing following checks.

D. Ensure both engines are operating.

E. Place windshield heat switch to NORMAL position, then to DEFOG position.

F. Observe green indicator lights are cycling.

NOTE: Following indications verify associated conditions.

(1) Underheating indicated by a cool feeling of glass and no indicator lights.

(2) Overheating indicated when light stays on continuously or for abnormal long periods.

G. Place windshield heat switch to EMERGENCY position. Front windshield indicating lights should be cycling while side and DV windows are off.

H. Place windshield heat switch to DEICE position. Observe that green indicator lights are cycling.

I. Place windshield heat switch off.

J. Shut down engine.

K. Inspect area for presents of foreign objects, security of all attachments.

L. Install FLR No. 2 if removed.

PITOT HEAT SYSTEM — DESCRIPTION / OPERATION

1. **Description**

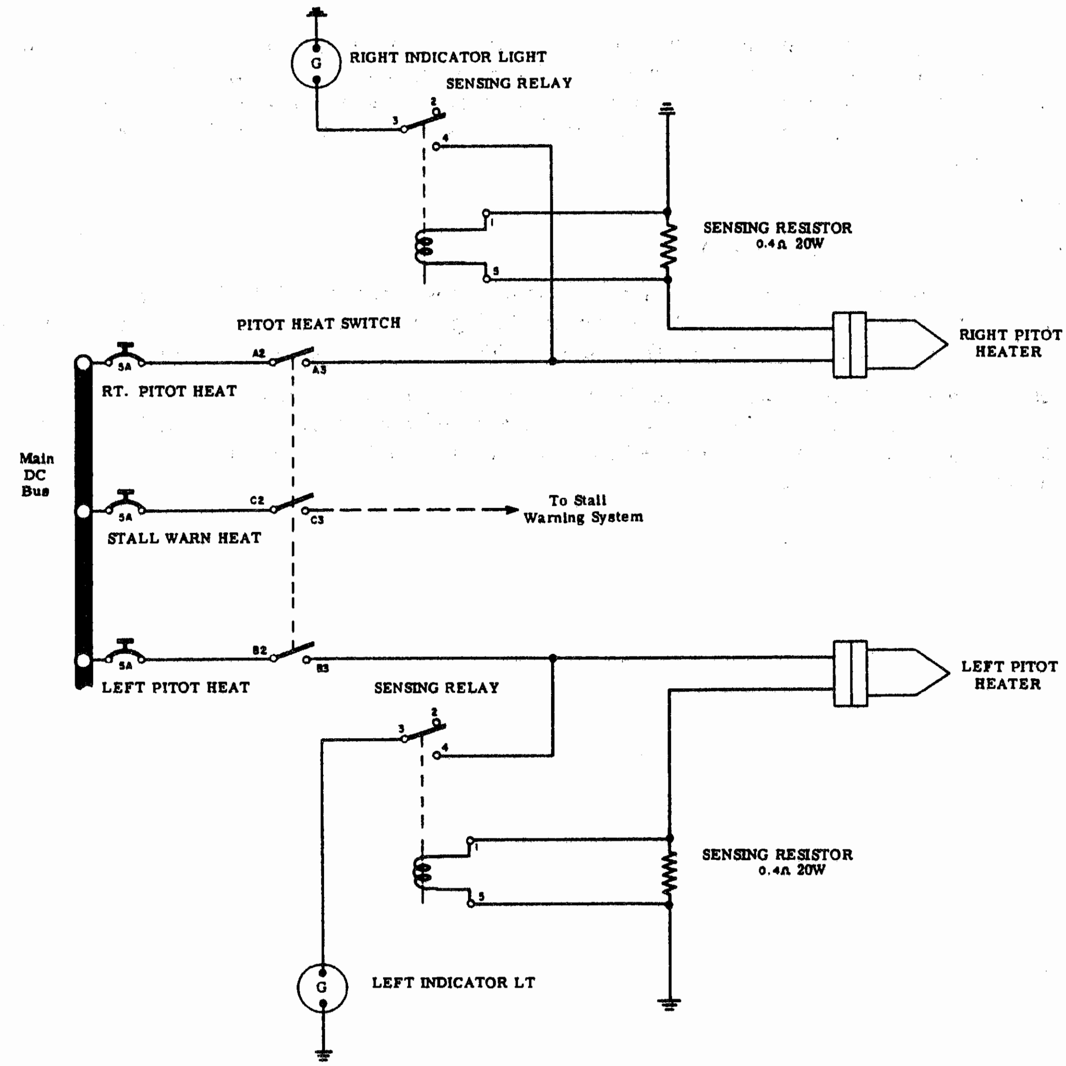
This system designed to prevent ice build-up on pitot tubes. System consists of 1 on/off switch, 2 circuit breakers, 2 sensor resistors, 2 sensing relays and 2 pitot heaters. pitot heat switch is located on upper overhead panel, in the de-icing section. indicator lights are located directly below pitot heat switch. Aircraft having ASC 216 adds a pitot heat lights test relay and lights dim relay. ASC 216 moves indicator lights to left side of pitot heat switch.

2. **Operation**

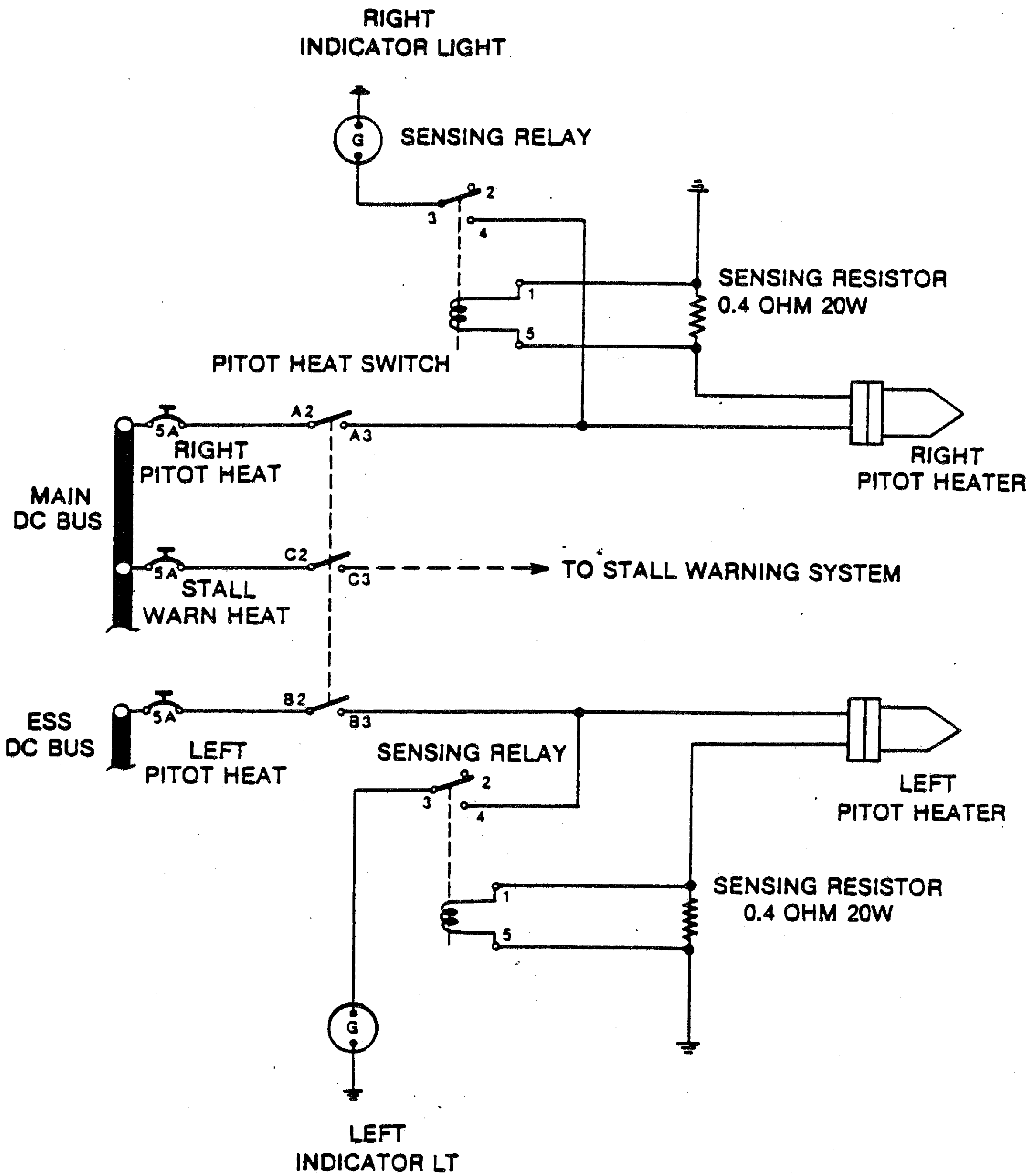
(See Figure 1, Figure 2 and Figure 3).

Aircraft not having asc 216, power is provided by main dc bus. when PITOT HEAT switch is placed to ON, pitot heaters are energized and indicator lights come on. During emergency operation, this system is inoperative. Aircraft 1 - 148, 322 and 323 having ASC 171 and 149 - 200, system will remain operative on left side only.

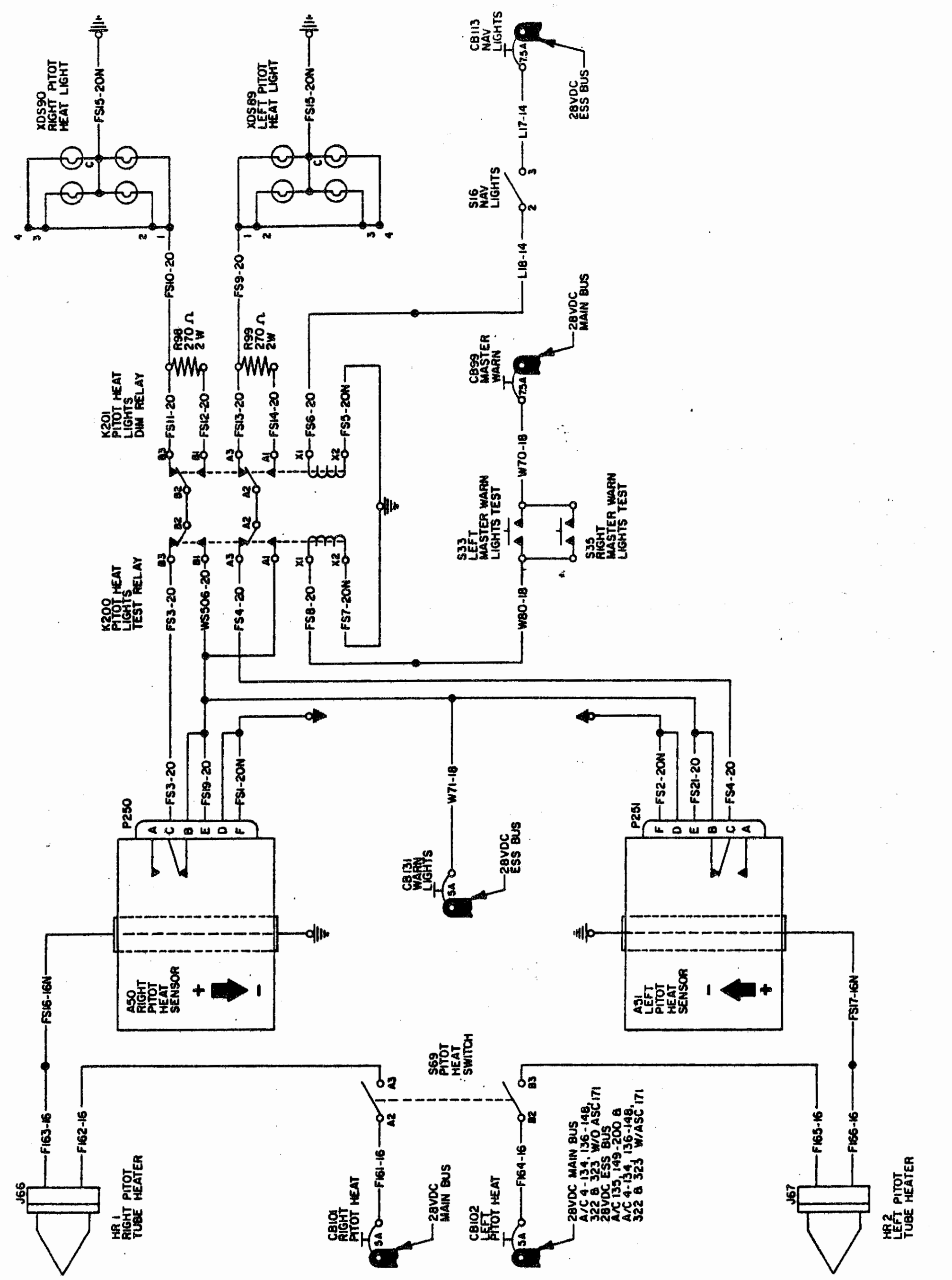
On aircraft having ASC 216, power is provided from main dc bus for right pitot heater and essential dc bus for left pitot heater. By placing pitot heat switch to on, pitot heaters are energized and indicator lights (which are normally energized on) will go out. with ASC 216, the indicator lights can be dimmed using the nav switch and tested using warn lts test switch.



Pitot Heat System — Schematic
Aircraft 1 - 148, 322, 323 not having ASC 171
Figure 1.



Pitot Heat System — Schematic
Aircraft 1 - 148, 322, 323 having ASC 171 and 149 - 200
Figure 2.



Pitot Heat System — Schematic
Aircraft 1 - 200, 322, 323 having ASC 216
Figure 3.

PITOT HEAT SYSTEM — MAINTENANCE PRACTICES

1. Pitot / Stall Warning Heat — Operational Test

A. Energize main and essential dc buses.

NOTE: Aircraft 1 - 148, 322 and 323 not having ASC 171; pitot heaters are inoperative in emergency dc as pitot heaters receive power from main dc bus. Aircraft having ASC 171 and Aircraft 148 - 200, left pitot heater is operative in emergency dc.

B. Pull Land R PITOT HTR circuit breakers and STALL HTR circuit breaker.

CAUTION: ENSURE LEFT AND RIGHT PITOT COVERS ARE REMOVED FROM PITOT TUBES.

DO NOT LEAVE HEAT SWITCHES ON WITH CIRCUIT BREAKER DEPRESSED FOR MORE THAN SEVERAL SECONDS. HEATING ELEMENT GETS EXTREMELY HOT AND CAN BURN ELEMENT WITH AIRCRAFT IN STATIC POSITION.

C. Place pitot heat switch ON.

D. Depress L PITOT HTR circuit breaker. Immediately touch pitot head lightly. DO NOT GRAB. As soon as heat is felt on pitot head place pitot heat switch OFF. Pull circuit breaker.

E. Place pitot heat switch ON. Depress R PITOT HTR circuit breaker. Immediately touch pitot head lightly. DO NOT GRAB. As soon as heat is felt on pitot head place pitot heat switch OFF. Pull circuit breaker.

F. With L and R PITOT HTR circuit breakers pulled, place pitot heat switch ON and depress STALL HTR circuit breaker. Stall warning vane, at left wing leading edge should begin to heat. Touch vane and vane plate carefully, again extreme care should be taken on length of time that heat is allowed to remain on.

G. Pull STALL HTR circuit breaker, place pitot heat switch OFF.

NOTE: Steps H. thru J. apply to aircraft not having ASC 216 only. Aircraft having ASC 216, proceed to Step K.

H. Place pitot heat switch ON. Depress L PITOT HTR circuit breaker. Observe that indicator light comes on. Place switch off. Pull circuit breaker. Perform same check for right indicator light.

I. Aircraft having ASC 171 and aircraft 149 - 200 turn battery switch to EMER. Place pitot heat switch ON and depress L and R PITOT HTR circuit breakers. Left pitot heat indicator light should come on, right indicator should be out as right pitot heat is inoperative in emergency dc operation.

J. Comply with Steps R, S. and T. below.

NOTE: Steps K. thru R. apply to aircraft having ASC 216 only. Aircraft not having ASC 216, proceed to Step S.

K. Ensure L and R PITOT HTR circuit breakers are reset.

L. Ensure pitot heat switch is OFF and LEFT and RIGHT PITOT HEAT monitor lights are ON.

M. Position pitot heat switch ON, LEFT and RIGHT PITOT HEAT monitor lights should go out.

N. Press warning light test switch, LEFT and RIGHT PITOT HEAT monitor lights should come on. Release warning light test switch, LEFT and RIGHT PITOT HEAT monitor lights should go out.

O. Position navigation light switch to ON. Press warning light test switch, LEFT and RIGHT PITOT HEAT monitor lights should come on in the dim mode. Release the warning light test switch and LEFT and RIGHT PITOT HEAT monitor lights should go out. Position the navigation light switch to OFF.

P. Pull R PITOT HTR circuit breaker, RIGHT PITOT HEAT monitor light should come on. Reset circuit breaker, RIGHT PITOT HEAT monitor light should go out.

Q. Pull L PITOT HTR circuit breaker, LEFT PITOT HEAT monitor light should come on. Reset circuit breaker, LEFT PITOT HEAT monitor light should go out.

R. Position pitot heat switch to OFF.

S. Remove dc power from aircraft.

T. Install pitot covers after pitot tubes have cooled.

CABIN WINDOW DEFOG SYSTEM — DESCRIPTION / OPERATION

1. **Description**

   A. Aircraft 1 - 178, 322 and 323. Fogging of the cabin windows is prevented by keeping the air dry in the space between the inner and outer panes of the windows. This is accomplished by manifolding these air spaces together and venting them overboard through a desiccator located beneath the cabin floor at Fuselage Station 384, FLR No. 15.

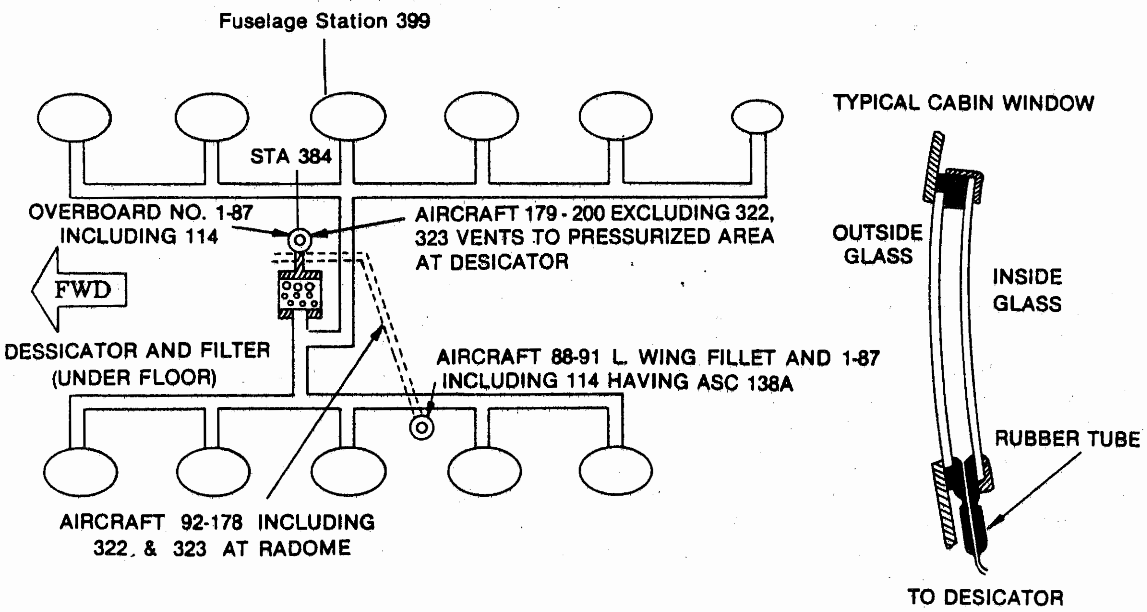
   B. Aircraft 179 - 200. The windows vent to the pressurized area through the desiccator located beneath the cabin floor at Fuselage Station 384 under FLR No. 15.

2. **Operation**

   (See Figure 1

   A. Aircraft 1 - 178,322 and 323. When the aircraft ascends, the air in the spaces between window panes expands due to the decreasing ambient pressure and a portion of it exits overboard through the desiccator. As the aircraft descends, the increasing ambient pressure forces air back into the spaces. This air is dried while passing through the desiccator. In this manner, the air between panes is kept dry so that no moisture will condense out of it upon the glass surfaces bounding the air space. The air space and the two panes of glass provide sufficient insulation to maintain the inner surface of the inner pane above the dew point of the cabin air.

   B. Aircraft 179 - 200, 322 and 323. As the aircraft is pressurized, the cabin pressure is maintained in the space between the window panes. Any air flow required to equalize the pressure between the space between the window panes and cabin through the desiccator, which removes moisture and prevents fogging as the aircraft depressurizes the pressure in the window, being higher than the cabin, exits through the desiccator.



NOTES:

A. When desiccant is pink in color, remove from desiccator and bake in oven for 2 hours (until blue).

B. Allow to cool.

C. Replace in desiccator.

D. Check filters and replace as necessary.

E. Reinstall desiccator in system.

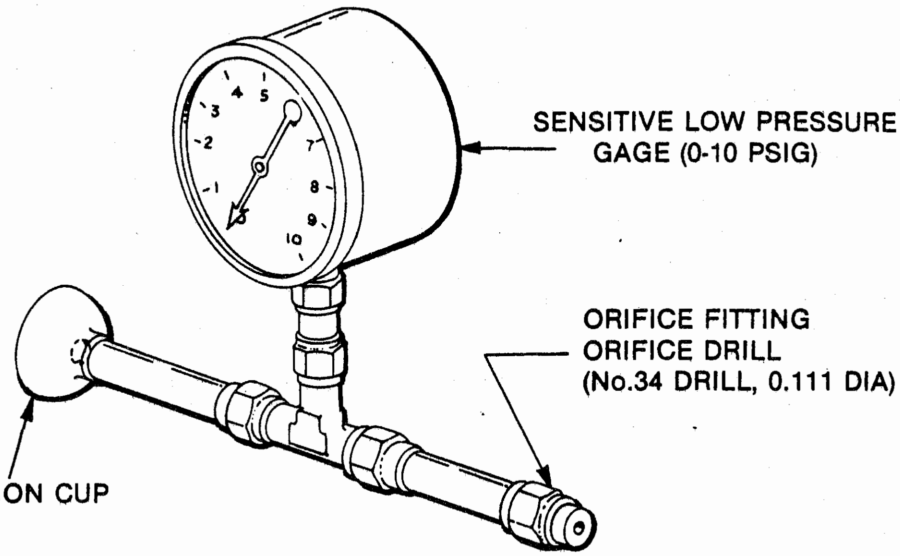
Cabin Window Defog System — Flow Diagram
Figure 1.

CABIN WINDOW DEFOG SYSTEM — MAINTENANCE PRACTICES

1. **System Leakage Test**

NOTE: Perform this test during fuselage leakage test.

A. Connect pressure gage and orifice (See Figure 201) cabin window defog vent as follows:



Cabin Window Defog System — Leakage Test Gage
Figure 201.

(1) Aircraft 1 - 87 and 114 at Fuselage Station 379.5 on bottom centerline of aircraft.

(2) Aircraft 1 - 87 and 114 having ASC 138A and Aircraft 88 - 91 in left wing fillet at Fuselage Station 388. Remove lower left wing root fairing cover at landing light.

(3) Aircraft 92 - 178, 322 and 323 excluding 114; located behind radome. Remove oxygen bottle access plate from left side of Fuselage Station 50 and 60 to attach gage and orifice for leakage test, remove suction cup from test gage.

B. When fuselage is pressurized to 6.5 psig and pressure has stabilized, observe pressure gage connected to defog vent. Pressure should not exceed 0.1 psig, 0.2 hg. or 2.8 inches of water.

C. If pressure exceeds that stated in Step B., check all fittings and lines for proper tightness and that they are not cracked.

D. Aircraft 179 - 200 the defog system vents through the desiccator to the pressurized area at the desiccator. The above test does not apply to these aircraft.

2. Desiccator — Check

A. Check condition of silica gel by comparison with color strip on container.

NOTE: When the silica gel crystals show by color comparison that they are no longer effective, they should be removed and replaced with serviceable crystals.

B. To return crystals to useable condition, proceed as follows:

(1) Remove desiccator from aircraft as outlined in this section.

(2) Remove four tie rods from container and disassemble.

(3) Remove filters and crystals for inspection and replacement if dirty.

(4) Place used crystals in shallow pan and heat in an oven at 350°F for 3 hours to make them serviceable again. Allow to cool before reinstallation into container.

(5) Install one filter and one end plate on container.

(6) Fill container with serviceable crystals.

(7) Install other filter and end plate.

(8) Position end plates and install four tie rods. Reinstall container in aircraft.

NOTE: If desiccators are being stored, the inlet and outlet ports must be plugged to prevent absorption of moisture.

3. Cabin Window Desiccator — Removal / Installation

A. Removal

(1) Remove FLR No. 15 Fuselage Station 353 to Fuselage Station 416.

(2) Disconnect desiccator and cap lines at each end of unit.

(3) Remove attaching screws from aft side.

(4) Remove desiccator.

B. Installation

(1) Ensure silica gel crystals are in serviceable condition.

(2) Install desiccator and attach mounting screws.

(3) Connect lines at each end of unit.

(4) Install floor boards.

4. Cabin Window Desiccator Crystals — Inspection

A. Remove FLR No. 15 (Fuselage Station 353 to 416).

B. Inspect condition of silica gel crystals by comparison with color strip on container.

NOTE: When silica gel crystals show by color comparison that they are no longer effective. they should be replaced.

C. Inspect filters and replace as required. See removal procedure this section.

D. Inspect area for presents of foreign objects, security of all attachments.

E. Install FLR No. 15.

5. **Emergency Window Desiccator — Inspection**

   A. Gain access to emergency window desiccator.

   B. Disconnect desiccator and cap lines at each end of unit.

   C. Remove desiccator.

   D. Inspect condition of silica gel crystals by comparison with strip on container.

   NOTE: When silica gel crystals show by color comparison that they are no longer effective, they should be replaced.

   E. Inspect filters and replace as required.

   F. Install desiccator.

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ENGINE AND PROPELLER DEICING SYSTEM — DESCRIPTION / OPERATION

1. Description

The propeller and engine air intake of each power plant are deiced by built-in electric heating elements. The elements are located on the leading edge of each propeller blade, inside the nose of each propeller spinner, on the leading edge of each oil cooler intake, on the leading edge of each engine air intake and pads on the inner and outer surfaces around each engine air intake. The electric power for the heating elements is obtained from a 212 volt, 11KVA three-phase alternator mounted on the respective engine gear box. The heating elements are powered in parallel from the three alternator phases through a cyclic timer which interrupts the supply of power to the propeller and spinner elements and the element pads of the engine air intake. The left alternator powers the left system. The right alternator powers the right system. There is no crossover in the event that one alternator fails. The narrow elements at the leading edge of the oil cooler and engine air intake are powered continuously. This cyclic timer has two speeds (FAST or SLOW), either of which can be selected by the aircraft crew.

2. Operation

Engine and propeller deicing is controlled from the two deice selector switches located on the upper overhead panel as shown in Figure 1. They are completely independent of each other, and identical in operation. In either system with the engine operating and alternator power available, moving the selector switch to FAST or SLOW will complete a circuit to the coil of the deice power relay located in the appropriate nacelle relay box, allowing alternator power to enter the adjacent cyclic timer. The timer will distribute power to the various elements described above, as shown on Figure 2 and Figure 3. A green indicator light is located above the selector switch. This light indicates to the crew that the system is working normally. To have the light come on steadily, two conditions must be met: The alternator for that side must be on the line and the ENGINE and PROP DEICE power switch for that side must be in either FAST or SLOW. If the timer motor is working, this light pulsates bright and dim at regular intervals depending on the speed of the motor. With the switch positioned to FAST, 2 second pulses result. With the switch positioned to SLOW, 6 second pulses result. This indicates that the timer is actually cycling the heat among all elements. For indication of proper system operation, the light must pulse. A sensing relay, located in the mid relay box (under FLR No. 8) closes when the alternator system is active. This relay is incorporated in the circuit to the indicating light to prevent indication if power supply is not functioning. One fast cycle of heat application to all elements takes a total time of 2 minutes, while a slow cycle takes a total time of 6 minutes. The cycle keeps repeating until the system is shut down or the rate is changed.

The operation of the de-icer portion of the system permits ice to form on the cyclically heated pads to act as an insulation blanket. When heat is applied by the timer switch, the ice melts, breaks off in small pieces and is carried away by the air stream. During this operation, the engine injests small pieces of ice as can be noted sometimes by puffs of blue smoke from the tailpipe. This is not a problem since it is caused by momentary changes in fuel to air ratio within the combustion chambers of the engine. It should be stressed, that proper use of the system is mandatory to obtain satisfactory results. The system is intended for use during flight, and temperature of system elements will only remain within limits when cooled by airflow. The propeller elements are adequately cooled by the motion of the blades at any stable engine rpm, even though the aircraft is stationary. The inlet cowl elements receive some cooling from active propeller blades. For adequate cooling of these elements, the aircraft must be moving with relation to the surrounding air. Operation for a length of time of the engine and propeller deicing system while the aircraft is standing on the ground will result in damage to the heating elements. Engine and propeller deicing on the ground should be limited to one or two test cycles. To avoid trouble, the engine and propeller ice protection system must be actuated before encountering icing conditions. When it is expected that icing conditions will be encountered, the ice protection system for both engines must be switched on immediately after becoming airborne. If the system is not activated in time it can be overpowered or too much ice can be broken off during a heating application, causing improper engine operation or even possible flameout.

NOTE: Refer to the Aircraft Flight Manual, for the proper airstart procedure should ice conditions be encountered before the system is placed in operation.

In addition to having the engine propeller ice protection system on before encountering any icing conditions, the correct cycling sequence must be selected. The highest rates of icing can occur at higher air temperatures. Fast speed must be selected at indicated outside air temperatures of −6°C and above, to ensure that excessive ice does not accumulate during the cyclic off periods and that run back icing is kept to a minimum during the on periods. Slow speed must be selected for indicated outside air temperatures of −6°C and below, to ensure that the heat is applied for a sufficiently long period to promote adequate shedding from the cycled portion. If these portions are not properly deiced during the heat on periods, excessive ice may progressively build up during the off periods. With these facts in mind, it may be seen that on no account should the pilot change from FAST to SLOW in heavy icing above −6°C indicated OAT, in order to obtain a longer heat on period.

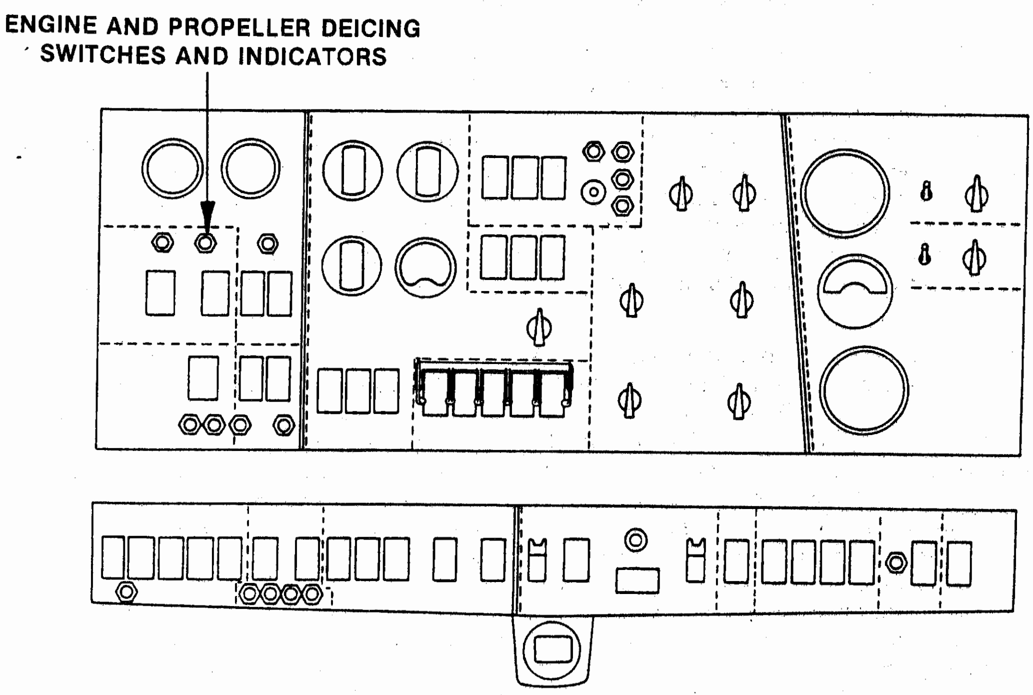
NOTE: The engine and propeller deice system is inoperative during emergency dc operation.

3. **Malfunction Indicating Lights — Operation**

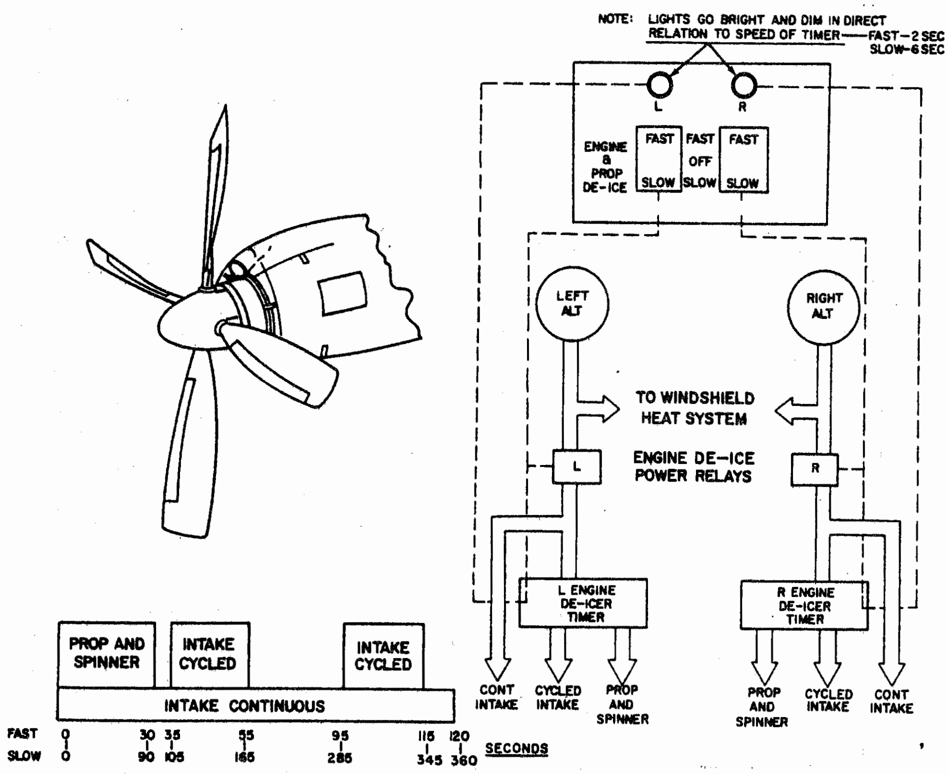
A. With alternator running and on the line, the engine and propeller deice switch positioned to FAST or SLOW and the timer motor not running, the indicator light will be on, but not pulsating.

B. With the alternator running and on the line and the switch positioned to OFF, the indicator light will be out.

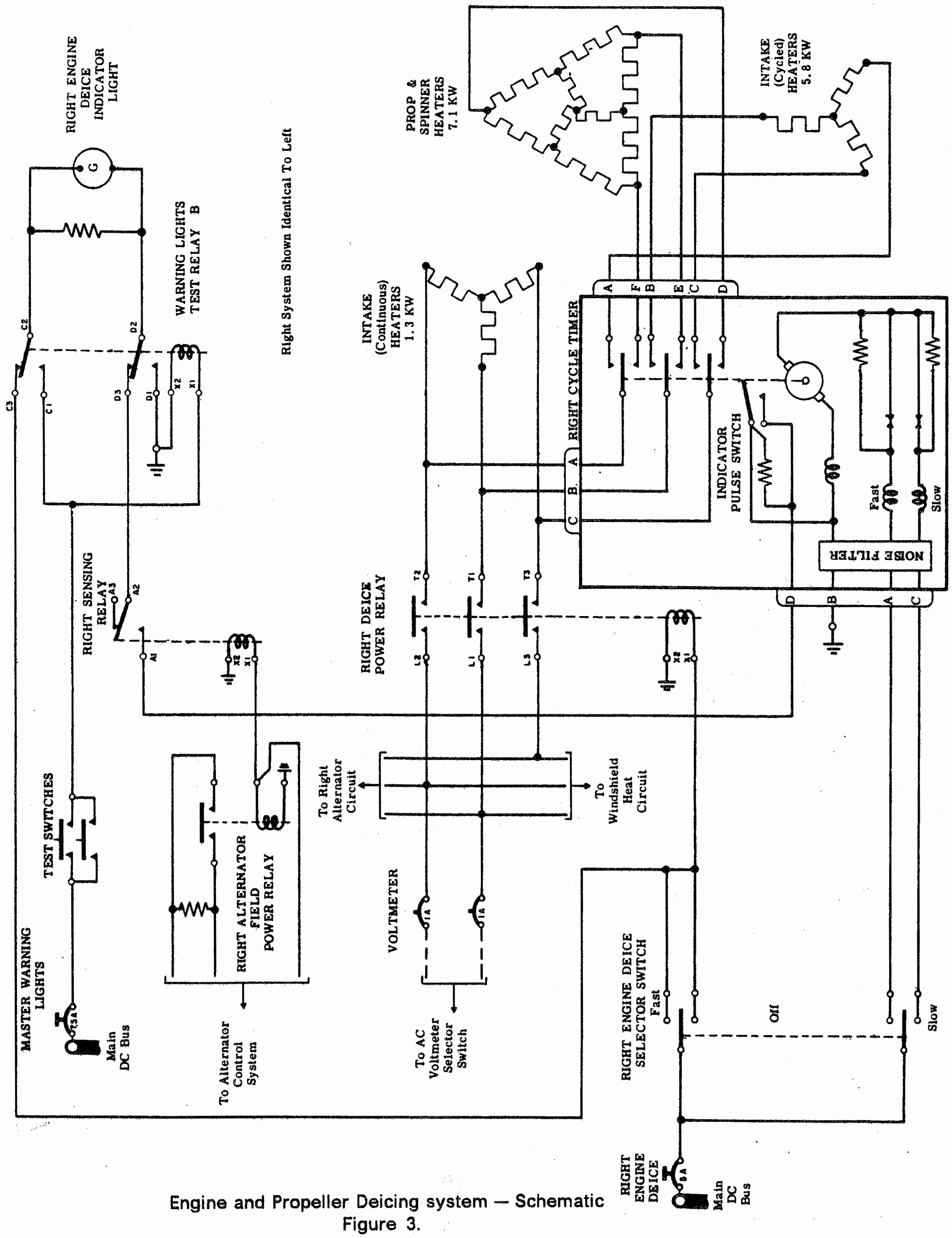
C. With the alternator not on the line and the switch positioned to FAST or SLOW, the indicator light will be out.



Location of Deicing Switches — Upper Overhead Panel
Figure 1.



Engine and Propeller Deicing system — Block Diagram
Figure 2.



Engine and Propeller Deicing system — Schematic
Figure 3.

ENGINE AND PROPELLER DEICING SYSTEM — MAINTENANCE PRACTICES

1. **Propeller De-ice System — Operational Test**

(See Figure 201).

NOTE: The following instructions contain procedures for checking out the prop deicers. Three test procedures are given, each one more exacting than the preceding test. Selection of the procedure to be used depends on time, equipment available and degree of accuracy required.

A. Voltage Check

NOTE: This method can be performed either on the ground or, if weather is such that the windshield heat system can be de-energized, in flight.

(1) Position WINDSHIELD HEAT selector switch to OFF.

(2) Operate at least one engine.

(3) Set AC VOLTS SELECTOR switch to monitor alternator on side being checked.

(4) Position alternator control switch to RESET, and then to ON.

(5) Select SLOW or FAST on engine and propeller deice selector switch.

(6) Monitor alternator voltage over at least one full deicing cycle (fast rate has a duration of 2 minutes, slow rate has a duration of 6 minutes). Voltage should react sharply to changing loads imposed by timer. Range of 206 volts to 216 volts is normal. During cycling period, deice indicating light should pulse dim/bright at regular intervals. The fact that the alternator system current balance relay does not trip the alternator is accepted as proof that the propeller and engine deicing loads are balanced within limits. Should the alternator trip off the line, the alternator and engine deice systems require investigation.

CAUTION: IF VOLTAGE CHECK IS BEING CONDUCTED ON THE GROUND, A MINIMUM ENGINE RPM OF 12,000 MUST BE MAINTAINED. DO NOT PROLONG TEST BEYOND TWO CYCLES.

(7) Repeat steps described previously for the other engine.

(8) Restore aircraft to flight condition.

B. Resistance Check

NOTE: This procedure requires an accurate ohmmeter or other resistance measuring device (Wheatstone Bridge is preferred).

(1) Remove spinner to eliminate its heat element.

(2) Remove removable cover panel assembly.

(3) Rotate propeller to line up panel opening with brush box assembly on brush housing.

(4) Withdraw brush box assembly.

(5) Check resistance of blade deicing element from any terminal strip at base of each blade. For identification three terminals at strip are called 1, 2 and 3. Check with meter as follows:

(a) Terminal 1 to 2

(b) Terminal 1 to 3

(c) Terminal 2 to 3

Resistance should be 17.4 to 20.0 ohms. If resistance is above 20.0 ohms but below 21.4 ohms, disconnect leads from each terminal strip and check each blade individually. If each is within 69.7 to 85.3 ohms propeller is serviceable. If resistance with leads connected is above 21.4 ohms propeller is unserviceable.

(6) With leads connected check insulation of each terminal to ground (prop metal structure) using a megger. Minimum in service resistance is 1 megohm.

(7) Install brush box assembly and removable panel.

(8) After satisfactory resistance readings are obtained, reconnect system to restore aircraft to normal.

(9) Operate propeller cyclic timer by positioning selector switch in the cockpit to FAST or SLOW. Listen to timer for indicators or operation. Motor operation and contact action should be audible.

(10) Repeat checkout procedure for other engine.

(11) Restore aircraft to flight condition.

C. Operational Check

NOTE: This method requires a power supply and can utilize ammeters. It is the most positive of the checks listed.

(1) Gain access to nacelle relay box in upper nacelle.

CAUTION: TAKE NECESSARY STEPS TO ENSURE THAT NO TEST POWER STRAYS INTO ALTERNATOR, IT'S RELAYS OR CONTROLS, REGARDLESS OF CONNECTOR POINT. REMOVE THE THREE WIRES AT THE DEICE POWER RELAY THAT COME FROM THE ALTERNATOR.

(2) Attach leads from power source to propeller deicing circuit. This can be done at the cyclic timer connector, using test receptacle, or at deice power relay. Connection can also be made at propeller blade terminals on front face of propeller (underneath the spinner).

(3) Apply power for checkout procedure. This should be 400 cps, single or three phase, and should be a low voltage of between 70 and 90 volts. Use of higher voltages can overheat and damage heating elements. Even with recommended reduced voltages, it is wise not to prolong tests unnecessarily.

(4) Feel deicing elements to detect presence of heat. All elements of propeller blades and spinner should heat, but if single phase test power is used, this heat is not uniformly distributed.

(5) Check line currents if desired. Portable ammeter or permanent ammeter in power supply line can be used. Currents should fall within limits specified by the Dowty Rotol Propeller Manual. If single phase power is used, currents will be 1/1.732 of the values listed.

(6) Operate cycle timer. This units activity can be monitored aurally or by allowing unit to switch power.

(7) Repeat preceding steps for other engine.

(8) Restore aircraft to flight condition.

2. Engine / Propeller Deicing Timer — Removal / Installation

A. Removal

(1) Remove aircraft electrical power.

(2) Remove nacelle relay box access cover.

(3) Disconnect large connector.

NOTE: Connectors may be frozen on timer box. Exercise care not to damage connector during removal.

(4) Remove timer box.

(5) Disconnect remaining connectors.

B. Installation

CAUTION: DURING OPERATIONAL CHECK ENGINE SPEED MUST BE MAINTAINED AT 12,000 RPM. DO NOT PROLONG TEST BEYOND TWO LOAD CYCLES.

(1) Install smaller electrical connectors and safety wire.

(2) Install timer box.

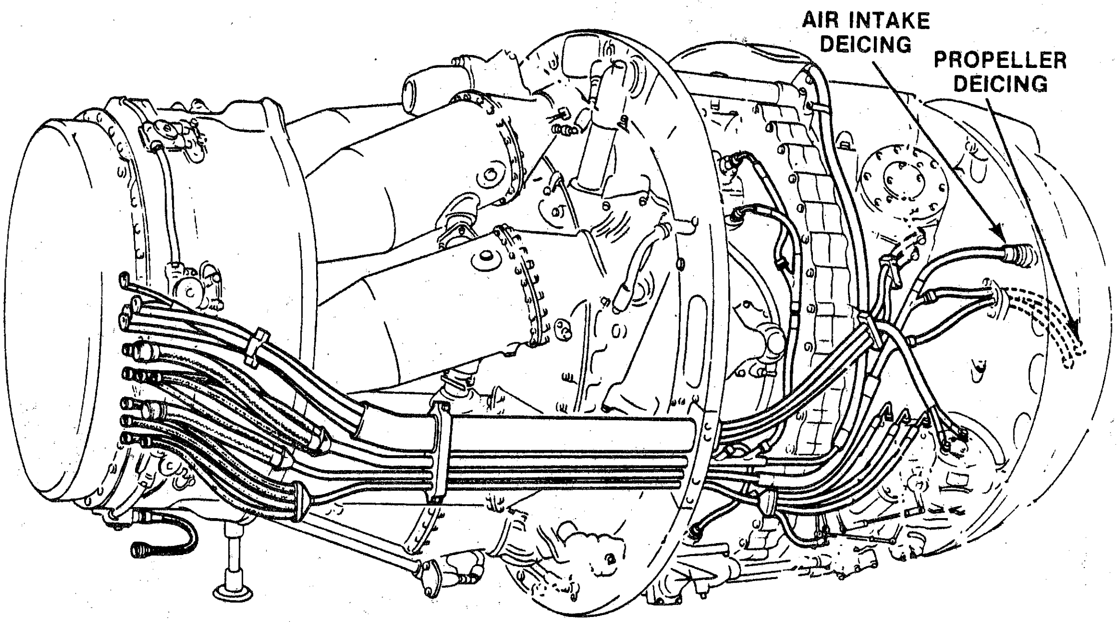
(3) Install large connector and safety wire.

(4) Perform Propeller De-ice System — Operational Test, see this Section.

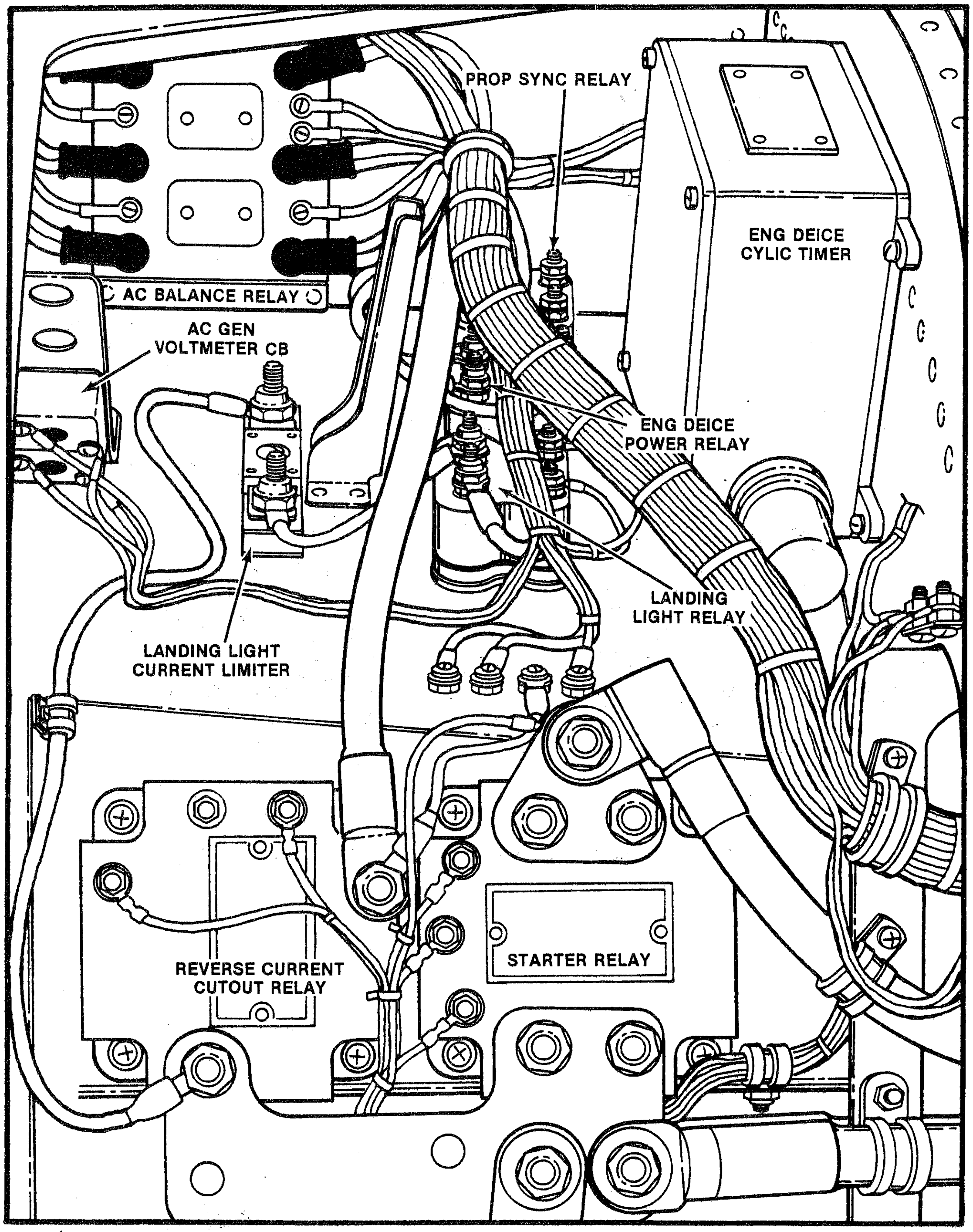
(5) Inspect area for presents of foreign objects, security of all attachments.

(6) Install nacelle relay box access cover.

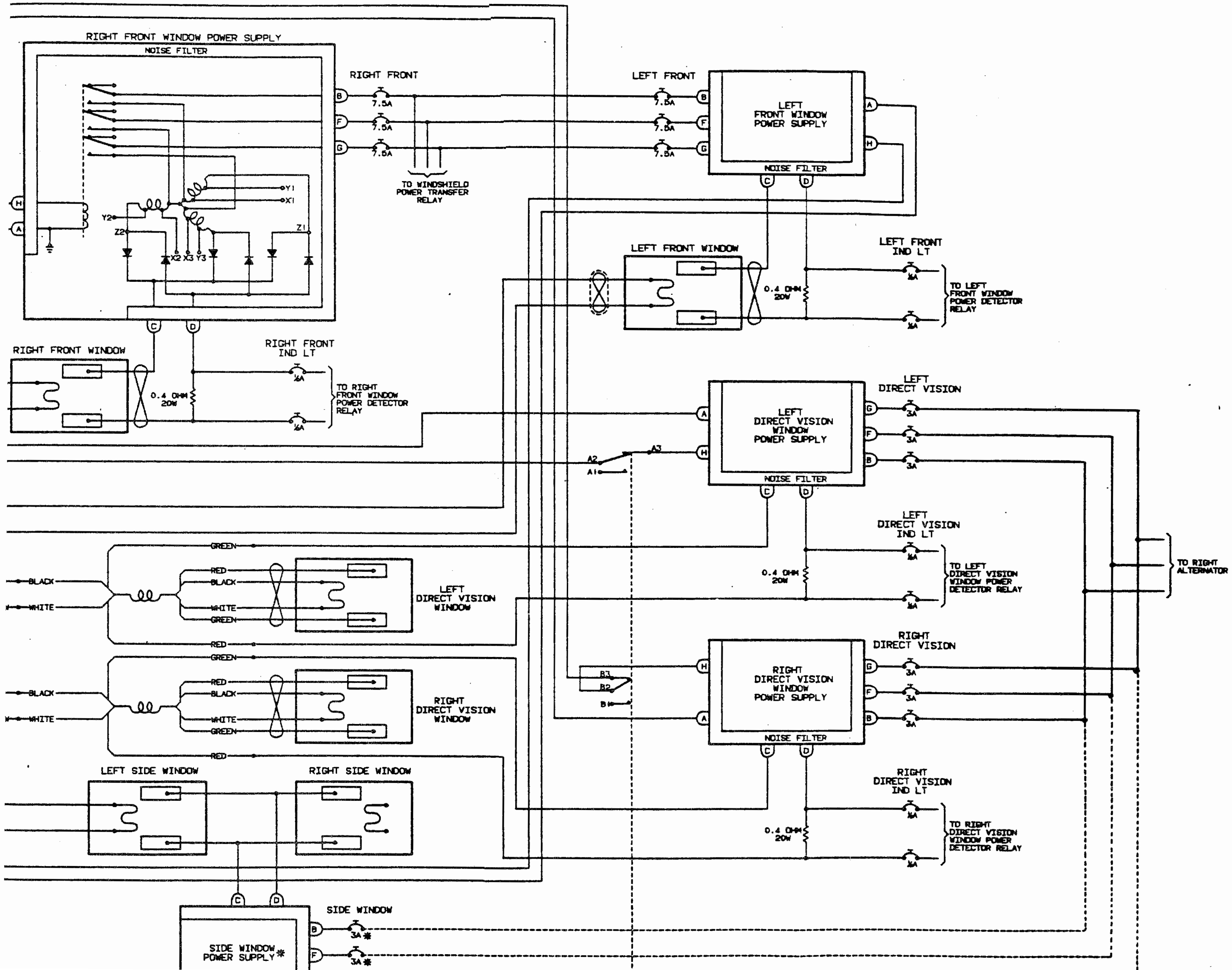
(7) Return aircraft to flight status.

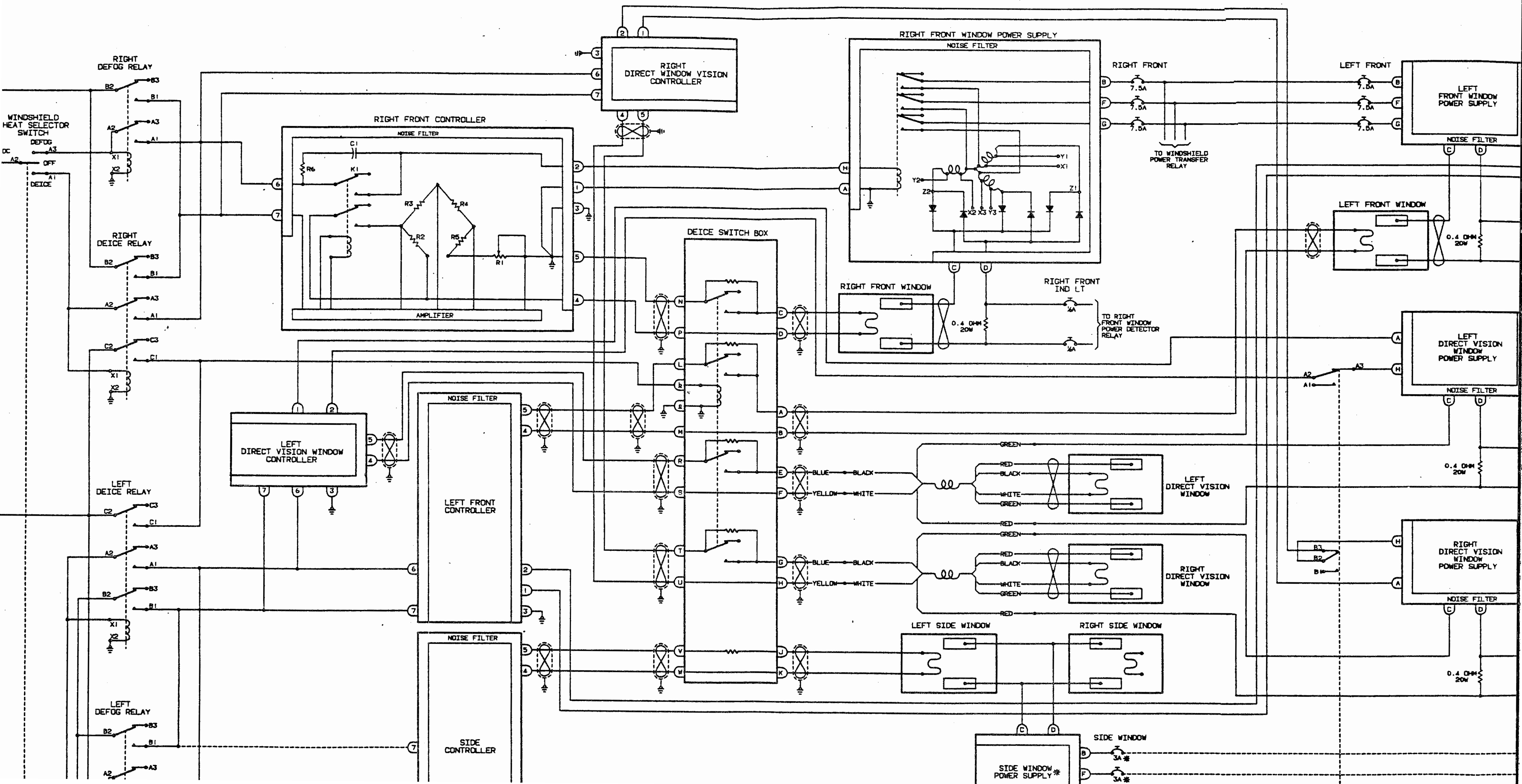


Engine and Propeller De-icing System Engine Electrical Connections — Right side
Figure 201.



Engine and Propeller De-icing System Location in Right Nacelle Relay Box
Figure 202.


RIGHT FRONT WINDOW POWER SUPPLY
NOISE FILTER
RIGHT FRONT
7.5A
TO WINDSHIELD POWER TRANSFER RELAY
LEFT FRONT
LEFT FRONT WINDOW POWER SUPPLY
NOISE FILTER
LEFT FRONT WINDOW
LEFT FRONT IND LT
0.4 OHM 20W
TO LEFT FRONT WINDOW POWER DETECTOR RELAY
RIGHT FRONT WINDOW
RIGHT FRONT IND LT
TO RIGHT FRONT WINDOW POWER DETECTOR RELAY
LEFT DIRECT VISION WINDOW POWER SUPPLY
LEFT DIRECT VISION
3A
LEFT DIRECT VISION IND LT
TO LEFT DIRECT VISION WINDOW POWER DETECTOR RELAY
TO RIGHT ALTERNATOR
GREEN
RED
BLACK
WHITE
LEFT DIRECT VISION WINDOW
RIGHT DIRECT VISION WINDOW
RIGHT DIRECT VISION WINDOW POWER SUPPLY
RIGHT DIRECT VISION
RIGHT DIRECT VISION IND LT
TO RIGHT DIRECT VISION WINDOW POWER DETECTOR RELAY
LEFT SIDE WINDOW
RIGHT SIDE WINDOW
SIDE WINDOW POWER SUPPLY*
SIDE WINDOW
3A*



RIGHT DEFOG RELAY
WINDSHIELD HEAT SELECTOR SWITCH
DEFOG
OFF
DEICE
RIGHT DEICE RELAY
LEFT DEICE RELAY
LEFT DEFOG RELAY
RIGHT DIRECT WINDOW VISION CONTROLLER
RIGHT FRONT CONTROLLER
NOISE FILTER
AMPLIFIER
RIGHT FRONT WINDOW POWER SUPPLY
RIGHT FRONT
LEFT FRONT
7.5A
TO WINDSHIELD POWER TRANSFER RELAY
LEFT FRONT WINDOW POWER SUPPLY
LEFT FRONT WINDOW
0.4 OHM 20W
DEICE SWITCH BOX
RIGHT FRONT WINDOW
RIGHT FRONT IND LT
TO RIGHT FRONT WINDOW POWER DETECTOR RELAY
LEFT DIRECT VISION WINDOW POWER SUPPLY
LEFT DIRECT VISION WINDOW CONTROLLER
LEFT FRONT CONTROLLER
BLUE
BLACK
YELLOW
WHITE
GREEN
RED
LEFT DIRECT VISION WINDOW
RIGHT DIRECT VISION WINDOW
RIGHT DIRECT VISION WINDOW POWER SUPPLY
LEFT SIDE WINDOW
RIGHT SIDE WINDOW
SIDE CONTROLLER
SIDE WINDOW
3A
SIDE WINDOW POWER SUPPLY

LIST OF EFFECTIVE PAGES

CHAPTER 31

INSTRUMENTS

* Pages revised by Revision 47
** Pages added by Revision 47
*** Pages deleted by Revision 47

TABLE OF CONTENTS

CHAPTER 31

INSTRUMENTS

INSTRUMENTS — DESCRIPTION / OPERATION

1. **Description**

A. Pilot and copilot Instrument Panels

The pilot and copilot instrument panel assemblies in the flight compartment consist of the pilot instrument panel (space provision only), center instrument panel and copilot instrument panel (space provision only). The center instrument panel is divided into three sections which are secured to the structure of the aircraft. The panel contains engine and fuel instruments which are visible to both the pilot and copilot. All center panel indicators are rear bezel mounted. Screws on the aft side of the panel secure each indicator to the panel. Provision has been made on the center panel for the installation of radar monitor. (See Figure 1)

Two skirt panels are installed at the lower edge of each main instrument panel. Mounted on the pilots left instrument skirt panel are the pilots oxygen flow indicator, oxygen cylinder pressure gage and oxygen supply shutoff valve. The pilots landing gear selector handle and (aircraft having ASC 108A, part 1) speedbrake indication lights are on the pilots right instrument skirt panel. On the copilots left instrument skirt panel are the copilots landing gear selector handle, main and nosewheel indicator lights, wheel light intensity switch, landing gear horn cutoff button and no horn indicator light. On the copilots right instrument skirt panel are the emergency and normal hydraulic pressure gages, copilots emergency hydraulic pump switch and oxygen flowmeter indicator. Each of the four panels are individually secured to the structure of the aircraft.

Lighting for all instrument panels installed in the flight compartment area is red or white, or a combination of red and white flood lighting. The lighting is controlled by rheostats from the left and right lighting control panels located above the side consoles. In addition, a white floodlight override switch is installed on the lower overhead panel. (See to Chapter 33 for Interior Lighting.) On aircraft 87-200, 322, 323 and 1-86 and 114 having ASC 115, instruments on the instrument panel are provided with additional individual lights.

B. Eyebrow Panel

(See Figure 2)

The eyebrow panel is installed above the main instrument panel and extends aft and over all three panels providing easy accessibility for the pilots. The eyebrow panel is secured to the aircraft structure. On this panel are the master warning lights, propeller below flight fine pitch lock, cruise lockout, cruise pitch lights, feather pump lights nacelle and auxiliary power unit fire warning lights. The switches and controls on the eyebrow panel include the warning light reset buttons, warning light test, fire detector, fuel gage test buttons, engine fire pull T-handles feathering pump buttons, propeller synchronizer switch, flight safety lock switch and auxiliary power unit fire extinguisher switch. The engine fire extinguisher switches are located in back of the fire pull T-handles. The engine fire extinguisher switch toggles are exposed when the T-handles are pulled out.

C. Center Pedestal Console/Radio Control Panel

(See Figure 3)

Located on center pedestal console are engine power levers the high pressure fuel cocks and flight fine pitch lock selector lever. These levers operate a series of cams and microswitches located inside the pedestal. Each switch is actuated by its own cam. In all cases four switches are grouped together along with four cams. The four cams are bolted together along with one cam lever. This assembly (cams and lever) move as one unit when actuated by the pedestal lever.

Each cam can be individually adjusted to meet the requirement of the switch. This can be done by rotating the cam, as required, in order to advance or retard the switch actuation.

The side face of each lever and of each cam has a raised boss. This boss is a ring 1 1/8 inches in diameter and 1/8 inch wide with 160 serrated teeth on the face. (one serration every 2° 15′). These serrations serve two purposes, one is to lock the cams and lever together and preclude slippage, sec-

ondly to give finite adjustment. The assembly is supported in the switch and cam assembly by a shaft which is inserted through the center of the cam assemblies.

Also included are the windshield wiper speed and selector controls, gust lock control lever, dual elevator trim tab control wheels, speedbrake and parking/emergency brake control handles, flap control lever power levers friction knob and the fuel trim switches. On Aircraft having ASC 116, a landing gear emergency dump valve reset control is mounted on a bracket installed on the copilots side of the center pedestal console. On Aircraft 80-200, 322 and 323 and Aircraft having ASC 100, a cockpit thermostat element and blower motor are installed on a bracket on the copilots side of center pedestal console.

The radio control panel is located aft of the pedestal console and has space provisions for radio the miscellaneous panels and autopilot controls. The aileron and rudder trim tab controls are located aft of the radio control panel.

Removal of the side panels mounted on the pedestal provides access for servicing of switches and wire connections. Removal of the right side panel provides access to the bracket assembly and linkage for the landing gear emergency dump valve reset control when installed (ASC 116). It also provides access to cockpit thermostat element when installed (ASC 100).

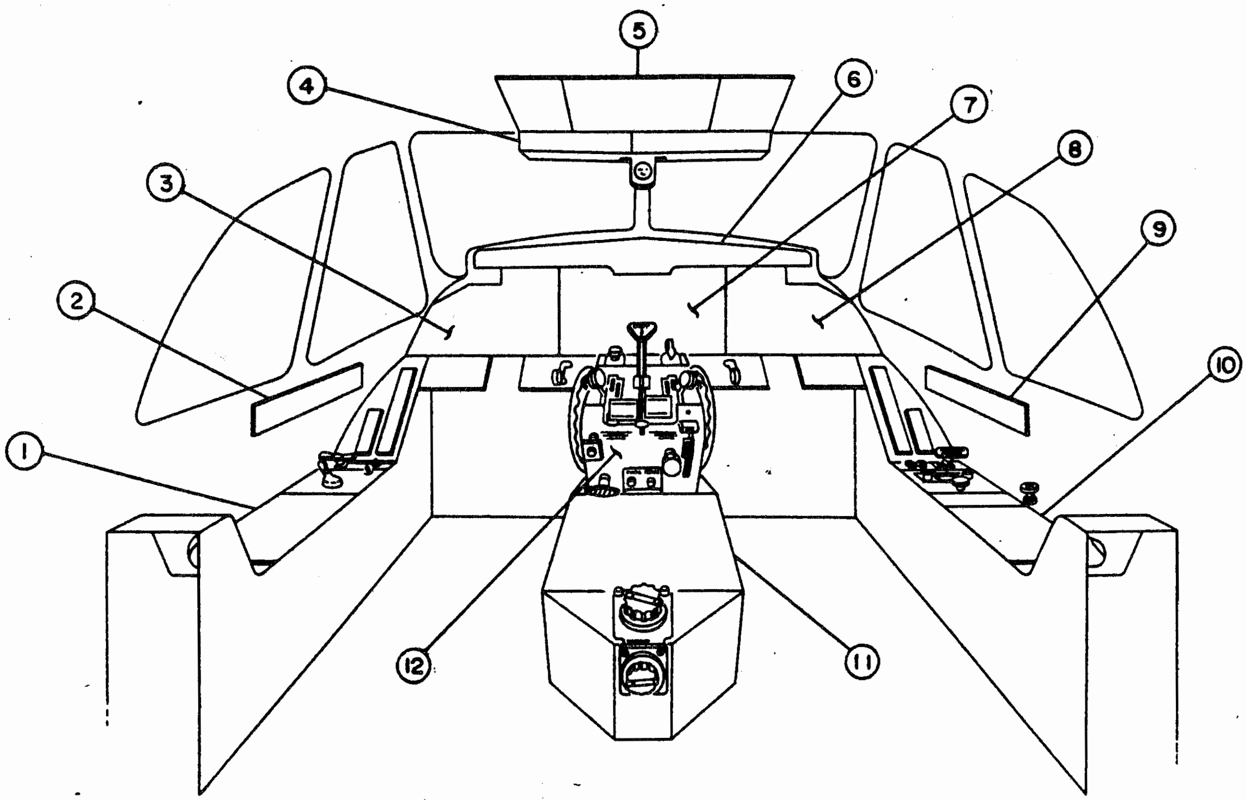
2. Operation

(See Figure 4)

Function and Operation of Pedestal Control Levers Microswitches and Cams are as follows:

FUNCTION OF SWITCH	ACTUATED BY	OPERATING CYCLE
1. Propeller Feather Circuit	Left HPFC	Operating Cycle Close circuit in feather position.
2. Propeller Auto Coarsening Circuit	Left HPFC	Close circuit when lever is in either position wherein fuel is ON.
3. Water/Methanol Circuit	Left HPFC	Open circuit in feather position.
4. Propeller Auto Coarsening Circuit	Left HPFC	Open circuit when lever is in FEATHER and in OFF positions.
5. Monitor Bus Power Control	Left Power Lever	Actuate switch at 10,500 rpm as rpm decreases.
6. Propeller Auto Coarsening Circuit	Left Power Lever	Close circuit at 11,500 rpm as rpm increases.
7. Landing Gear Warning Horn	Left Power Lever	Close Circuit at 11,000 rpm as rpm decreases.
8. Gearbox Low Oil Pressure Warn Light (LH)	Left Power Lever	Close circuit at 12,000 rpm as rpm increases.
9. Propeller Auto Coarsening Circuit	Fine Pitch Lock Selector	Close circuit with lever in FLIGHT position.
10. Propeller Flight Fine Pitch Lock Circuit	Fine Pitch Lock Selector	Close circuit with lever GROUND position. Open circuit when lever has traveled 20° toward FLIGHT position.
11. Propeller Auto Coarsening Circuit	Fine Pitch Lock Selector	Close circuit with lever in FLIGHT position.
12. Propeller Flight Fine Pitch Lock Circuit	Fine Pitch Lock Selector	Close circuit with lever GROUND position. Open circuit when lever has traveled 20° toward FLIGHT position.
13. Monitor Bus Power Control	Right Power Lever	Actuate switch at 10,500 rpm as rpm decreases
14. Propeller Auto Coarsening Circuit	Right Power Lever	Close circuit at 11,500 as rpm increases
15. Landing Gear Warning Horn	Right Power Lever	Close circuit at 11,000 rpm as rpm decreases.
16. Gearbox Low Oil Pressure Warn Light (RH)	Right Power Lever	Close circuit at 12,000 rpm as rpm increases.

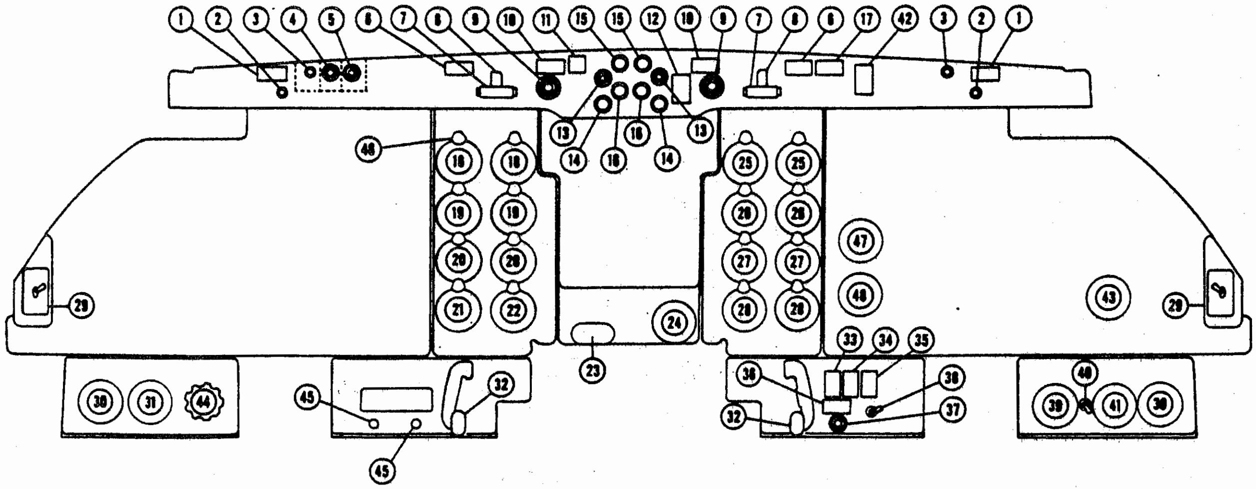
FUNCTION OF SWITCH	ACTUATED BY	OPERATING CYCLE
17. Propeller Feather Circuit	Right HPFC	Close circuit in feather position.
18. Propeller Auto Coarsening Circuit	Right HPFC	Close circuit when lever is in either position when fuel is ON.
19. Water/Methanol Circuit	Right HPFC	Open circuit in feather position.
20. Propeller Auto Coarsening Circuit	Right HPFC	Open circuit when lever is in FEATHER and in OFF position.
21. Flap Control	Flap Lever	Close circuit in UP position only.
22. Flap Control	Flap Lever	Close circuit in TAKEOFF position only.
23. Flap Control	Flap Lever	Close circuit in APPROACH position only.
24. Flap Control	Flap Lever	Close circuit in DOWN position only.



1. Pilots Side Console
2. Pilots Lighting Control Panel
3. Instrument Assembly-Pilots Panel*
4. Lower Overhead Panel
5. Upper Overhead Panel
6. Eyebrow Panel
7. Instrument Assembly-Center Panel
8. Instrument Assembly-Copilots Panel*
9. Copilots Lighting Control Panel
10. Copilots Side Console
11. Radio Control Panel*
12. Center Pedestal Console

*Space Provision Only

Flight Compartment, Looking Forward
Figure 1.



1. Master Warning Light (2)
2. Warning Light Reset Button (2)
3. Warning Light Test Button (2)
4. Fire Dectection Test Button
5. Fuel Gage Test Button
6. Aft Nacelle Fire Warning Light (2)
7. Fire Pull Switch (2)
8. Nacelle Fire Extinguisher Switch (2)
9. Feathering Pump Button (2)
10. Feather Pump Light (2)
11. Propeller Synchronize Switch
12. Propeller Flight Safety Lock Switch
13. Cruise Lock Out Light (2)
14. Below Flight Fine Pitch Lock Light (2)
15. Cruise Pitch Light (2)
16. Flight Fine Pitch Lock Light (2)
17. APU Fire Warning Light
18. Tachometer (2) (See Note 1)
19. Torque Indicator (2)
20. Turbine Gas Temperature Indicator (2) (See Note 1)
21. Water / Methanol Quantity Indicator (See Note 2)
22. Outside Air Temperature Indicator (See Note 2)
23. Fuel Datum Indicator
24. Flap Position Indicator
25. Oil Pressure Indicator (2)
26. Oil Temperature Indicator (2)
27. Fuel Flow Indicator (2)
28. Fuel Quantity Indicator (2)
29. Static Pressure Selector Switch (2)
30. Oxygen Flow Indicator (2)
31. Oxygen Pressure Indicator
32. Landing Gear Selector Handle (2)
33. Left Wheel Indicator Light
34. Nose Wheel Indicator Light
35. Right Wheel Indicator Light
36. No Horn
37. Horn Cutoff Switch
38. Wheel Light Intensity Switch
39. Emergency Hydraulic Pressure Indicator
40. Emergency Hydraulic Pump Switch
41. Normal Hydraulic Pressure Indicator
42. APU Fire Extinguisher Switch
43. Hydraulic Brake Accumulator Pressure Gage (Mounted on Panel)
44. Oxygen Shutoff Valve
45. Speedbrake Indication Lights (See Note 3)
46. Lighting Fixture (See Note 4)
47. (See Note 2)
48. (See Note 2)

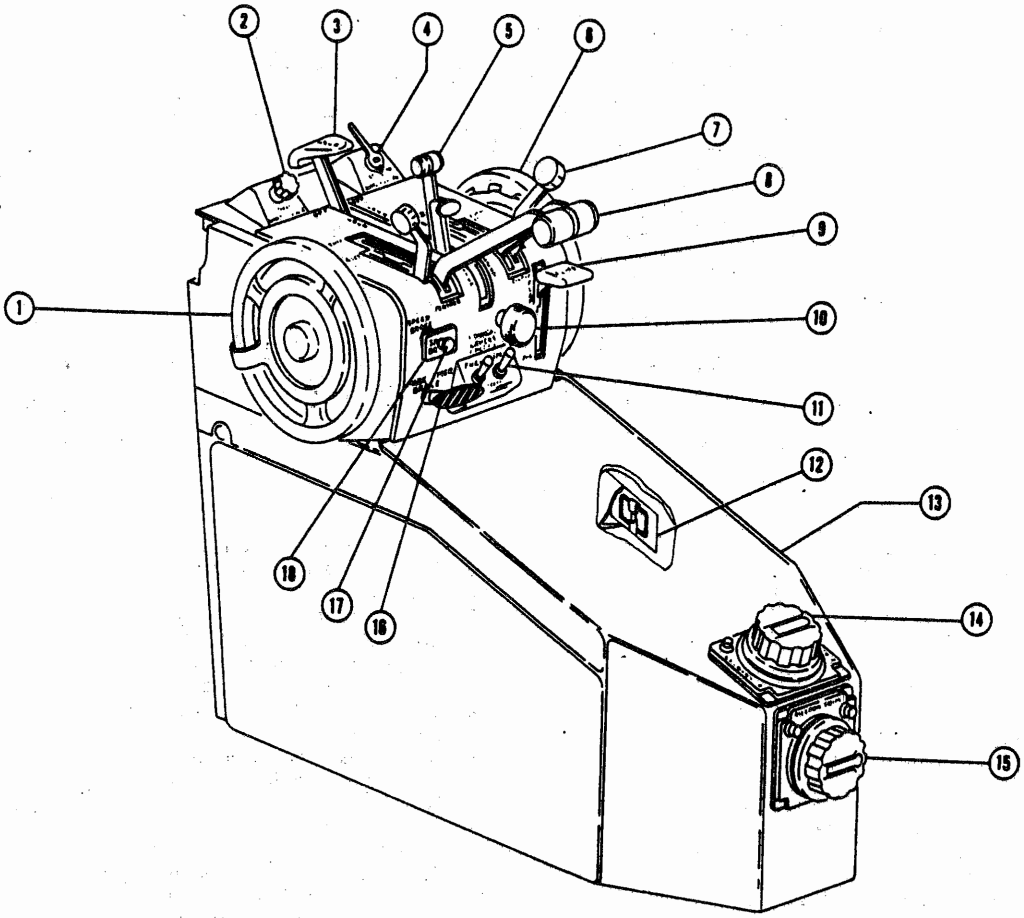
NOTE: On Aircraft 87-200 including 322, 323 and Aircraft 1-86 including 114 having ASC 115 Tachometer and Turbine Gas Temperature indicators are interchanged.

On aircraft having ASC 114: Water / Methanol Quantity is located at position 48, Outside Air Temperature Indicator is located at position 47, Fuel Flow Indicators located at positions 21 and 22, Fuel Temperature Indicators (2) located at positions 27.

Speedbrake Indication Lights are on Aircraft 88-200 including 322, 323 and Aircraft 1-87 including 114 having ASC 108A, part I, only.

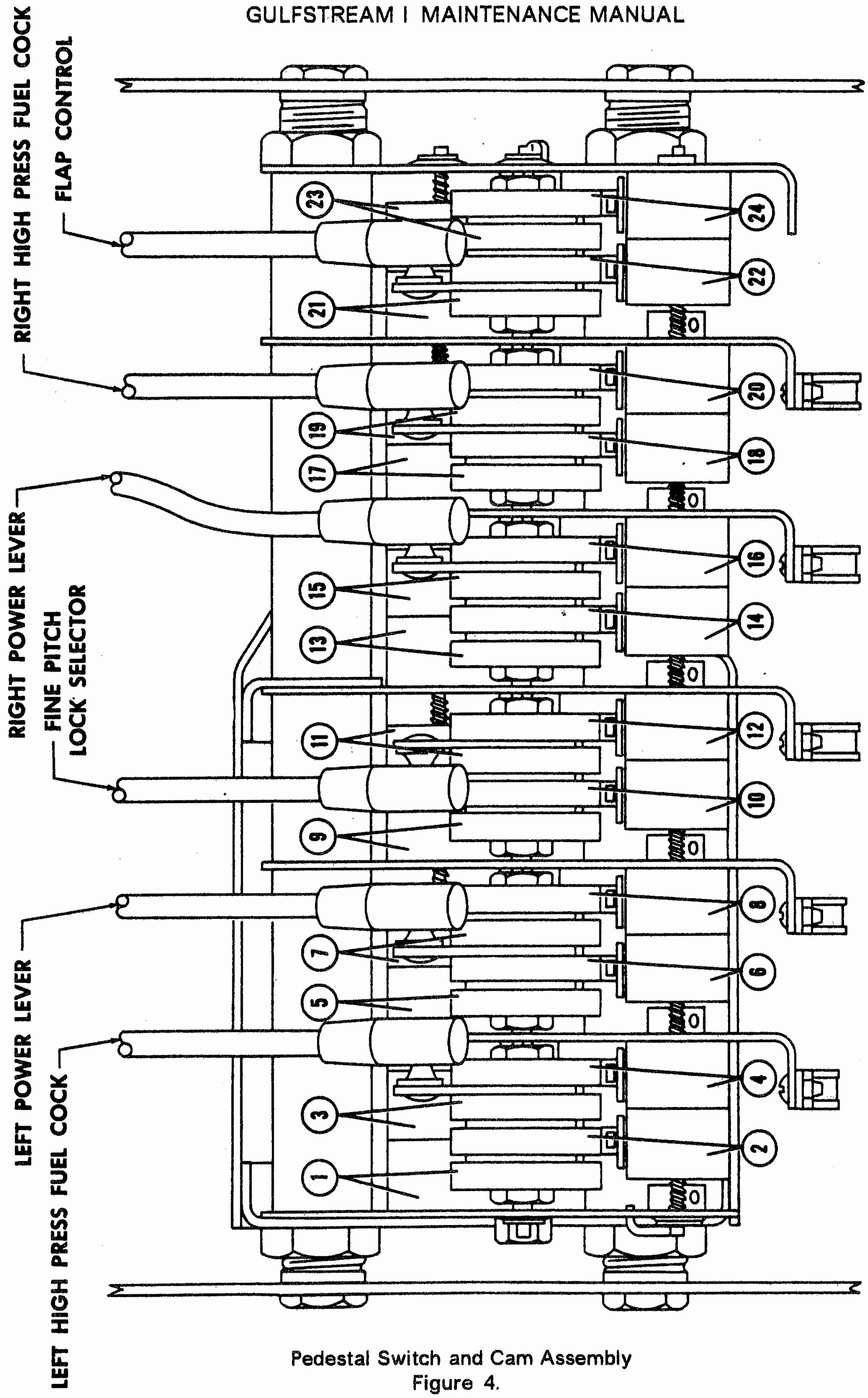
Lighting fixtures are on Aircraft 87-200 including 322, 323 and Aircraft 1-86 including 114 having ASC 115 only.

Pilot and Copilot Instrument and Eyebrow Panels
Figure 2.



1. Pilots Elevator Trim Tab Control Wheel
2. Windshield Wiper Speed Control
3. Gust Lock Control Lever
4. Windshield Wiper Selector Control
5. Fine Pitch Lock Selector Control Lever
6. Copilots Elevator Trim Tab Control Wheel
7. High Pressure Fuel Cock (2)
8. Power Lever (2)
9. Flap Control Lever
10. Power Lever Friction Knob
11. Fuel Trim Switches
12. Landing Gear Emergency Dump Valve Reset Control
(Aircraft 87-200 including 322, 323 and Aircraft 1-86 including 114 having ASC 116)
13. Radio Control Panel*
14. Rudder Trim Tab Control
15. Aileron Trim Tab Control
16. Parking and Emergency Brake Control Handle
17. Speedbrake Control Release Button
18. Speedbrake Control Handle

Center Pedestal Console/Radio Control Panel
Figure 3.



Pedestal Switch and Cam Assembly
Figure 4.

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INSTRUMENTS — MAINTENANCE PRACTICES

1. Pilot and Copilot Instrument Panels — Removal / Installation

A. Removal

(1) De-energize all power buses.

(2) Remove face mounting screws.

(3) Disconnect all electrical and plumbing connections.

(4) Partially rotate panels forward until they can be lifted up and aft for removal.

B. Installation

(1) Ensure instruments, switches, lights and static selector valves are serviceable and properly installed.

(2) Identify, uncap and connect instrument hose fittings.

(3) Connect all electrical connectors.

NOTE: Ensure all lines and electrical connectors are properly connected, and that all clamps and clips are properly installed.

(4) Push panel forward and tighten panel face mounting screws.

2. Center Pedestal Console/Radio Control Panel — Removal / Installation

A. Removal

(1) De-energize all power buses.

(2) Remove access panels on each side of the pedestal to gain access in order to disconnect electrical connections to radio panels on the pedestal.

NOTE: Removal of the pedestal is not recommended.

B. Installation

(1) Position radio panels on the pedestal and install electrical connectors.

(2) Install access panels removed for access.

(3) Perform an operational test.

3. Pedestal Bottom Microswitches — Removal / Installation

(See Figure 201)

NOTE: The following procedures involve the use of the following special Gulfstream Aerospace tools which are part of the packed parts delivered with the aircraft.

TOOL No.	QUANTITY	USE
TW 7744-F	2	Wrench, Switch Adjustment
TW 7743-F	1	Allen Wrench, Switch Adjustment

A. Removal

NOTE: Procedure will vary according to the accessibility of the switches.

(1) Insert rigging pin through sectors, locking pedestal levers in full aft position.

(2) Loosen setscrews in each of six switch adjusting nuts.

(3) Remove cotter pin and washer from end of hooked rod.

(4) Remove cotter pin and washer from that end of straight rod which is nearest defective switch.

NOTE: Insert a spare rod from the opposite end to prevent the nuts and springs from falling out of the assembly as hooked rod is removed.

(5) Pull hooked rod outboard until defective switch is free.

(6) Disconnect wires from defective switch.

(7) Pull straight rod out and remove defective switch.

B. Installation

(1) Connect wires to switch.

(2) Push both rods through switches, nuts and compression springs. Engage hook with hole.

(3) Install washers and cotter pins.

(4) Set switches for proper cam contact and tighten setscrews in adjusting nuts.

NOTE: Ensure that nuts are locked and do not rotate.

(5) Remove rigging pin and check operation of entire switch bank.

4. Pedestal Forward Microswitches -- Removal / Installation

(See Figure 201)

A. Removal

(1) Insert rigging pin through sectors, locking power lever in full aft (minimum rpm) position.

(2) Remove autopilot trim servomotor from forward side of switch and cam assembly.

(3) Mark aft face of cams to indicate proper position of cams in assembly.

(4) Remove cotter pins and washers from straight rod and remove completely.

NOTE: All cams will hang freely from their connecting rods.

(5) Remove four supporting rod assemblies. Switch assembly is now free and can be rested on floor.

(6) Loosen setscrews in each of six switch adjusting nuts.

(7) Remove cotter pin and washer from hooked rod.

(8) Remove cotter pin and washer from that end of straight rod which is closest to defective switch.

(9) Disconnect wires from defective switch.

NOTE: Insert a spare rod from the opposite end to prevent the nuts and springs from falling out of the assembly as hooked rod is removed.

(10) Pull hooked rod outboard until defective switch is free.

(11) Pull straight rod out and replace defective switch.

B. Installation

(1) Connect wires to switch.

(2) Push hooked rod through switches, nuts and compression springs. Engage hook with hole.

(3) Install washers and cotter pins.

(4) Set switches for proper cam contact. Tighten setscrews in adjusting nuts.

NOTE: Ensure that nuts are locked and do not rotate.

(5) Locate complete assembly in its proper position in pedestal.

(6) Reinstall support rod assembly and shaft support.

(7) Install all cam assemblies, observing alignment of mark made during removal.

(8) Push straight rod through complete assembly and install washers and cotter pins.

(9) Install bottom rod assembly and shaft support.

(10) Remove rigging pin and check out operation of switch bank.

(11) Install autopilot trim servomotor.

NOTE: If a potted switch is removed it is easier to connect the wires and pot connections before installing.

5. Microswitch Adjustment

(See Figure 201)

A. Determine which switch requires adjustment.

NOTE: Adjusting any one switch will affect the adjustment of the switch on either side of it.

B. Locate adjusting nut closest to it.

C. Loosen setscrews approximately one turn.

D. Rotate adjusting nut approximately 1/8 turn using special wrench in pack supplied with aircraft.

E. Check out switch for proper actuation.

F. Tighten setscrews in adjusting nut.

NOTE: Ensure that nut is tight and does not rotate.

6. Microswitch Cam Adjustment

(See Figure 201)

A. Draw an index line across four cams and onto lever.

B. Loosen nut on threaded camshaft bolt approximately 1/2 turn using special wrench supplied with aircraft.

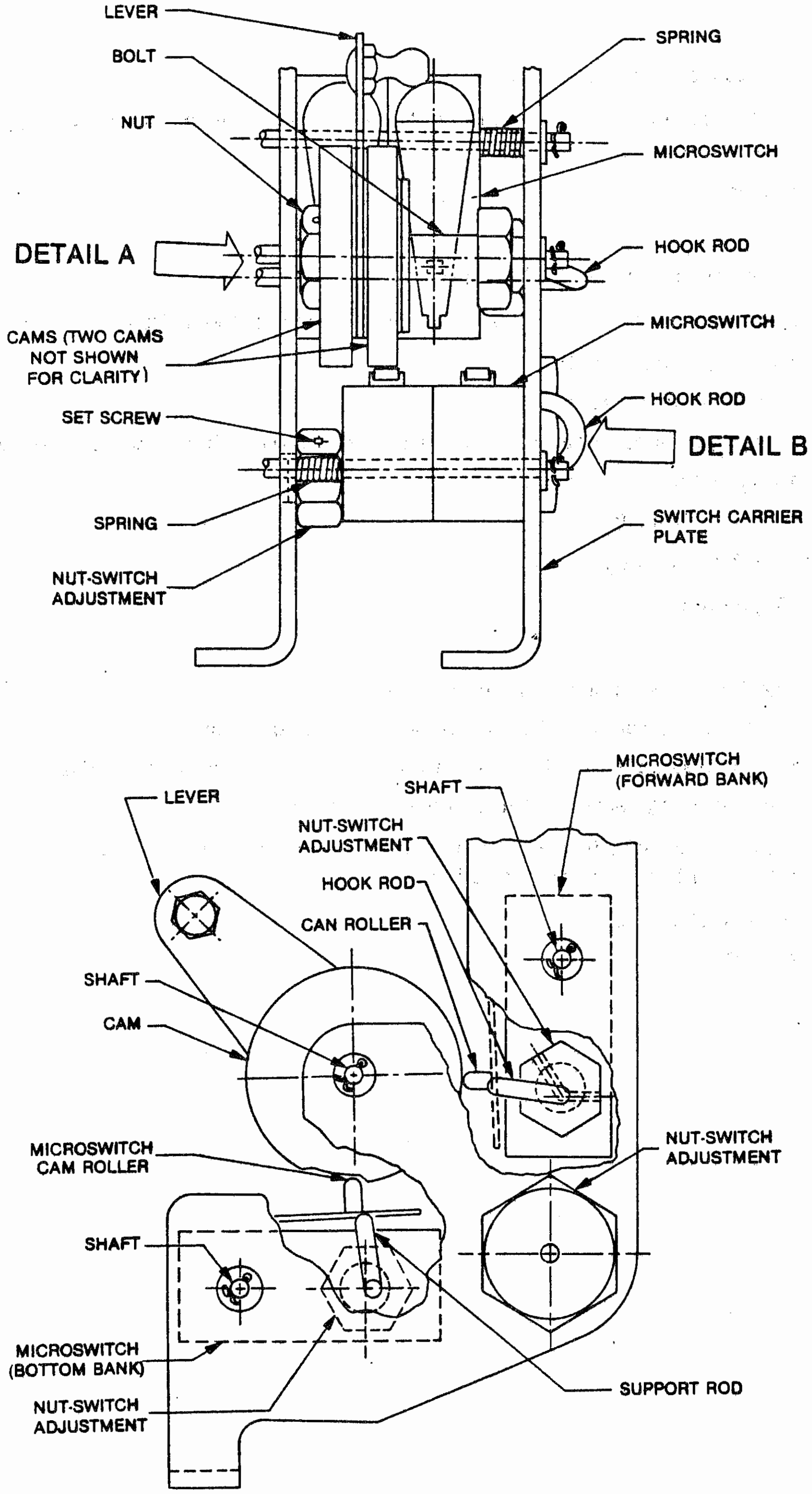
C. Grip cam that requires adjustment and separate it from its adjacent cam by applying side pressure.

D. Rotate as required being careful not to disturb other cams.

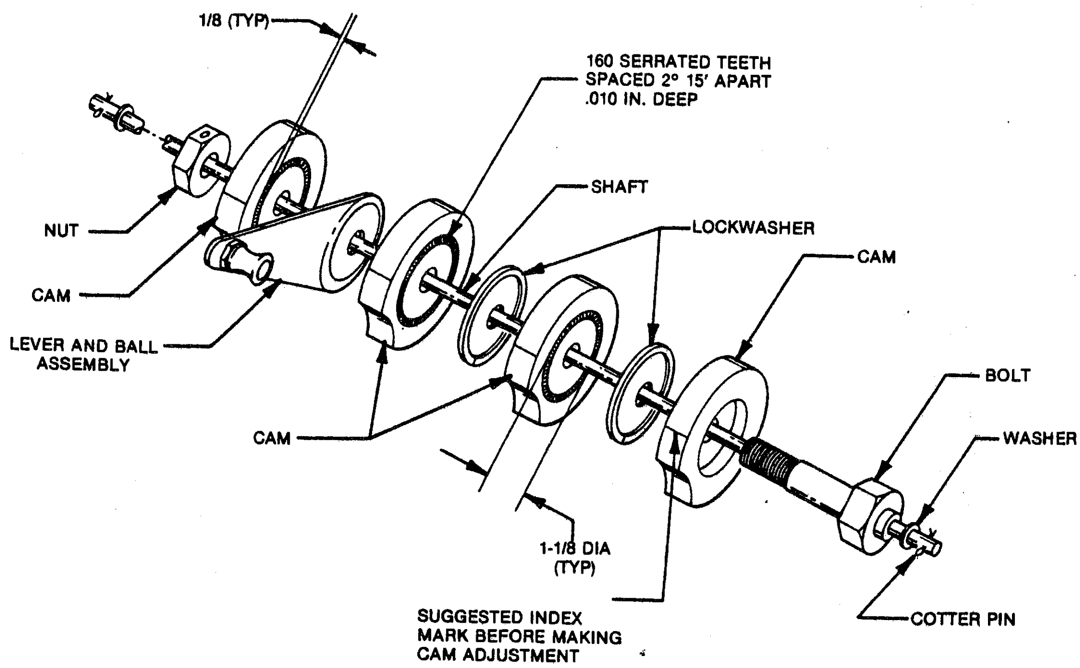
NOTE: Side movement of 0.010 inch is sufficient to disengage serrations. The 160 serrations produce an adjustment of 2° 15′ per serration.

E. Tighten nut on camshaft bolt.

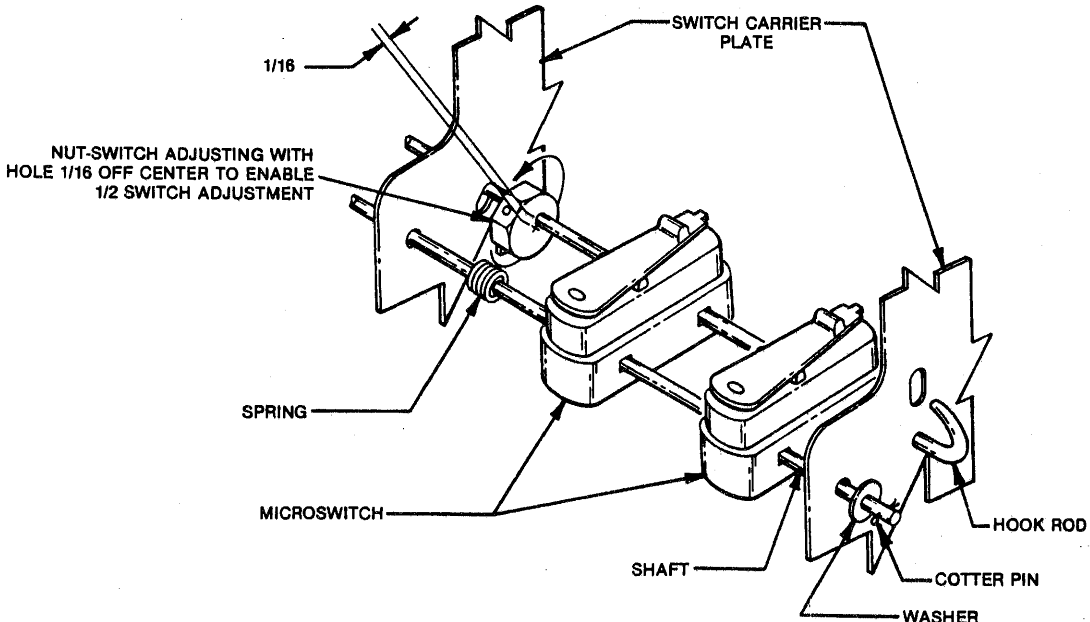
F. Check all cam settings.



Flap Cam Assembly Shown
(Typical for all cam adjustments
Figure 201 (Sheet 1 of 2).



DETAIL A



DETAIL B

Flap Cam Assembly Shown
(Typical for all cam adjustments
Figure 201 (Sheet 2 of 2).

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PILOT AND COPILOT UPPER AND LOWER OVERHEAD AND SIDE CONSOLE PANELS — DESCRIPTION / OPERATION

1. **Pilot and Copilot Upper and Lower Overhead Panels**

(See Figure 1)

The upper overhead panel is secured to the aircraft structure. The panel contains various switches and instruments grouped together on it. These switches and instruments, from left to right, are: deicing control switches, lights, gages, electrical, lighting, APU control panel and the temperature control and pressurization panel. The lower overhead panel is mounted below the base of the upper panel.

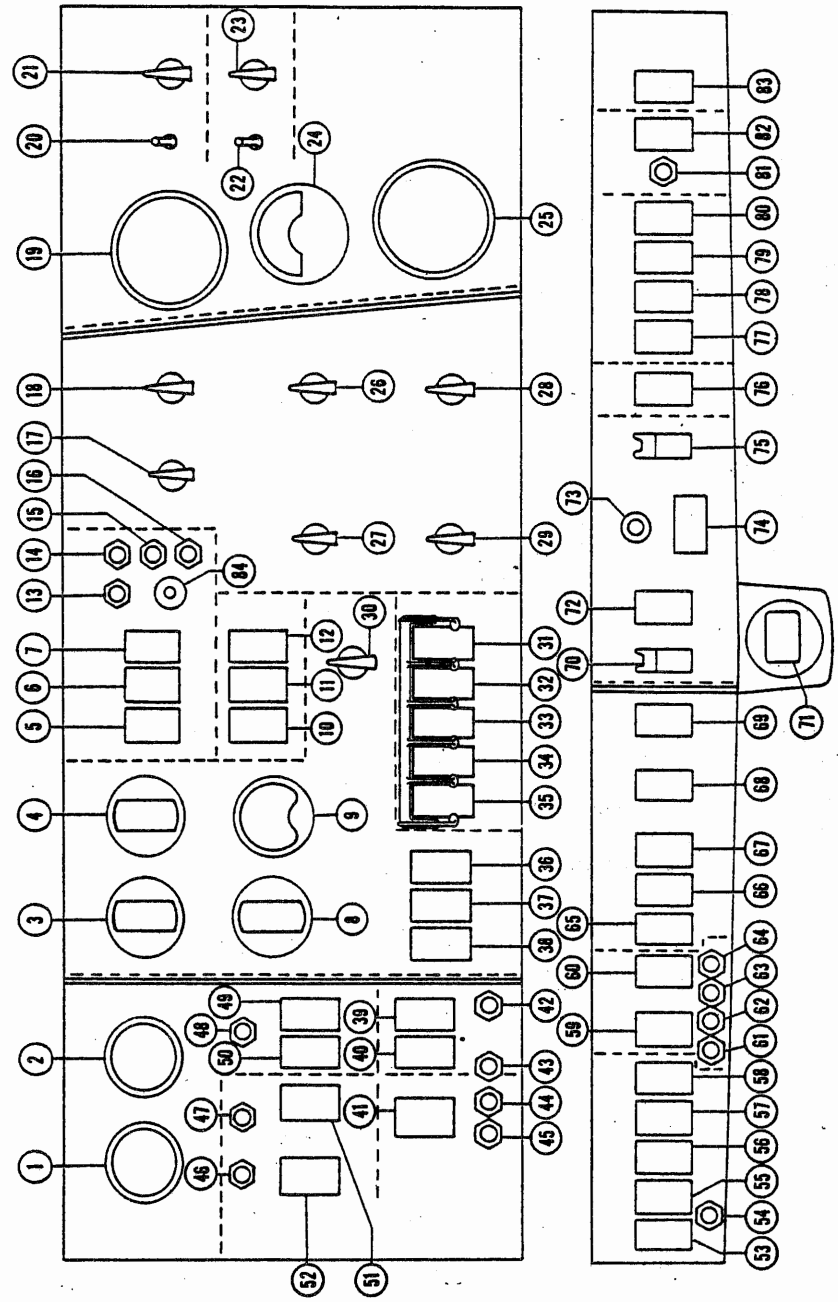
2. **Side Console Panels**

(See Figure 2

The side console panels are located to the left of the pilots and to the right of the copilots, respectively. Located on the left side console panel are the pilots master caution light display panel, propeller brake control lever, pilots auxiliary hydraulic pump switch, nosewheel power steering tiller bar, nose power steering switch, pilots oxygen mask and hose stowage container and regulator. On the right side console panel are the copilots master caution light display panel, emergency landing gear extension handle, air conditioning vent, blower on-off switches, automatic pressurization controller, manual pressurization control valve, emergency wing and tail deice switches, copilots oxygen mask, hose stowage container and regulator, emergency flap control lever, stall warning test switch, spare lights panel and the spare fuse panel. A lighting control panel is installed above each side console and is secured to the aircraft structure. Rheostats installed on these panels control instrument and side console panel lighting. Also included is a map floodlight switch on each lighting control panel (Aircraft 1-86 and 114).

Aircraft 87-200, 322 and 323 and Aircraft 1-86 and 114 having ASC 115 map light switches are of the push-button type and are located on the fairings above the panels. Additional rheostats are installed in place of these switches. The gooseneck type map lights are installed forward of the switches in this modification.

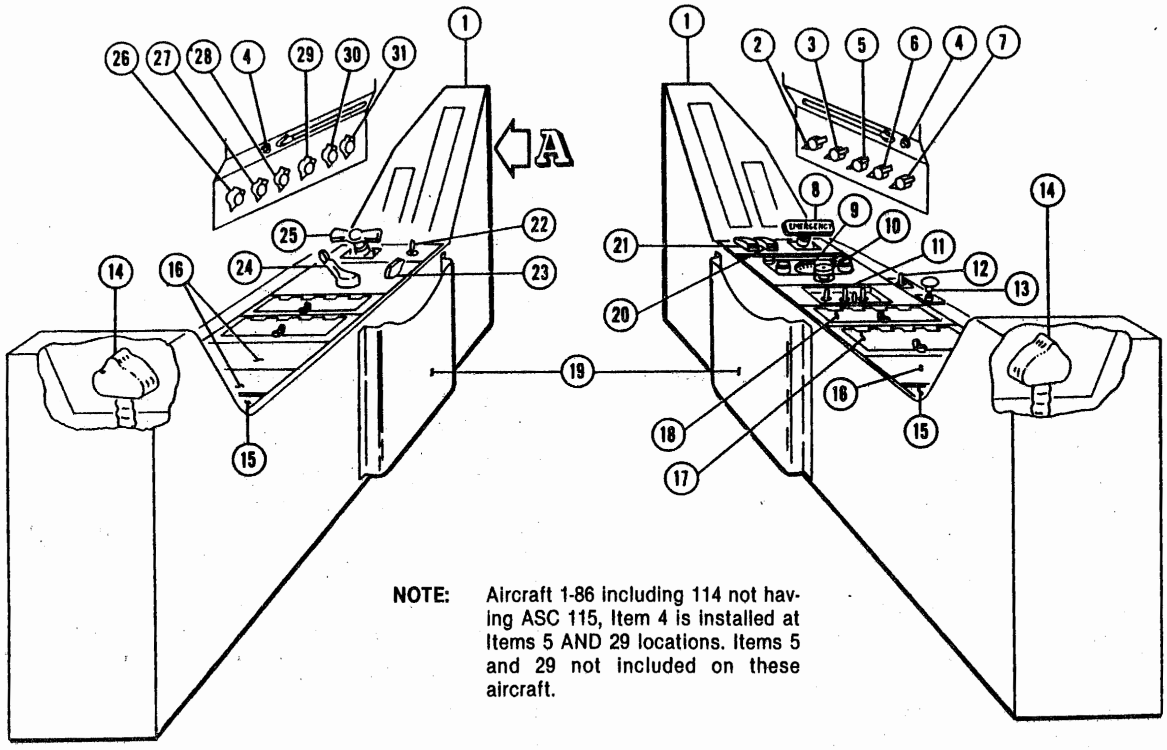
The upper overhead, lower overhead, and lighting control panels are removed by removing panel face mounting screws. Side console panels, with the exception of the nosewheel steering panel, are mounted to the console with Dzus fasteners for easy access and servicing.



Pilot and Copilot Upper and Lower Overhead Control Panel
Figure 1 (Sheet 1 of 2).

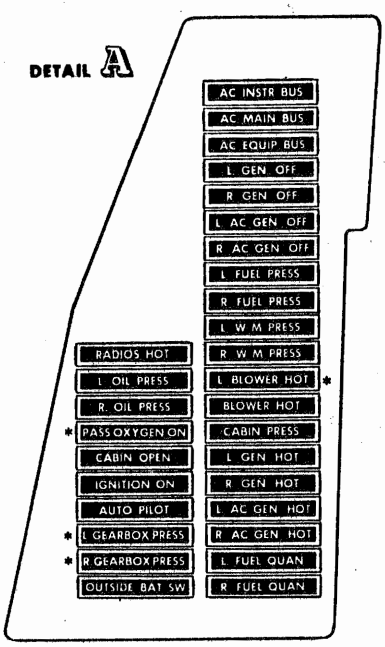
1. Wing and Tail Deicing Pressure Gage
2. Wing and Tail Deicing Suction Gage
3. Left Generator Voltammeter
4. Right Generator Voltammeter
5. APU Air Switch
6. APU Generator Switch
7. APU Master Switch
8. APU Generator Ammeter / DC Bus Voltmeter
9. AC Voltmeter
10. Radar Bus Switch
11. Main Bus Switch
12. Essential Bus Switch
13. APU Oil Pressure Light
14. APU Start Light
15. APU Operating RPM Light
16. APU Generator Off Light
17. Pedestal Flood Light Rheostat
18. Overhead Flood Light Rheostat
19. Differential Pressure and Cabin Altitude Indicator
20. Cabin Temperature Increase Selector
21. Cabin Temperature Rheostat
22. Cockpit Temperature Increase Selector
23. Cockpit Temperature Rheostat
24. Cabin Temperature Gage
25. Cabin Rate of Climb Indicator
26. Eyebrow and Lower Overhead Edge Light Rheostat
27. Overhead Panel Edge Light Rheostat
28. Pedestal Radio Edge Light Rheostat
29. Pedestal Edge Light Rheostat
30. AC Voltage Selector Switch
31. Electrical Master Right Alternator Switch
32. Electrical Master Right Generator Trip Switch
33. Electrical Master Battery Switch
34. Electrical Master Left Generator Trip Switch
35. Electrical Master Left Alternator Switch
36. Supervisory Right Generator Control Switch
37. Supervisory External Power Switch
38. Supervisory Left Generator Control Switch
39. Right Fuel Filter Deice Switch
40. Left Fuel Filter Deice Switch
41. Pitot Heat Switch
42. Right Fuel Filter Heat Indicator Light
43. Left Fuel Filter Heat Indicator Light
44. Right Pitot Heat Light
45. Left Pitot Heat Light
46. Left Engine and Propeller Deice Light
47. Right Engine and Propeller Deice Light
48. Tail Boot Indicator Light
49. Wing and Tail Deice Switch
50. Wing and Tail Deice Rate Switch
51. Right Engine and Propeller Deice Rate Switch
52. Left Engine and Propeller Deice Rate Switch
53. Left Landing Light Switch
54. Landing Lights Extended Indicator Light
55. Right Landing Light Switch
56. Left Landing Light EXTEND-RETRACT Switch
57. Right Landing Light EXTEND-RETRACT Switch
58. Taxi Light Switch
59. Windshield Heat NORM-EMERG Switch
60. Windshield Heat DEICE-DEFOG Switch
61. Left Direct Vision Panel Heat Indicator Light
62. Left Front Panel Heat Indicator Light
63. Right Front Panel Heat Indicator Light
64. Right Direct Vision Panel Heat Indicator Light
65. Navigation Lights Switch
66. Wing Lights Switch
67. Passenger Warning ST. BELTS-NO SMOKING Switch
68. Anticollision Lights Switch
69. White Override Lights Switch
70. Left Air Start Switch
71. Standby Compass
72. Starter Selector Switch
73. Starter Button
74. Engine Selector Start Switch
75. Right Air Start Switch
76. Door Safety Switch
77. Left Normal Boost Pump Switch
78. Left Auxiliary Boost Pump Switch
79. Right Normal Boost Pump Switch
80. Right Auxiliary Boost Pump Switch
81. Fuel Crossfeed Open Indicator Light
82. Fuel Crossfeed Switch
83. Water / Methanol Arming Switch
84. APU Start Button

Pilot and Copilot Upper and Lower Overhead Control Panel
Figure 1 (Sheet 2 of 2).



NOTE: Aircraft 1-86 including 114 not having ASC 115, Item 4 is installed at Items 5 AND 29 locations. Items 5 and 29 not included on these aircraft.

DETAIL A



1. Master Caution Light Display Panel (2)
2. Red Flight Instrument Lights Rheostat (Right)
3. Instrument Panel Red Flood Lights Rheostat (Right)
4. Map Light Switch (2) (See Note)
5. Instrument Panel White Flood Lights Rheostat (Right) (See Note)
6. Side Console Panel Red Edge and Flood Lights Rheostat (Right)
7. Side Console Panel White Flood Lights Rheostat (Right)
8. Emergency Gear Extension Handle
9. Cabin Pressure Controller
10. Manual Cabin Pressure Controller
11. Wing and Tail Emergency Deice Panel
12. Stall Warning Test Switch
13. Emergency Flap Control Handle
14. Oxygen Regulator
15. Oxygen Mask and Hose Stowage
16. Spare Panel
17. Spare Fuse Panel
18. Spare Lamp Panel
19. Map Case (2)
20. Ram Air Switch
21. Cabin Blower Switch
22. Emergency Hydraulic Pump Switch
23. Nose Wheel Power Steering Switch
24. Nose Wheel Steering Control
25. Propeller Brake Handle
26. Side Console Panel White Flood Lights Rheostat (Left)
27. Side Console Panel Red Edge and Flood Lights Rheostat (Left)
28. Instrument Panel White Flood Lights Rheostat (Left)
29. Instrument Panel Red Flood Lights Rheostat (Left) (See Note)
30. Red Flight Instrument Lights Rheostat (Left)
31. Red Engine Instrument Lights Rheostat

* Denotes optional items

Side Console and Lighting Control Panels
Aircraft 1-86 and 114 Having ASC 115 and Aircraft 86-200.
Figure 2.

PILOT AND COPILOT UPPER AND LOWER OVERHEAD AND SIDE CONSOLE PANELS — MAINTENANCE PRACTICES

1. **Pilot and Copilot Upper and Lower Overhead and Side Console — Removal / Installation**

A. Removal

(1) De-energize all power buses.

(2) Remove side console and lower overhead edge light panels.

(3) On upper and lower overhead and side console panels, remove panel face mounting screws or fasteners. Panels will be removed with removal of last mounting screws or fasteners.

(4) Identify, disconnect and cap all instrument and hose connections.

(5) Identify and disconnect all electrical connections.

B. Installation

(1) Ensure that instruments, switches and lights are serviceable and properly installed.

(2) Connect all electrical connectors.

NOTE: Ensure that all line and electrical connectors are properly connected, and that all clamps and clips are properly installed.

(3) Mount panel in proper location and tighten panel face mounting screws or fasteners.

(4) Install edge lighting panels and refasten.

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OUTSIDE AIR TEMPERATURE SYSTEM — DESCRIPTION / OPERATION

1. **Description**

The outside air temperature (OAT) indicator is a 2 inch, hermetically sealed unit, mounted on the left of the FUEL DATUM indicator in the center engine instrument panel. It operates from the essential dc bus system, and is calibrated from − 50°C to + 50°C.

NOTE: In certain installations, operators have elected to move this instrument to other locations on the panel. However, with ASC 114, the OAT indicator is moved to the copilots flight panel (if space is available). A visual check of your own aircraft will locate it for your particular installation.

The OAT bulb is a flush, free air, resistance type and is mounted in the lower part of the aircraft near the nose. Two wires connect the indicator to the bulb.

2. **Operation**

(See Figure 1

System power is supplied by the essential dc bus through the OAT circuit breaker on the pilots circuit breaker panel. One of the two wires to the bulb is grounded near the bulb as per manufacturers recommendations. Both bulb and indicator are plug disconnect.

Essentially, the system is a resistance bridge which is composed of fixed resistances in the indicator and a temperature sensitive resistance in the bulb. The temperature-resistance characteristic of the bulb is a positive coefficient, and it is affected by the outside air temperature, since the bulb is exposed to ambient air.

3. **Emergency DC Operation — Effect on System**

The OAT is operative in emergency.

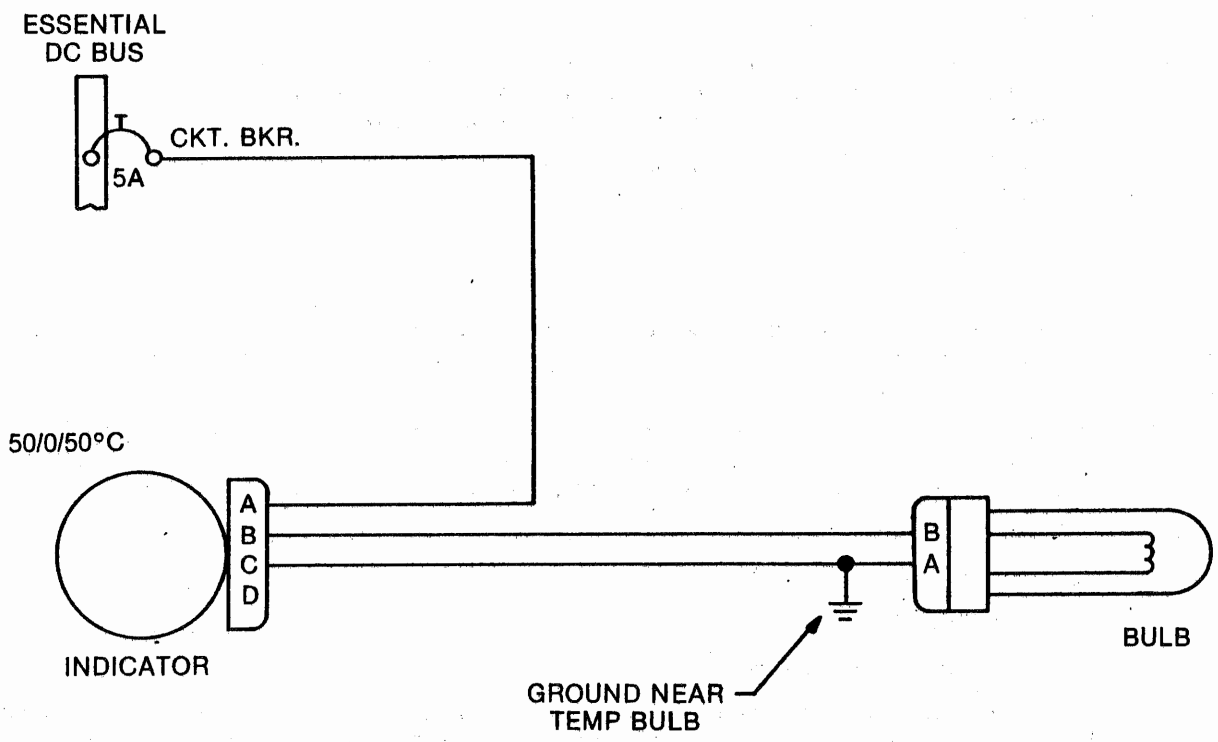
4. **Resistance vs. Temperature Chart**

(Good for OAT or cabin air temperature indicators)

NOTE: The following information is supplied for fault isolating purposes. It gives the approximate resistance of the bulb at surrounding temperature ranges from − 35 to + 120 degrees F, and also degrees C.

OHMS C	TEMP.	OHMS F
78.97	−35	78.25
80.57	−30	79.13
82.17	−25	80.01
83.77	−20	80.90
85.39	−15	81.79
87.04	−10	82.68
88.69	−5	83.58
90.38	0	84.48
92.08	5	85.39
93.80	10	86.30
95.55	15	87.22
97.31	20	88.14
99.11	25	89.07
100.91	30	90.00
101.65	32	90.38
102.75	35	90.94
104.60	40	91.89
106.49	45	92.84

OHMS C	TEMP.	OHMS F
108.39	50	93.80
110.33	55	94.76
112.28	60	95.73
114.27	65	96.71
116.27	70	97.70
118.31	75	98.70
119.13	77	99.11
120.36	80	99.70
122.45	85	100.71
124.55	90	101.73
126.70	95	102.75
128.85	100	103.78
131.05	105	104.82
133.26	110	105.86
135.51	115	106.91
137.78	120	107.97



Outside Air Temperature System — Schematic
Figure 1.

LIST OF EFFECTIVE PAGES

CHAPTER 32

LANDING GEAR

SECTION	PAGE	DATE	SECTION	PAGE	DATE
*32	i	September 11/92	*32-1-0	215	September 11/92
*32	ii	September 11/92	*32-1-0	216	September 11/92
*32	iii	September 11/92	*32-1-0	217	September 11/92
*32	iv	September 11/92	*32-1-0	218	September 11/92
*32	1	September 11/92	*32-1-0	219	September 11/92
*32	2	September 11/92	*32-1-0	220	September 11/92
*32	3	September 11/92	*32-1-0	221	September 11/92
*32	4	September 11/92	*32-1-0	222	September 11/92
*32	5	September 11/92	*32-2-0	1	September 11/92
*32	6	September 11/92	*32-2-0	2	September 11/92
*32	7	September 11/92	*32-2-0	201	September 11/92
*32	8	September 11/92	*32-2-0	202	September 11/92
*32-0	1	September 11/92	*32-2-0	203	September 11/92
*32-0	2	September 11/92	*32-2-0	204	September 11/92
*32-0	3	September 11/92	*32-2-0	205	September 11/92
*32-0	4	September 11/92	*32-2-0	206	September 11/92
*32-0	5	September 11/92	*32-2-0	207	September 11/92
*32-0	6	September 11/92	*32-2-0	208	September 11/92
*32-0	101	September 11/92	*32-2-0	209	September 11/92
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*32-0	206	September 11/92	*32-2-0	216	September 11/92
*32-0	207	September 11/92	*32-2-0	217	September 11/92
*32-0	208	September 11/92	*32-2-0	218	September 11/92
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*32-1-0	2	September 11/92	*32-3-0	2	September 11/92
*32-1-0	201	September 11/92	*32-3-0	201	September 11/92
*32-1-0	202	September 11/92	*32-3-0	202	September 11/92
*32-1-0	203	September 11/92	*32-3-0	203	September 11/92
*32-1-0	204	September 11/92	*32-3-0	204	September 11/92
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*** Pages deleted by Revision 48

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CHAPTER 32

LANDING GEAR

LANDING GEAR — DESCRIPTION / OPERATION

1. Description

The fully retractable tricycle type landing gear consists of two main gear, one in each engine nacelle, and a nose gear in the forward section of the fuselage. Main gear and nose gear struts are conventional nitrogen-hydraulic shock struts. Individual wheel braking is available for each of the main wheels. Nose wheels are steerable with a steering unit which can be mechanically disconnected without tools for towing purposes. Dual main magnesium alloy split type wheels mount dual 7.50 by 14 tubeless tires inflated to 110 psi at 33,600 pounds takeoff gross weight, 120 psi at 35,100 pounds takeoff gross weight and 123 psi at 36,000 pounds takeoff gross weight. The dual nose magnesium alloy split type wheels mount dual 6.50 by 8 tubeless tires inflated to 50 psi. Retraction and extension are accomplished by the main (normal) hydraulic power system. Emergency extension is provided by a 1900 to 2000 psi air pressure system. If hydraulic pressure fails, the emergency air system opens the clamshell doors and releases the landing gear. The nose gear is powered down, the main gear is free falling. The clamshell doors remain open after emergency extension.

A ground service valve has been incorporated in the nose wheel well to allow personnel to open the wheel well doors (clamshell) to facilitate ease of maintenance. The position indicating and warning system functions to provide positive indication of the position of the landing gear. A nutcracker (scissor switch) system is incorporated and furnishes on ground or in flight reference to those systems requiring this information.

A. Main Gear Component Location

(See Figure 1)

Unit	No. Per A/C	Location
Actuating Cylinder	2	Inboard side of main wheel well
Uplock Cylinder	2	Top forward side of main wheel well
Door Actuating Cylinder	2	Attached to forward portion of doors
Main Gear and Speedbrake Selector Valve	1	Top of nose wheel well, center line
Drag Brace and Downlock	2	Attached to aft side of main gear shock strut
Downlock Cylinder	2	Attached to aft side of main gear strut, in top part of drag brace
Timer Valve	4	Inboard Side of main wheel well, aft end of doors
Manual Reset Dump Valve	1	Right side of wheel well
Door Control Valve	2	Inboard side of main wheel well, fwd of strut
Shuttle Valve	4	Attached to door cylinders and uplock cylinders in wheel wells
Main and Nose Near Restrictors	3	One in nose wheel well and one in main wheel well, inboard of nacelle
Main and Nose Gear Check Valves	12	Various line locations
Main and Nose Gear Line Filters	18	At timer valves and door control valves
Main Gear Pressure Relief Valves (Aircraft 88 - 200, 322 and 323 excluding 114 and Aircraft having ASC 140 part II)	2	One in each main wheel well

NOTE: With exception of main gear and speedbrake selector valve, all of the components of left main gear are opposite of the components of the right main gear.

B. Nose Component Location

(See Figure 2)

Unit	No. Per A/C	Location
Actuating cylinder	1	Side of nose wheel strut
Uplock Cylinder	1	Top center of nose wheel well
Door Actuating Cylinder	1	Top fwd of nose wheel well
Selector Valve	1	Top of nose wheel well
Timer Valve	2	One above each end of nose gear door cylinder
Door Control Valve	1	Forward of nose hinge point
Thermal Relief Valve	1	Nose wheel well, aft right side
Shuttle Valve	3	Near or attached to gear cylinder, door cylinder,and uplock cylinder in nose wheel well

C. Emergency Extension Component Location

Unit	No. Per A/C	Location
Air Bottle	1	Nose Wheel Well, left forward side
Air Filler Valve	1	Nose wheel well right forward side
Air Gage	1	Nose wheel well right forward side
Air Release Valve	1	Nose wheel well right forward of air bottle
Check Valve	1	Between air release valve and air bottle
Shuttle Valve	1	Nose wheel well right forward side of bulkhead at approximately Fuselage Station 85
Dump valve (159SCH123) Installed in Air Bottle	1	Nose wheel well right forward of air bottle

2. Operation

A. Normal Operation

Each of the three gear has three actuating hydraulic cylinders (extension-retraction, uplock release-locking, and door opening-closing). The landing gear is extended and retracted by the pilot LDG GEAR handle, located below the instrument panel, on the pilot right skirt panel just to the left of the center pedestal console, or by the copilot LDG GEAR handle, located below the instrument panel, on the copilot left skirt panel just to the right of the center pedestal console. Both landing gear control handles are mechanically connected and operate in unison. They control two selector valves, one nose gear selector valve, and one main gear and speedbrake selector valve. Initially with the landing gear down, the main and nose gear doors are closed. When landing gear UP is selected, the main and nose gear struts retract forward to the up and locked position. The main and nose gear clamshell doors close, completing the up cycle. When landing gear DOWN is selected, the main and nose gear clamshell doors open, the main and nose gear struts extend to the down and locked position. The main and nose gear clamshell doors close, completing the down cycle.

CAUTION: IF ANY UPLOCKS ARE INADVERTENTLY CLOSED WITH RESPECTIVE GEAR DOWN, THAT GEAR WILL NOT RETRACT. A VISUAL PREFLIGHT CHECK MUST BE MADE TO ENSURE THAT ALL THREE UPLOCKS ARE OPEN.

CONDITION OF SHOULDER PIN (P/N 174519), LOCATED INSIDE STRUT, MAY ALLOW BEARING ADAPTER TO BACK OFF. WHICH, IN TURN, ALLOWS THE PISTON / AXLE ASSEMBLY TO EXTEND, CAUSING RETRACTION DIFFICULTIES. REPLACEMENT OF SHOULDER PIN WILL RETURN GEAR TO NORMAL OPERATION.

NOTE: A safety lock solenoid is located behind the copilots inboard skirt panel. This solenoid prevents inadvertent movement of gear handle while the aircraft is on the ground. The solenoid receives its energizing signal from the left nutcracker switch. In case of malfunction, a finger tab is provided to manually release the lock.

The speedbrakes are extended by pulling a speedbrake handle on the pilots side of the control pedestal. This handle is mechanically linked to the normal landing gear control system but extends the main gear only. The nose gear remains up and locked with its clamshell doors closed.

The speedbrakes can be retracted by pushing the handle back to its stowed and latched position. If it is desired to go from speedbrake to normal gear down, the LDG GEAR handle is moved to DOWN. This brings only the nose gear down (since the main gear are already down) and at the same time mechanically returns the speedbrake handle to its stowed position.

B. Emergency Operation

In the event of a hydraulic system failure, the landing gear can be extended pneumatically by means of an emergency air system. Limiting airspeed for extending the landing gear by the normal or emergency air system is 193 knots. The emergency gear extension handle placarded EMER LDG GEAR, is located on the forward side of the copilots console panel. When the EMER LDG GEAR extension handle is pulled out, all gear clamshell doors open, (regardless of the position of the normal landing gear handles) and two main gear and nose gear extend.

With the EMER LDG GEAR extension handle operated, the landing gear system is isolated completely from normal hydraulic source by a valve which cannot be reset in air except on Aircraft 87 - 200, 322 and 323 and Aircraft having ASC 116. Air under pressure is directed through separate lines from the pneumatic storage bottle to:

(1) Open side of all three door actuating cylinders.
(2) Open side of all three door uplock cylinders.
(3) Down side of the nose gear actuating cylinder.
(4) Manual reset dump valve.
(5) Several valves are opened to return fluid displaced by the air to the reservoir.

When the handle is pulled, commitment is made for pneumatic extension. Total action takes approximately two seconds. As noted above, the nose gear is powered down and air pressure remains in the downside of the actuating cylinder. This acts to assist the nose gear downlock. Main gear is not powered down. The uplocks are released, weight of the main gear and wheels causes them to free fall until they are caught in the slipstream. The slipstream then drives the gear down and they are locked by the downlocks through their own mechanical action.

WARNING: AS NOTED ABOVE, THE MAIN GEAR FREE FALL. IF, IN NORMAL EXTENSION, ONE OR BOTH OF THE MAIN GEAR CLEARS THE DOORS, BUT DOES NOT FULLY EXTEND OR LOCK DOWN, RECYCLING WITH THE NORMAL GEAR SYSTEM MAY CLEAR THE FAULT. ONCE COMMITTED TO PNEUMATIC EXTENSION, GEAR CANNOT BE RETRACTED OR RECYCLED. (EXCEPT ON AIRCRAFT 87 - 200, 322 AND 323 AND AIRCRAFT HAVING ASC 116.) AFTER LANDING, ENSURE THAT NORMAL LANDING GEAR HANDLES ARE IN DOWN POSITION. WHEN THE EMERGENCY GEAR EXTENSION HANDLE IS RETURNED TO THE STOWED POSITION (DURING INSPECTION OR MAINTENANCE), THE AIR SUPPLY THAT BROUGHT THE GEAR DOWN WILL BE RELEASED TO ATMOSPHERE THROUGH A VENT LOCATED ON THE EXTERNAL SKIN AT THE FORWARD RIGHT SIDE OF THE NOSE WHEEL WELL. ALL PERSONNEL MUST KEEP CLEAR OF THIS VENT.

CAUTION: EMER LDG GEAR HANDLE MUST BE LEFT IN THE PULLED POSITION UNTIL AIRCRAFT IS PREPARED FOR INSPECTION OF MALFUNCTION AND/OR THE GEAR IS MADE SAFE. UNDER NO CIRCUMSTANCES IS THE HANDLE TO BE STOWED WHILE AIRCRAFT IS IN FLIGHT.

NOTE: Although the emergency system bypasses the normal landing gear control system, the LDG GEAR extension handle should be placed in the DOWN position for proper safety indications.

After emergency extension of the landing gear, all landing doors remain open.

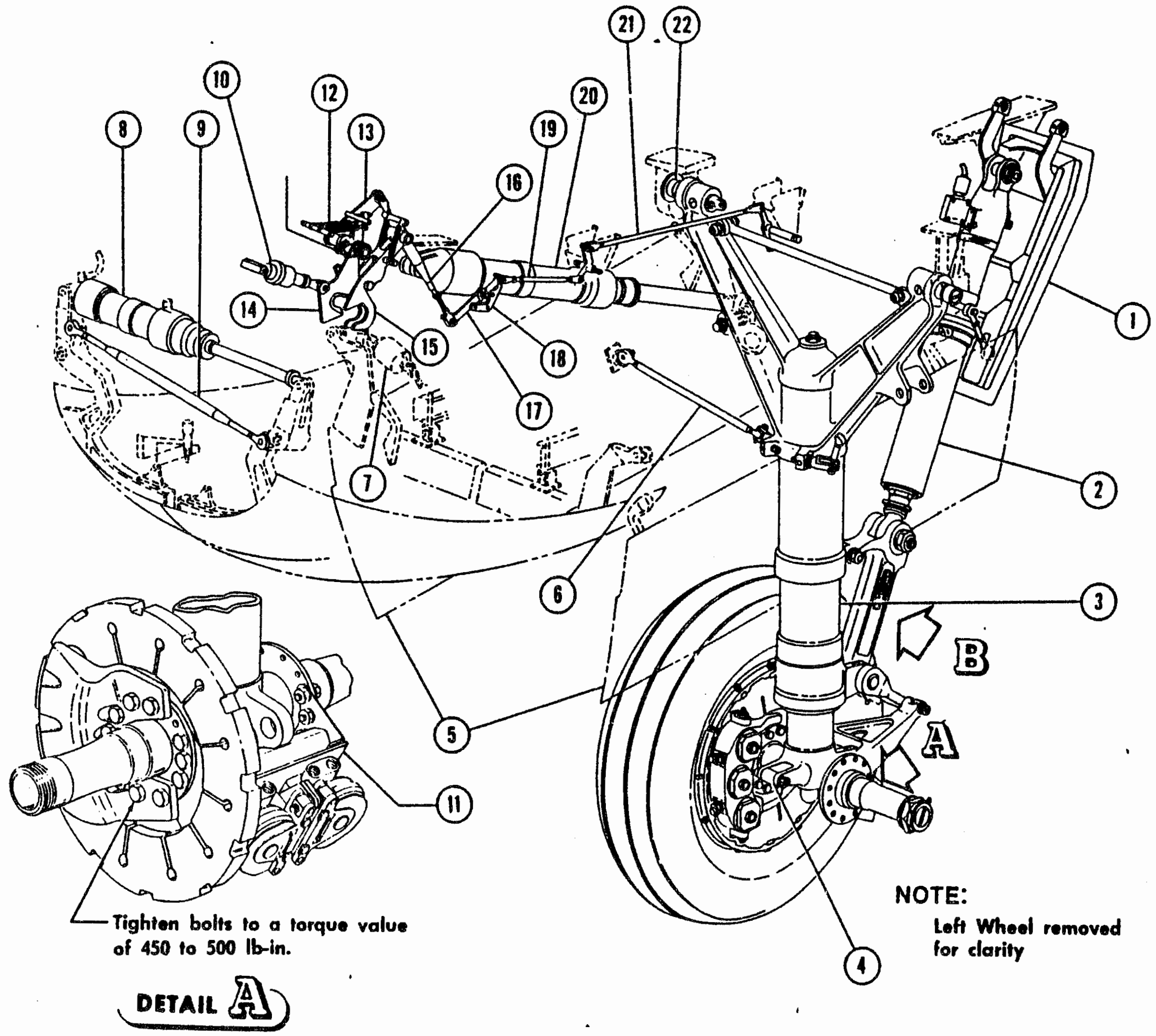
On Aircraft 87 - 200, 322 and 323 and Aircraft having ASC 116 a manual control is provided at the copilots position (right side of the control pedestal) to reset the landing gear emergency dump valve for normal hydraulic operation of the gear after emergency extension of the gear.

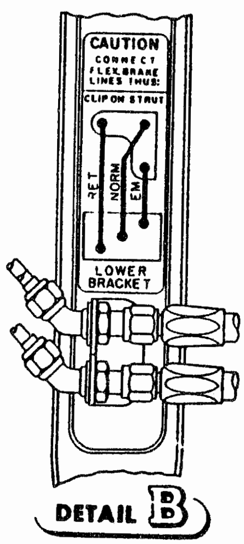
On Aircraft having ASC 116, should the landing gear be pneumatically extended in flight and subsequently the crew desires to return gear to UP position while still in flight, the following procedure must be taken:

(1) Push down emergency gear extension handle to normal position.
(2) Move LANDING GEAR HANDLE to UP.
(3) Pull dump valve reset control handle fully.
(4) As soon as gear starts to retract, move LANDING GEAR HANDLE to DOWN. This procedure allows proper orientation of shuttle valves within the system and will reduce the amount of hydraulic fluid lost overboard through hydraulic system vent (located on the right side of the nose).
(5) Place LANDING GEAR HANDLE in UP position and allow gear to fully retract.

CAUTION: A FULLY CHARGED NITROGEN BOTTLE WILL ALLOW APPROXIMATELY THREE EMERGENCY EXTENSIONS OF THE LANDING GEAR.

THE ABOVE PROCEDURE IS APPLICABLE ONLY FOR A FLIGHT RESET. FOR GROUND RESET AFTER PNEUMATIC EXTENSION AND FOR MAINTENANCE PRACTICES INVOLVED WITH THE PNEUMATIC EXTENSION AND FOR MAINTENANCE PRACTICES INVOLVED WITH THE PNEUMATIC EXTENSION SYSTEM, SEE SECTION 32-7-0.

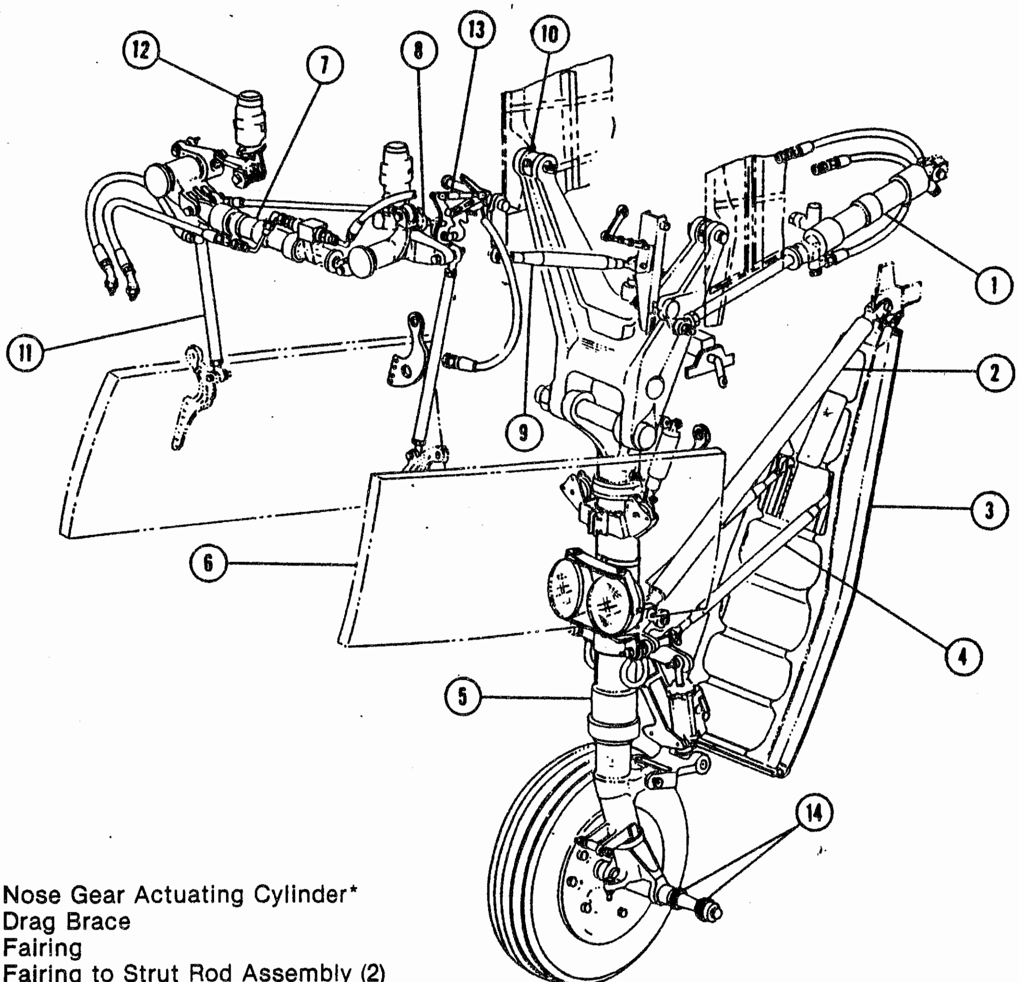




1. Fairing Assembly
2. Drag Strut Assembly
3. Shock Strut Assembly
4. Uplock Roller
5. Doors*
6. Door Rod Assembly
7. Timer Valve Bungee Assembly
8. Doors Actuating Cylinder*
9. Tie Rod Assembly
10. Uplock Actuating Cylinder*
11. Decelostat Assembly
12. Springs
13. Arm Assembly*
14. Hook Assembly*
15. Latch Assembly*
16. Door Sequence Spring Rod Assembly*
17. Crank Assembly
18. Neutral Pin Bracket
19. Door Timer Rod Assembly*
20. Gear Actuating Cylinder*
21. Door Timer Rod Assembly*
22. Trunnion Pin

*See Adjustment Paragraph

Main Landing Gear Installation
Figure 1



1. Nose Gear Actuating Cylinder*
2. Drag Brace
3. Fairing
4. Fairing to Strut Rod Assembly (2)
5. Strut
6. Door (2)*
7. Door Actuating Cylinder
8. Up Lock Cylinder*
9. Spring Rod Assembly*
10. Shim
11. Door Rod Assembly (2)
12. Timer Check Valve*
13. Up Lock Rod Assembly*
14. Wheel Bearing

*See Adjustment Paragraph

Nose Landing Gear Installation
Figure 2.

LANDING GEAR — FAULT ISOLATION

FAULT	PROBABLE CAUSE	REMEDY
Nose gear fails to retract.	Defective selector valve or valve control linkage.	Replace selector valve, or correct defective valve control linkage.
	Misadjusted timer-valve actuating linkage.	Correct adjustment of timer-valve actuating linkage.
	Defective or jammed nose gear door control valve.	Replace door control valve.
	Ground safety lock installed.	Remove ground safety lock.
	Linkage to door control valve out of adjustment.	Adjust linkage.
	Uplock latch tripped.	Reset uplock latch manually.
	Manual door valve in ground-servicing position.	Reset valve handle to flight position.
	Emergency system not reset.	Reposition cockpit control handle to flight position.
	Landing gear dump valve not reset.	Reset landing gear dump valve in nose wheel well.
Nose gear fails to extend.	Defective selector valve or valve control linkage.	Replace selector valve, or correct defective valve control linkage.
	Defective down system timer..	Replace down system timer.
	Defective nose gear actuating cylinder.	Replace nose gear actuating cylinder.
	Defective uplock cylinder.	Replace uplock cylinder.
	Uplock clearance incorrect.	Adjust uplock clearance.
	Defective or jammed door control valve.	Replace door control valve.
	Linkage to door control valve out of adjustment.	Adjust linkage.
		If all above have been checked and appear to be correct, incorporation of ASC 195 should preclude improper function of door control valve.
Main gear fails to retract.	Defective selector valve or valve control linkage.	Replace selector valve, or correct defective valve control linkage.
	Linkage to door timer valve out of adjustment.	Adjust linkage.
	Defective door control valve.	Replace door control valve.

FAULT	PROBABLE CAUSE	REMEDY
Main gear fails to retract.	Ground safety locks installed.	Remove ground safety locks.
	Uplock latch tripped.	Reset uplock latch manually.
	Manual door valve in ground-servicing position.	Reset valve handle to flight position.
	Emergency system not reset.	Reposition cockpit control handle to flight position.
	Landing gear dump valve not reset.	Reset landing gear dump valve in nose wheel well.
Main gear fails to extend.	Defective selector valve or valve control linkage.	Replace selector valve, or correct defective valve control linkage.
	Improper uplock clearance.	Adjust uplock clearance.
	Defective uplock cylinder.	Replace uplock cylinder.
	Defective down system timer valve.	Replace timer valve.
	Linkage to door control valve out of adjustment.	Adjust linkage.
	Defective door control valve.	Replace door control valve.
Main and/or nose landing gear doors fail to open or close.	Defective or jammed door control valve.	Replace door control valve.
	Door linkage not properly adjusted.	Adjust door linkage.
	Defective door cylinder shuttle valve.	Replace shuttle valve.
	Defective landing gear selector valve.	Replace selector valve.
	Defective landing gear selector valve control linkage.	Correct control linkage.
	Defective door cylinder.	Replace cylinder.
Landing gear position lights remain on.	Jammed downlock switch plunger.	Clean switch. Replace switch.
	Downlock switch out of adjustment.	Adjust switch.
	Short in circuit or switch.	Clear short or replace switch.
Landing gear position lights remain off.	Jammed downlock switch plunger.	Clean switch. Replace switch.
	Indicator bulbs burned out.	Replace bulbs.
	Defective circuit wiring.	Check continuity. Repair or replace defective wiring.

LANDING GEAR — MAINTENANCE PRACTICES

1. **General**

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES, KEEP AWAY FROM FLAMES.

Keep struts, pistons, cylinders, locks and other retracting or extending units clean. Use cleaning solvent to remove caked grease and dirt. Follow up by lubricating these parts with hydraulic fluid. Replace tires if damaged. Check proper security of axle nuts and brake attaching nuts. Replace damaged wheels. Replace worn bearings.

CAUTION: DO NOT USE HI-LO GREASE TO INSTALL SEALS AND PACKINGS IN THE HYDRAULIC SYSTEMS.

NOTE: Use same type of hydraulic fluid that is used in the aircraft hydraulic system. Seals, O-rings and gaskets should be coated with MCS-352 (Monsanto Company) lubricant or HyJet IV or equivalent before assembly. Coat threaded or rough surfaces over which seals, O-rings or gaskets must pass with same lubricant and use care in assembly to prevent damage or distortion. Use Hi-Lo grease or anti-seize compound sparingly on male thread areas of fittings and fasteners during assembly.

2. **Landing Gear System — Functional Test**

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

CAUTION: HEAD AIRCRAFT INTO WIND IF JACKING IS TO BE DONE OUT OF DOORS. DO NOT JACK IF WIND VELOCITY EXCEEDS 10 MILES PER HOUR UNLESS AIRCRAFT IS SNUBBED AT ALL MOORING POINTS.

DO NOT JACK AIRCRAFT IF MAXIMUM GUSTS EXCEED 20 MILES PER HOUR.

RELEASE BRAKES, JACKING AIRCRAFT WITH BRAKES PARKED COULD RESULT IN AIRCRAFT SLIPPING OFF JACKS.

WALKING ON EMPENNAGE OR OUTER WING WHILE AIRCRAFT IS ON JACKS COULD CAUSE MOVEMENT OF AIRCRAFT ON JACKS. CAUTION MUST BE OBSERVED AT ALL TIMES AND PERSONNEL MUST BE LIMITED TO THOSE ABSOLUTELY NECESSARY FOR PROPER PERFORMANCE OF THIS OPERATION.

LOWER SECTION OF MAIN ENTRANCE DOOR IS NORMALLY SUPPORTED ON GROUND. IF STAIRWAY IS USED FOR ANY REASON WHILE AIRCRAFT IS ON JACKS, STAIRWAY MUST BE SUPPORTED UNDER LOWER STEP.

IF JACKING AIRCRAFT WITH ENGINES REMOVED IS REQUIRED, APPROXIMATELY 1000 POUNDS OF BALLAST SHOULD BE ADDED TO THE FORWARD BAGGAGE COMPARTMENT AND ENTRANCE AREA TO PREVENT A CRITICAL REARWARD CENTER OF GRAVITY SHIFT. THE MAXIMUM BALLAST WEIGHT REQUIRED TO MAINTAIN CORRECT CENTER OF GRAVITY FOR ALL CONDITIONS OF ENGINE AND OTHER EQUIPMENT REMOVAL 1500 POUNDS. NO BALLAST IS REQUIRED FOR PROPELLER REMOVAL ALONE.

WHEN JACKING AIRCRAFT AT THREE POINTS (EACH WING AND NOSE) THE AIRCRAFT GROSS WEIGHT MUST NOT EXCEED 36,000 POUNDS.

A. Jack aircraft as follows:

(1) Remove wing jack pad access plates located at rear outboard underside of each engine nacelle at Wing Station 158.3.

CAUTION: POSITION EACH JACK DIRECTLY BENEATH JACK PADS.

NOTE: When using standard tripod jack, ensure that one of the legs face forward.

(2) Place 12 ton capacity jack in each wing position. Run jacks up to contact wing jack pads on each wing.

(3) Attach fuselage nose jack pad to two pickup holes in fuselage at Fuselage Station 120.10.

(4) Place 5 1/2 ton capacity jack in position and run jack up to contact fuselage jack pad.

(5) Proceed to jack aircraft, ensuring that all jack feet remain on ground. Operate all jacks evenly so aircraft remains as level as possible.

(6) When aircraft is jacked up to required height, set locking devices on jack stands to prevent accidental lowering of aircraft.

B. Landing Gear Normal — Operational Test

NOTE: On all aircraft an inspection of nose gear down and locked bungees and associated fittings is required at each functional test of normal and emergency landing gear system when aircraft is on jacks.

(1) Check operation of bungees as follows. (See Figure 201)

(a) Exert an upward force on downlock latch (1) and restrain latch in this position.

(b) Manually break downlock linkage by rotating downlock compression link assembly aft (out of overcenter position).

(c) Insert a block of wood or suitable material between compression link and latch to hold linkage in unlocked position.

(d) Inspect 1.56 inch dimension on compression link. See Figure 201. (If oversize compression link is found, recommended rework).

(e) Remove block of wood.

(f) Release downlock latch (1), that was rotated out of locked position in Note a. above.

(g) Slowly rotate downlock compression link assembly (3) until it just touches downlock latch (1).

(h) Release downlock compression link assembly. Bungees (4) should exert enough force to cause system to lock.

(i) If compression link does not displace hook and lock into position, raise compression link and place a shim (1/8 to 1/4 inch thick) between the hook and link. Where the link is resting against the shim, remove shim providing free motion of the link to strike hook and lock. Failure to lock may indicate a malfunction of one or both of the bungees (4), binding in compression link or linkage malfunction.

(j) Lubricate sliding surfaces of latch spring clevis and compression link bungees with a light coating of silicone oil SF-81-50 or SF-96-50 (or equivalent) during DOWN and LOCK test of landing gear system when aircraft is on jacks or 600 landings/12 months (1, 2 and 3 respectively).

(2) Check fluid level of hydraulic system reservoir, service if required.

(3) Clear all areas around aircraft.

(4) Connect hydraulic test rig to service connection in right engine nacelle and apply dc power to aircraft.

(5) Open landing gear doors if necessary.

(6) Remove landing gear ground safety locks, nose wheel door hydraulic manual selector valve ground safety lock and nose gear door ground safety strut.

NOTE: Extend main gear door ground safety struts so that timer valves are fully depressed.

(7) Disconnect main gear trunnion and nose gear door rods. Tie doors and door rods in open position.

CAUTION: DO NOT RETRACT GEAR UNLESS SHOCK STRUTS ARE PROPERLY INFLATED AND EXTENDED, GROUND SAFETY LOCKS REMOVED AND A VISUAL CHECK HAS BEEN MADE TO ENSURE ALL UPLOCKS ARE OPEN.

NOTE: Before handle is raised, assign one man in cockpit, one man near each gear and one man at hydraulic test rig.

(8) Raise landing gear. Check gear retraction as follows:

(a) Main and nose gear operating mechanisms must clear entire structure, including hydraulic components, lines, electrical wiring etc.

(b) Check for hydraulic leaks.

(c) Check for NORMAL position on main and nose gear door control valves.

(d) Ensure that both main gear engage their uplocks.

CAUTION: CONDITION OF SHOULDER PIN (P/N 174519), LOCATED INSIDE STRUT, MAY ALLOW BEARING ADAPTER TO BACK OFF. WHICH, IN TURN, ALLOWS THE PISTON/AXLE ASSEMBLY TO EXTEND, CAUSING RETRACTION DIFFICULTIES. REPLACEMENT OF SHOULDER PIN WILL RETURN GEAR TO NORMAL OPERATION.

(e) Ensure that nose gear is engaged on its uplock.

(9) Lower landing gear. All landing gear should extend and lock.

(10) Connect main gear trunnion and nose gear door rods. Remove both main gear door ground safety struts.

(11) Completely cycle landing gear. Tires and gear must clear door structure.

(12) On all Aircraft having ASC 109 perform operational test of landing gear by using auxiliary pump to check for a possible bypass condition (slow gear) on the auxiliary to main selector valve as follows:

(a) Remove external hydraulic pressure from aircraft and bleed off main system pressure.

(b) Move control lever on selector valve located in left side of nose wheel well to AUX TO MAIN ON.

(c) Insert ground safety lock (159GT1007) to hold selector lever in ON.

(d) Pressurize main system with auxiliary pump. (Main and auxiliary systems should read 1500 psi).

(e) Operate landing gear throughout cycle observing steps as outlined in normal functional test.

(f) If landing gear system operates normally shut off auxiliary pump and place valve control lever back to FLIGHT.

(g) Apply external hydraulic power.

(13) Depress following circuit breakers: LG WARN LIGHTS, LG DOWN LIGHTS and LG HORN (Aircraft 1 - 93 and 114 not having ASC 108A.) or LG WARN LTS, LG HORN and NUTCRACKER CONT. (Aircraft 1 - 93 and 114 having ASC 108A and Aircraft 94 - 200, 322 and 323).

(14) With landing gear handle down and gear down and locked, three green lights will be shown in cockpit and no red handle lights.

(15) Advance throttles above 30° quadrant position. Retract flaps to UP.

NOTE: Only on Aircraft having ASC 108A must have flaps up.

(16) Place landing gear handle up. The following should occur:

(a) Landing gear doors should open.

(b) Red lights in handles should come on as clamshell doors open.

(c) Landing gear locks open, green lights should go out.

(d) Landing gear up and locked, clamshell doors close, red lights should go out. (No lights on).

(17) Retard both throttles, warning horn should blow. Push HORN CUTOUT button to silence horn. NO HORN warning light should come on.

NOTE: Steps (18) thru (22) below must be performed only on Aircraft 1 - 93 and 114 having ASC 108A, Part 1 and Aircraft 94 - 200, 322 and 323.

(18) Select APPROACH or FULL with normal flap control handle. Horn should blow as flap handle is selected to either APPROACH or FULL and NO HORN warning light should go out.

(19) Advance throttle, retract flaps to UP position. Horn should silence.

NOTE: Step (21) below must be performed only on Aircraft 1 - 93 and 114 having ASC 108A, Part II.

(20) Turn on auxiliary pump and extend flaps to APPROACH position or lower using emergency flap control handle. Horn should blow when flaps reach APPROACH or lower.

(21) Retract flaps to UP turn off auxiliary pump and reset flap dump valve(s) to NORMAL in hydraulic compartment.

(22) Select and extend flaps to APPROACH position or lower. Horn should blow when flaps reach APPROACH position, and NO HORN warning light should go out. Horn cannot be silenced by button.

(23) Place landing gear handle down.

(a) Red light in handles should come on.

(b) As landing gear extends to DOWN and LOCKED position, green lights should come on.

(c) Red lights in handles should go out.

(d) Warning horn should silence as all three gears are down and locked. On Aircraft 1 - 93 and 114 not having ASC 108A, NO HORN light should go out when one throttle is advanced, after three gear are down and locked.

C. Speedbrake Indication

NOTE: Steps (1) and (2) below are for Aircraft, 1 - 93 and 114 not having ASC 108A, Part I and Aircraft 94 - 200, 322 and 323.

(1) Raise landing gear to UP position. Pull speedbrake handle (flaps must be up on Aircraft having ASC 108A, Part II. The following should occur:

(a) Both main landing gear doors open.

(b) Red lights in landing gear handles come on as gear doors open.

(c) Both main landing gear extend and lock down.

(d) Left and right green indicating lights should come on.

(e) Clamshell doors close and red lights in landing gear handles should go out.

(f) Nose gear indicating light should not come on.

NOTE: Warning horn operation same as normal landing gear operation.

(2) Place landing gear handle in DOWN. Nose gear should extend and lock. Nose gear green light should come on. Speedbrake handle should move in.

NOTE: Steps (3) and (4) below are for Aircraft 1 - 93 and 114 having ASC 108A, Part I.

(3) Raise landing gear to up. Pull speedbrake handle. The following should occur:

(a) Both main landing gear doors open.

(b) Red lights in landing gear handles come on as gear doors open.

(c) Both main gears should extend to DOWN and LOCKED, and the left and right amber speedbrake lights should come on.

(d) Main gear doors should close and red lights in landing gear handles should go out.

NOTE: Left and right green indicating lights should not come on.

(4) Move landing gear handle to down. The following should occur:

(a) Red lights in handles should come on.

(b) Speedbrake handle should automatically retract and amber speedbrake lights should go out.

(c) Left and right main gear green lights should come on.

(d) Nose gear should extend to DOWN and LOCKED.

(e) Nose gear green light should come on and red lights in handle should go out.

(5) Ground safety solenoid check (aircraft must be on jacks).

(a) Pull LG HORN circuit breaker (or NUTCRACKER circuit breaker on Aircraft 1 - 93 and 114 having ASC 108A, Part I).

(b) Remove hydraulic power from aircraft.

(c) Attempt to raise landing gear handle. Handle should remain locked in DOWN.

(d) Depress LG HORN circuit breaker (or NUTCRACKER circuit breaker) pulled in Step (a) above.

D. Emergency Operational Test

(1) Ensure that emergency nitrogen bottle has a charge of 1900 - 2000 psig. Charge if required.

(2) With all landing gear safety devices removed and aircraft on jacks, raise landing gear handle. All landing gear should retract and landing gear doors should close.

(3) Shut down hydraulic test rig and dissipate remaining hydraulic pressure in system.

(4) Place landing gear handle in DOWN. Landing gear should remain in uplock. Red light in gear handle should come on.

(5) Pull emergency landing gear handle. Check for the following:

(a) All gear doors should open.

(b) All landing gear should unlock and nose gear extends and locks in DOWN.

(c) Main gear should free fall but should not lock. Place a man on each gear and force gear into DOWN and LOCKED.

NOTE: In flight, slipstream would force main gear into downlock where they would be locked by their own mechanical action.

(d) All landing gear doors should stay open.

(e) Red light in gear handle should go out, three green lights should stay on.

WARNING: WHEN EMERGENCY LANDING GEAR HANDLE IS RETURNED TO STOWED POSITION EMERGENCY AIR MIXED WITH HYDRAULIC FLUID WILL BE RELEASED VIOLENTLY TO ATMOSPHERE FROM VENT LOCATED ON EXTERNAL SKIN AT FORWARD RIGHT SIDE OF NOSE WHEEL WELL. ALL PERSONNEL MUST KEEP CLEAR OF THIS VENT.

(6) Return emergency landing gear handle to stowed position.

(7) When all air pressure is bled off, reset dump valve in nose wheel well, (Aircraft not having ASC 116).

NOTE: Use dump valve reset D-ring in cockpit if incorporated, (Aircraft having ASC 116).

(8) Remove nose gear door safety strut.

(9) Apply hydraulic power gradually. All landing gear doors should close.

(10) Place landing gear handle to UP, then as gear starts to retract, place landing gear handle to DOWN. This is to ensure proper repositioning of shuttle valves.

(11) Operate landing gear through six complete cycles to bleed nitrogen from system and ensure proper operation.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

DO NOT ATTEMPT TO REMOVE EMERGENCY BOTTLE BLEEDER VALVE WITH ANY PRESSURE REMAINING IN BOTTLE.

(12) Bleed nitrogen from bottle by loosening bleeder valve approximately two turns.

(13) After all nitrogen is bled from bottle, remove bleeder valve and allow all moisture to drain.

(14) Install bleeder valve and torque to 80 inch-pounds.

(15) Recharge emergency bottle to 2000 psig. Check bottle for leakage.

(16) Check fluid level in hydraulic reservoir and service.

E. Lower aircraft as follows:

(1) Clear all equipment and personnel from areas beneath aircraft. Install landing gear ground locking pins.

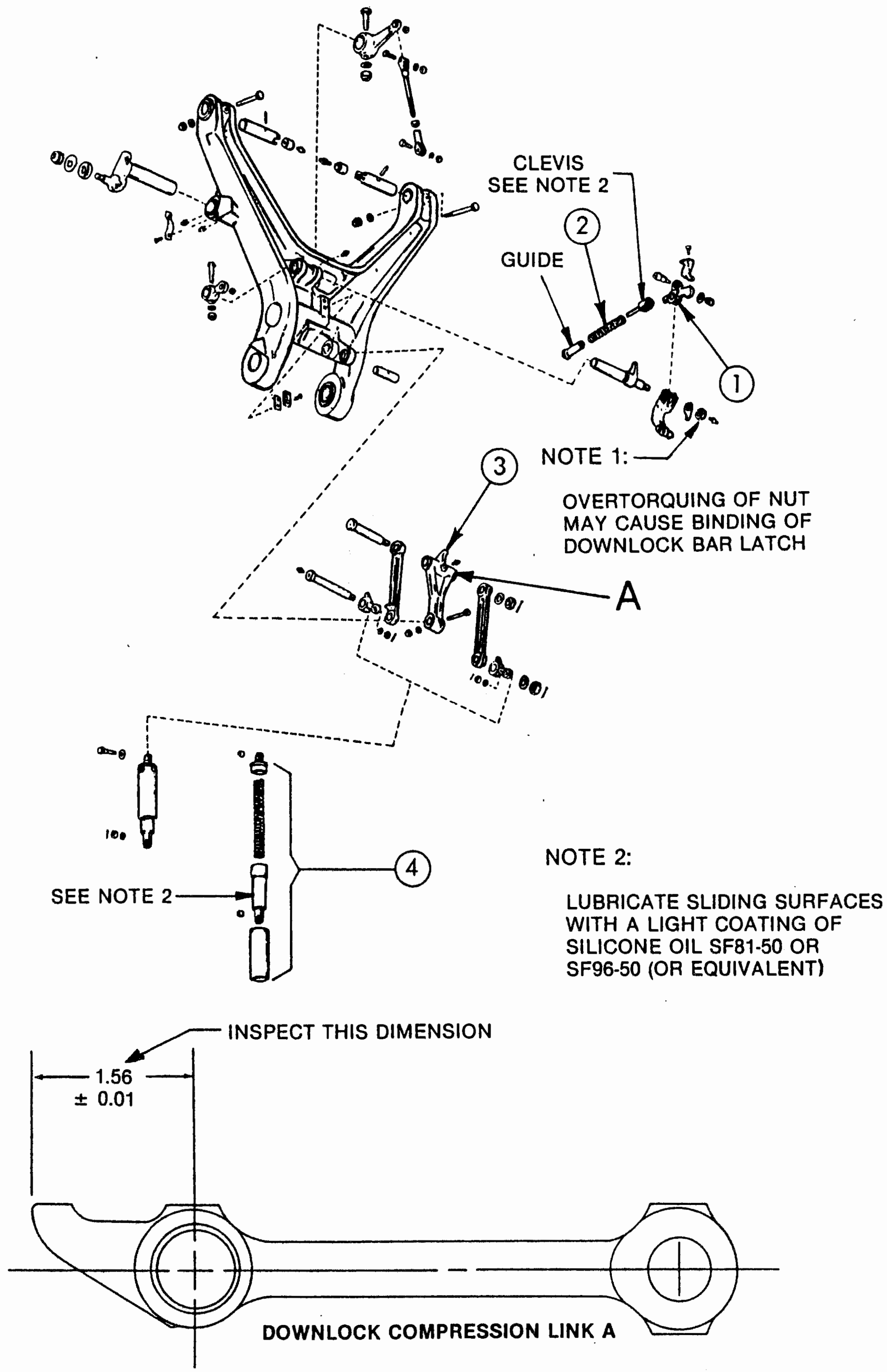
(2) With one man at each jack, raise jacks only enough to permit locking devices to be unlocked.

(3) Unscrew locking devices until jack shafts are free to be fully lowered.

(4) Lower jacks evenly and slowly until aircraft is firmly on ground.

(5) Remove jacks and nose jack pad.

(6) Secure wing jack pad access doors.



Nose Gear Downlock Assembly
Figure 201

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MAIN LANDING GEAR — DESCRIPTION / OPERATION

1. General

(See 32-0 Figure 1)

Each main landing gear comprises two wheels and tires and two brake assemblies, mounted on a conventional Bendix nitrogen-hydraulic shock strut housed in the nacelle and attached to wing structure. Further support is obtained from the drag brace which attaches to the lower end of the shock strut. The gears retract forward and up into the wheel wells and extend aft and down. Before retraction starts, the main wheel doors open; after completion of retraction, they close.

CAUTION: AS A SAFETY PRECAUTION, INSTALL TWO MAIN LANDING GEAR GROUND SAFETY LOCKS, 98GT1030, AND NOSE LANDING GEAR GROUND SAFETY LOCK, 159GT1015, WHENEVER THE AIRCRAFT IS ON THE GROUND.

REMOVE ALL GROUND SAFETY LOCKS BEFORE FLIGHT.

2. Main Gear Shock Struts

(See 32-0 Figure 1)

Each shock strut is a conventional nitrogen-hydraulic unit designed to absorb the shock of landing, takeoff, and taxiing. It is essentially a steel piston riding in a cylinder and utilizes O-rings to prevent leakage between the piston and outer cylinder. The wheel axles are welded to the strut and the brakes are bolted to flanges on the axles. An air-oil filler valve is provided at the top of the strut and a placard on the main gear door indicates the correct strut pressures for given strut extensions. The strut is filled with hydraulic fluid and pressurized with dry nitrogen. Jack points are provided at the base of the main struts to permit removal of tires, wheels and brakes. Rings are provided for tie-down.

3. Main Gear Drag Brace

When the main gear is in the down position, each shock strut is restrained from pivoting about its trunnion by the drag brace which is attached to the lower end of the shock strut outer cylinder on one end and to a fitting on the wing at the other end. When gear retraction is selected, the drag brace is unlocked by the hydraulic action at the unlock port of the downlock and the drag brace extends in length as the retraction continues (See Figure 201). With the gear extended, the drag brace retracts. The landing gear is held down and locked by hydraulic pressure at the lock port of the downlock cylinder. Drag brace assemblies 159L10003-3 and 159L10003-5, have been replaced by the 159L10003-7 with ASC 144. The 159L10003-7 drag brace assemblies are updated to a 159L10003-11 configuration after having CB 262.

4. Main Gear Downlock Cylinders

(See 32-0 Figure 1)

Each main landing gear drag brace incorporates a downlock cylinder. The plunger end of the cylinder is inside the drag brace, and the piston end is on the outside. The piston is rigidly attached to the end fitting of the drag brace, this allows the body of the cylinder to move in relation to the piston. A spring, built into the cylinder, works in conjunction with hydraulic pressure to cause a mechanical locking action when the main gear is in the down position. Hydraulic pressure is required to operate the cylinder to release the downlock mechanism so that the gears may be retracted. The thread-size of the unlock and lock ports on the body of the cylinder is 7/16 inch, the tubing size is 1/4 inch.

NOTE: There is no downlock cylinder in the nose installation.

5. Main Gear and Speedbrake Selector Valve

This valve is a manually controlled, two position, four way hydraulically operated rotary valve. It is located at the top of the nose wheel well and is controlled by the landing gear control handles, and the speedbrake control handle.

See Main and Nose Gear Hydraulic System Pressure section.

Main hydraulic up pressure enters P1 port and is directed through the C1 port to four points; through the check valve and pump valve to the unlock port of the drag brace downlock cylinder, through the up system timer valve, through the head end of the uplock cylinder and to the door control valve.

Main hydraulic down pressure enters the P1 port and is directed through the C2 port through the check valve and dump valve to the lock port of the drag brace downlock cylinder, through the down system timer valve, through the door control valve and to the head end of the main gear cylinder.

For operation of the speedbrakes portion of the selector valve, see Speedbrakes System — Description / Operation section. (32-6-0)

MAIN LANDING GEAR — MAINTENANCE PRACTICES

1. Main Gear Shock Strut — Service

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

NOTE: Comply with directions on data plate on shock strut.

If different type hydraulic fluid is used, revise data plate as to type installed.

A. Jack aircraft until tires clear ground.

WARNING: DO NOT LOOSEN VALVE BODY WHILE UNIT IS DISCHARGING NITROGEN. SERIOUS INJURY OR LOSS OF LIFE MAY RESULT IF VALVE ASSEMBLY IS EJECTED UNDER PRESSURE.

B. With wrench, check to ensure that top 3/4 inch hexagon swivel nut is tight, then remove valve cap. Loosen swivel nut slightly until nitrogen is discharging. Wait until all nitrogen is discharged.

C. When all nitrogen is discharged, remove filler valve.

CAUTION: DO NOT USE MIL-H-5606 FLUID IN SHOCK STRUT.

D. Fully compress main shock strut and fill to overflow with hydraulic fluid (as required in particular shock strut) slowly to avoid foaming.

E. Slowly extend and compress shock strut several times to ensure that all air is expelled when shock strut is fully compressed. Add fluid to fill void if necessary.

F. On those Aircraft having ASC 169 and 175, extend shock strut and add an additional 3 1/2 ounces of hydraulic fluid into shock strut.

NOTE: Do not use air filler valve, part number MS28889-1 on Gulfstream I aircraft shock struts. The seals used in this valve are not compatible with phosphate Ester type IV fluid.

G. Allow shock strut to fully extend and install filler valve. Torque to 100-110 inch-pounds. Lockwire filler valve to shock strut.

WARNING: USE DRY NITROGEN ONLY. NEVER USE OXYGEN OR AIR.

H. Loosen 3/4 inch hex swivel nut 3 1/2 turns and attach nitrogen supply line.

I. Inflate shock struts to pressures required in the following configurations.

(1) Aircraft having ASC 169, inflate to 275 psi.
(2) Aircraft having ASC 175, inflate to 280 psi.
(3) All other aircraft, inflate to 381 psi.

J. After inflating torque swivel nut to 50-75 inch-pounds.

K. Shut off nitrogen supply and remove hose. Install valve cap.

L. Lower jacks.

2. Main Gear Drag Brace — Removal / Installation

(See Figure 201)

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Connect a source of external hydraulic pressure to aircraft.

(3) Place cockpit landing gear control handle in up position and slowly increase hydraulic pressure until drag brace unlocks. Shut off power before gear retracts.

(4) Block gear in position to prevent gear from going through down and locked position.

(5) Remove drag brace fairing attachment hardware. Retain attaching hardware.

(6) Relieve hydraulic system pressure and disconnect hydraulic lines at top end of drag brace. Cap all open lines and fittings.

CAUTION: DO NOT LET LANDING GEAR SWING TO VERTICAL POSITION UNRESTRICTED SINCE DAMAGE TO DOOR VALVE LINKAGE WILL RESULT.

(7) Detach lower end of drag strut from main gear. Retain attaching hardware.

(8) Remove and retain upper attachment pin to facilitate removal of drag brace.

(9) Remove drag brace.

B. Installation

(1) Place drag brace in position and install upper attachment pin.

(2) Attach lower end of drag brace to main gear with hardware retained during removal.

(3) Connect hydraulic lines to fittings.

(4) Connect external source of hydraulic pressure to main hydraulic system. Place gear handle to DOWN.

(5) Remove ground safety braces from main and nose gear doors. Remove blocks and ground safety locks from gears and clear area for operational test.

(6) Cycle gear to check for proper operation of main gear. Adjust drag brace fairing to close flush in gear up position with landing gear actuator under full hydraulic system pressure.

NOTE: Trunnion door rods must be disconnected to make this adjustment possible.

(7) Perform a Landing Gear System — Operational Test. (See Section 32-0)

(8) Lower jacks.

3. Main Gear Drag Brace — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

See Figure 202 thru Figure 207.

A. Remove drag brace. (See Main Gear Drag Brace — Removal / Installation, see this section)

B. To disassemble drag brace:

(1) Remove and retain strap assembly 159H10154-1 (flex lines attachment).

(2) Remove and retain clamp assembly 159L10063-1 (drag brace fairing attachment).

(3) Remove and retain downlock limit switch (3CE3).

(4) Break lockwire at outer cylinder nut 159LM10004-1 and unscrew nut. Slide outer cylinder 159LM10002-1 down inner cylinder 159LM10001-1 (159LM10001-3 on brace assembly 159L10003-11). Slide off large scraper RW3-3-1/2-1/4-3/8 and bearing 159LM10003-3 (159LM10003-5 on brace assembly 159L10003-11).

(5) Break lockwire and unscrew rod end fitting nut 159LM10006-1. Slide out rod bearing 159LM10011-5 (159LM10011-7 on brace assembly 159L10003-11) and small scraper MS28776M2-18.

(6) Unscrew rod assembly end fitting 159LM10005-1 (159LM10005-5 on brace assembly 159L10003-11) and slide rod assembly 159LM10008-1 and downlock 159LM10007-1 out of outer cylinder.

NOTE: If difficulty is experienced in sliding out bearing from end fitting, it may be tapped out using an appropriate hardwood drift pin.

(7) Remove downlock fitting from rod assembly be removing bearing retainer 159LM10014-1, bearing 159LM10012-1 and lockpin 159LM10016-1. Downlock may then be unscrewed from rod.

(8) Remove two large bearings 159LM10010-3 (159LM1001 0-5 on brace assembly 159L10003-11) from inner cylinder.

CAUTION: ENSURE NOT TO DAMAGE SPECIAL LOCKSCREWS THAT INDEX INNER CYLINDER AND END FITTING TO DOWNLOCK SLEEVE.

NOTE: Steps 9. and 10. below the downlock sleeve and end fitting are mating parts. The rod assembly and downlock (finger lock) are mating parts and should be kept together for reassembling.

(9) Remove inner cylinder 159LM10001-1 (159LM10001-3 on brace assembly 159L10003-11) by removing special lockscrew 159LM10013-1 at end of inner cylinder and unscrewing inner cylinder from downlock sleeve 159LM10009-1.

CAUTION: DO NOT BREAK LOCKWIRE OR REMOVE THREADED STUDS AND NUTS FROM END FITTING. REMOVAL OF STUDS ARE NOT REQUIRED AND HAVE BEEN INSTALLED AND ADJUSTED WITH RESPECT TO DOWNLOCK CYLINDER.

(10) Remove large special washer SP4300-189. The downlock cylinder 159H10025-3 will remain attached to end fittings and need not be disassembled for this inspection.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(11) Clean all parts with solvent. Remove all grease sludge and dirt. Moisture drain holes in rod end fitting must be cleaned thoroughly.

(12) Clean outside of downlock cylinder with clean, lint-free, solvent dampened cloth. Do not wash in a cleaning solvent.

(13) Inspect downlock (finger lock) fitting 159LM10007-1 as follows:.

(a) Inspect for deep nicks, brinelling and corrosion.

(b) Visually inspect for 2.740 / 2.735 inch diameter shoulder at base of fingers.

(14) Inspect downlock sleeve 159LM10009-1 for deep nicks and brinelling from downlock. Inside of sleeve should be smooth and free of burrs.

(15) Inspect blind rivet at end of piston rod assembly 159LM10008-1 for looseness. Replace if required.

(16) Inspect small MS28776M2-18 and large RW3-3 1/2-1/4-3/8 scraper rings. Replace as required.

NOTE: Scraper rings may be darkened in color, but should be free from any binding and visible cuts.

(17) Inspect condition of all parts for corrosion and any obvious defects, especially check the following areas for corrosion:.

(a) Rod end fitting 159LM10005-1 (159LM10005-5 on brace assembly 159L10003-11) inner diameter.

(b) Outer cylinder 159LM10002-1 inner diameter.

NOTE: The following reassembly and installation procedures apply to drag braces 159L10003-1, -3, -5 and -7 and those drag braces that have been reworked into a 159L10003-11, which shall be used for all replacement and spares.

When reassembling drag brace, ensure original downlock sleeve 159LM10009-1, end fitting 159LM10000-5 and downlock 159LM10007-1 are used since these parts are not interchangeable between drag brace assemblies.

(c) Screw sleeve 159LM10009-1 into place and install washer SP4300-189.

(d) Install lockscrew 159LM10013-1 locking two parts together.

NOTE: Ensure that downlock sleeve extends 0.850 to 0.860 inch beyond end of inner sleeve. See Figure 206.

Lockscrews are correctly installed when bottomed in holes.

(e) If new downlock (finger lock) 159LM10007-1 is required after inspection, assemble to piston rod as follows: (See Figure 207)

(1) Thread new downlock fitting on piston rod. Downlock fitting is to be tight against shoulder on piston rod.

(2) Fabricate a drill bushing having 0.249 to 0.251 inch outer diameter and 0.189 to 0.190 inch inner diameter.

(3) Insert drill bushing in 0.249 to 0.251 inch hole in downlock fitting that is farthest from existing 0.189 to 0.190 inch hole in piston rod. Drill new 0.189 to 0.190 inch diameter hole in piston rod.

(4) Remove downlock fitting and deburr parts.

(5) Assemble downlock fitting on piston rod and install retaining pin. Lubricate parts as required.

(6) Install bronze bearing and bearing retainer. Lubricate and secure with two screws and lockwire screws.

(f) If replacement of three downlock studs 159RDL160-3 is necessary, an adjustment is required as follows: (See Figure 202)

NOTE: The hydraulic downlock actuator must be installed in the upper drag brace end fitting.

(1) If this adjustment is made after assembling drag brace, it must be unlocked and extended and hydraulic pressure reduced to zero.

(2) Install screws 159RDL160-3 and nuts NAS509-4 and adjust by turning stud until 45° chamfer touches hydraulic downlock cylinder 159H10025-1 or -3.

NOTE: Downlock cylinder 159H10025-1 is used on brace assembly 159H10003-1, -3 and -5 and 159H10025-3 is used on brace assembly 159H10003-7 and -11.

(3) Back off studs one-half turn. Maintain this position and torque locknuts (NAS509-3) to 15-30 inch-pounds and safety-wire nuts against both directions.

(g) Lubricate the following surfaces with grease MIL-G-23827 or equivalent, coat parts with light application of grease.

(1) Inside and outside diameter of downlock fitting 159L10007-1 (finger lock).

(2) Inside and outside diameter of downlock sleeve 159L10009-1.

(3) Outside of downlock hydraulic cylinder 159H10025-1 or -3.

(h) Install special washer SP4300-189 over downlock sleeve 159LM10009-1 and screw downlock sleeve with inner cylinder 159LM10001-1 or -5 until lockscrew holes are aligned. Install and tighten lockscrew 159LM10013-1 with washer AN960-416L. This indexes end fitting to sleeve. The inner cylinder, special washer and end fitting should be tight. Lockwire two lockscrews together.

(i) Slide rod end nut 159LM10006-1, small scraper ring MS28776M2-18, small bearing 159LM1001 1-5 or -7, retainer 159LM10014-1 and bearing 159LM10012-1 over piston rod.

NOTE: Ensure rod end nut, scraper and rod end fitting are installed in correct position.

(j) Screw downlock fitting 159LM10007-1 to piston rod. Fitting must be tight against piston rod and lockpin holes must line up. Install lockpin 159LM10016-1 in lockpin hole, slide bearing 159LM10012-1 over lockpin. Slide bearing retainer 159LM10014-1 in place and attach with two screws AN502-10-8 and safety-wire screws together.

(k) Place new bearing 159LM10010-5, bearing 159LM10003-5, large scraper RW3- 3 1/2-1/4-3/8 and nut 159LM10004-1 in place on inner cylinder. Slide inner cylinder inside of outer cylinder, slide rod assembly and attaching parts into inner cylinder.

(l) Tighten nut (159LM10004-1) into outer cylinder and torque 75-100 inch-pounds and lockwire.

(m) Screw piston rod end fitting 159L10005-1 (159LM10005-5 on brace 159L10003-11) into outer cylinder.

(n) With piston rod bearing 159LM10011-5 (159LM10011-7 on brace 159L10003-11) and small scraper MS28776M2-18 in place, tighten rod end nut 159LM10006-1 just enough to remove all end play. Lockwire nut, fitting and outer cylinder together.

(o) Install drag brace fairing and flexible line attachments.

(p) Install drag brace in aircraft using retained hardware. (See Main Gear Drag Brace — Removal / Installation, see this section)

(q) Install downlock limit switch (3CE3). See Figure 202 for adjustment and sealing of switch.

(r) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(s) Ensure drag brace fairings close flush in gear up position with gear actuator pressurized.

(t) Adjust fairing rods if required.

NOTE: The trunnion door rods must be disconnected in order to adjust fairing. Do not disturb adjustment of trunnion door rods.

4. Main Gear Drag Brace, Rod End Bearing — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Jack aircraft.

B. Disconnect lower end of drag brace from main gear. Retain attaching hardware.

C. Inspect rod assembly, end bearing for security of attachment, galling, wearing, absence of lubrication or other visual defects.

NOTE: If bearing fails the inspection replace complete drag brace assembly.

D. Attach lower end of drag brace to main gear, using hardware retained in Step B. above.

E. Cycle gear through an abbreviate normal operational test. See Landing Gear Normal / Emergency Operational Test Section of Landing Gear System — Functional Test. (See Section 32-0)

F. Lower jacks.

5. Main Gear Uplock Roller Bolt — Removal / Installation

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Removal

(1) Jack aircraft.

(2) Remove main and nose wheels.

(3) On main gear remove brake to provide clearance for removal of roller bolt.

(4) Remove uplock roller and bolt assembly.

B. Installation

CAUTION: ENSURE ROLLER CAN BE TURNED FREELY AND THAT CLAMP-UP BUSHING AND BOLT DO NOT TURN.

(1) Torque nut to 95-100 inch-pounds.

(2) Install cotter pin in main and nose gear uplock roll bolts and nuts.

(3) Lubricate as required.

(4) Install brake on main gear.

(5) Install main and nose wheels.

(6) Lower aircraft.

6. Main Gear Uplock Roller Bolt — Inspection

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

A. Remove uplock roller bolt. (See this section)

B. Clean parts with cleaning solvent, ensure all grease passages are clear.

C. Inspect all parts for galling, minor imperfections can be polished out with fine emery cloth.

D. Inspect clamp-up bushings and roller bolt assembly for cracks. Replace if necessary.

E. Install parts on strut. Ensure lubricant fitting on stud is in accessible position for periodic lubrication.

F. Install uplock roller bolt. (See this section)

7. Main Gear Drag Brace — Draining

Once every week, remove either drain screw from lower end fitting and rotate outer cylinder only to position drain hole to lowest point. Allow all moisture to drain from cylinder. Install and lockwire drain screw.

8. Main Gear Strut Actuator Adapter — Removal / Installation

A. Removal

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Jack aircraft.

(2) Apply penetrating oil to ease removal of adapter; if available, a heat lamp may be applied to area.

(3) Use puller to remove adapter.

NOTE: If puller is not available an alternate method of removal may be used.

(4) Attach hydraulic pump to gear down side of retract cylinder.

(5) Back off large nut on adapter fitting, 1/2 turn.

(6) Remove adapter by slowly applying hydraulic pressure and backing off nut 1/2 turn at a time until actuator is bottomed.

(7) Remove adapter from actuator.

NOTE: This is a Type Certificate Data Sheet No. 1A17 item and is life limited to 2500 landings and scrap. This complies with Section IV of Chapter 5.

B. Installation:.

(1) Apply a coating of cosmolene on shaft of adapter and on two tapered bushings (inside and out-side diameter).

(2) Install adapter (172252) with attaching hardware and torque nut to 1500-1800 inch-pounds.

(3) Connect main gear actuator piston to mounting adapter and apply torque as follows:.

(a) Bolt AN5-15 100 - 140 inch-pounds
(b) Nut MS20364-720A 270 - 300 inch-pounds

(4) Install adapter (173625) with attaching hardware and torque nut to 1500-1800 inch-pounds.

(5) Connect main gear actuator piston to mounting adapter and apply torque as follows:.

(a) Bolt AN5-21A 100 - 140 inch-pounds
(b) Nut MS20364-918A 480 - 600 inch-pounds

(6) Lube all fittings with grease.

(7) Remove landing gear safety devices as required.

(8) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(9) Install landing gear safety devices and lower jacks.

9. Main Gear Strut Actuator Adapter — NDT Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove adapter.

B. Thoroughly clean all parts removed and adapter mounting boss with solvent.

C. Inspect all parts for wear and visual signs of cracks.

D. Magnaflux (NDT) mounting adapter, if cracks are evident, replace and record condition found.

E. Install adapter.

10. Main Gear Uplock Beam Support Angles — NDT Inspection

NOTE: This inspection applies to aircraft equipped with main landing gear uplock beam forward bracket, 159W10150-51 and -52. (Aircraft 1 - 83 and 114 not having ASC 179).

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Determine part number of the main landing gear uplock beam bracket. See to Figure 206 which indicates difference in outline between 159W10150-51, -52 angles and 159W10150-71, -72 angles.

NOTE: If 159W10150-71, -72 angles are installed, disregard remainder of this inspection.

B. Inspect main landing gear uplock beam forward brackets, 159W10150-51 and -52 for cracks using dye penetrant methods (NDT) in conjunction with a glass of at least a ten power or an approved equivalent inspection method, inspect attachment of each bracket to firewall bulkhead for loose rivets caused by elongated rivet holes.

C. If cracks or loose rivets are found, replace brackets. Refer to ASC 179.

11. Main Gear Shock Strut Seal — Removal / Installation

A. Removal (See Figure 209)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Remove ground safety lock only from shock strut requiring seal replacement.

(3) Retract gear to approximately 45° of full retraction using external hydraulic power, or auxiliary pump (if ASC 109 is installed on AUX to MAIN selector).

(4) Shut down external hydraulic power if used.

WARNING: DO NOT LOOSEN VALVE BODY FROM STRUT UNTIL ALL PRESSURE HAS BEEN RELEASED. SERIOUS INJURY OR LOSS OF LIFE CAN RESULT IF VALVE ASSEMBLY IS EJECTED UNDER PRESSURE.

(5) On shock strut check 3/4 inch hex swivel nut for tightness.

(6) Remove valve cap.

(7) Loosen swivel nut slowly to bleed nitrogen from shock strut (maximum 3 1/2 turns). Allow all pressure to be released, then remove filler valve.

(8) Remove torque arm pin at apex joint.

(9) Cut safety wire and loosen packing nut (16), provide drain pan under shock strut assembly.

(10) Slowly withdraw piston assembly (containing wheels) from cylinder, drain all fluid from piston.

(11) Remove upper bearing sleeves (1) and locking pin (3), unscrew and remove bearing adapter (2) then remove all parts (See Figure 209, Items (4) through (13)).

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

NOTE: It is possible that shock strut may not include rubber backup ring (12), however upon assembly, replacement seals should include this ring.

When replacing seals, use seals contained in GSK modification kit.

(12) Wash all metal parts and surfaces with dry cleaning solvent and dry with clean filtered air.

(13) Before assembly, apply a generous coating of grease to all rubber rings and to inside and outside diameter of packing adapter (9) and lower bearings (14) to facilitate assembly.

CAUTION: POSITION RUBBER BACKUP RING (12) ON PISTON WITH CONCAVE SIDE FACING UPWARD AS SHOWN IN FIGURE 209.

(14) Assemble parts (13 through 4) on piston (17), then install bearing adapter (2), align hole in adapter with hole in piston, then install locking pin (3) and upper bearing (1).

B. Installation

NOTE: Adjust position of partially retracted shock strut cylinder to maintain alignment while installing piston (17) (wheel may be used to support piston during installation).

(1) Install piston into outer cylinder, use extreme care during installation.

(2) Tighten packing nut (16) and safety-wire nut to lug on cylinder.

(3) Install torque arm.

(4) Extend gear into down and locked position and install ground safety lock.

(5) Service shock strut.

(6) Lower jacks.

12. Main Gear Down Pressure Relief Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Connect a source of external hydraulic pressure to quick disconnect fittings at right engine hydraulic pump.

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(2) Jack aircraft.

CAUTION: SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO ALLOW RESIDUAL PRESSURE TO BLEED FROM RESERVOIR.

(3) Place a suitable container with a minimum capacity of five gallons under valve in main wheel well.

(4) Disconnect lines and remove fittings from valve. Remove valve from aircraft

B. Installation

(1) Replace fitting in valve using new O-rings.

(2) Place valve in position and install hydraulic lines.

(3) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(4) Check reservoir for capacity and service as required.

13. Main Gear And Speedbrake Selector Valve — Removal / Installation

A. Removal

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Jack aircraft.

(2) Connect a source of external hydraulic pressure to ground test connection in right engine nacelle.

(3) Locate selector valve in nose wheel well upper bulkhead and place container below valve.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS, SLOWLY REMOVE HYDRAULIC RESERVOIR FILLER CAP TO RELIEVE ANY RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

(4) Disconnect and remove hydraulic lines necessary to gain access to valve. Retain lines and cap open ports.

(5) Disconnect lines and fittings from valve and plug openings.

(6) Remove and note position of valve shaft to control linkage.

(7) Remove screws. Remove valve with its attached support bracket from underside of cockpit floor. Retain screws and Stat-O-Seals.

NOTE: The seals are located above support bracket at screws.

Note valve position and ensure that valve to be installed is in the same position.

(8) Remove valve from support bracket.

B. Installation

NOTE: When installing main gear selector valve, 30° either side of neutral (dot) hole, to left is landing gear down, to right is landing gear up.

(1) Install valve to support bracket.

(2) Place valve in position on aircraft and install screws and Stat-O-Seals.

(3) Reconnect valve shaft to control linkage in same position as removed.

(4) Remove plugs and reconnect all lines and fittings to valve.

(5) Replace any lines that were removed to gain access to valve.

(6) Check fluid level in hydraulic reservoir and service.

(7) Remove landing gear safety devices and perform a Landing Gear System — Functional Test. (See Section 32-0)

(8) Replace landing gear safety devices as required.

(9) Remove external hydraulic pressure source, lower jacks and remove from aircraft.

14. Main Gear — Adjustment

(See 32-0, Figure 1)

A. Inboard side trunnion pin must be free, thereby permitting rotation of the trunnion swivel joint throughout its travel.

B. After installation of drag brace, adjust limit switch so that switch plunger operates only when in contact with both down lock actuating cylinder and lock in fully locked position. When drag brace is fully retracted and down lock is locked, the circuit between pins A and C on plug should be complete. With down lock cylinder in other than locked position, circuit between pins D and B should be complete.

C. Ensure that 0.850-0.860 inch dimension for sleeve was met before screws were installed. This dimension is obtained during manufacture of drag brace.

D. With main gear strut in down and locked position (drag brace fully retracted and locked), adjust length of retracting cylinder before attaching it to trunnion. In fully extended position, it should be a minimum of 1/8 inch longer than actual distance between cylinder head attaching point on wing and trunnion pin on shock strut.

E. In addition to meeting the requirements of Step D, the terminal on the retract cylinder should be adjusted if necessary ,to provide a 7/32 - 9/32 inch clearance from fully retracted piston position with landing gear up and locked and no hydraulic pressure to the retract side of cylinders.

15. Main Gear Torque Arm — Removal / Installation

A. Removal

(1) Remove brackets supporting flexible hydraulic lines.

(2) Remove bolts connecting torque arms to strut.

(3) Remove torque arm.

B. Installation

(1) Install torque arm.

(2) Install torque arm connecting bolts, nut and cotter pin.

(3) Install flexible hydraulic line brackets.

16. Main Gear Torque Arms — Inspection (NDT)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove torque arm. (See this section)

WARNING: USE SOLVENT IN WELL VENTILATED AREA, AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

B. Clean and remove paint on from entire forging with stripper See Figure 210.

C. Perform dye check or Zyglo inspection (NDT) on torque arms (P/N 171417 and 171418 or 86336) in areas as shown in Figure 210.

NOTE: If cracks are found replace assembly.

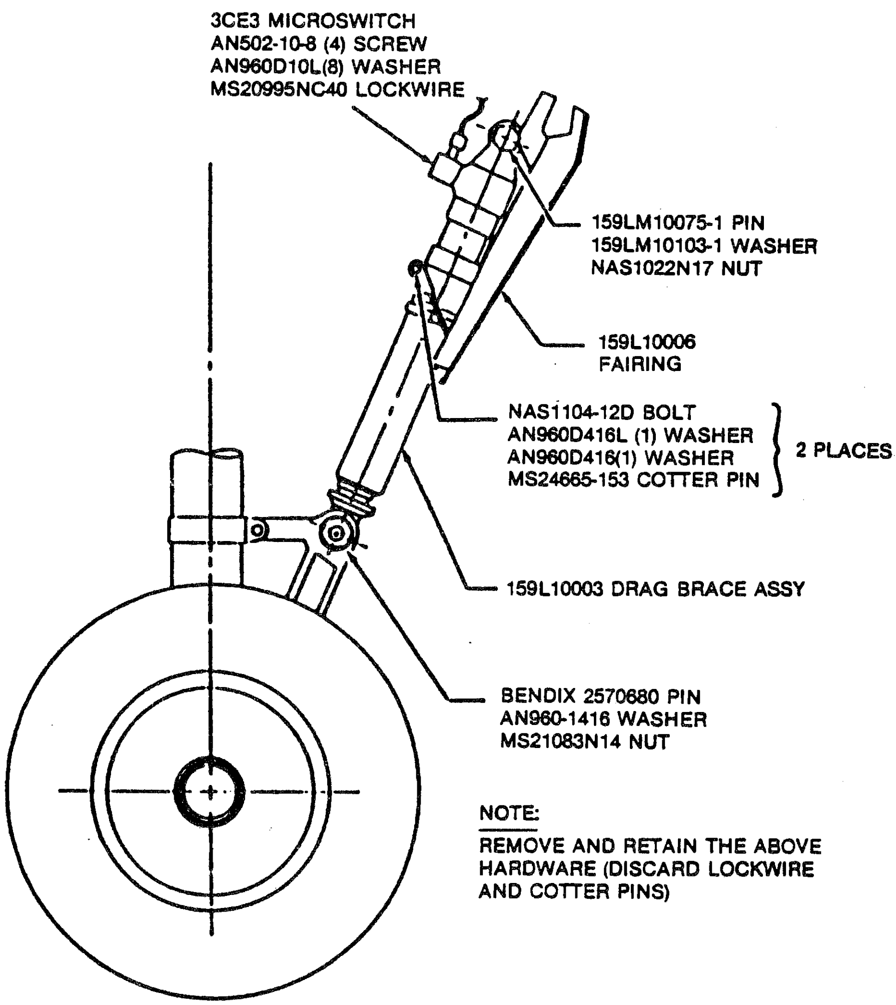
D. Upon completion of inspection (NDT) apply approved alodine, primer and paint to assembly.

E. Install torque arms. (See this section)

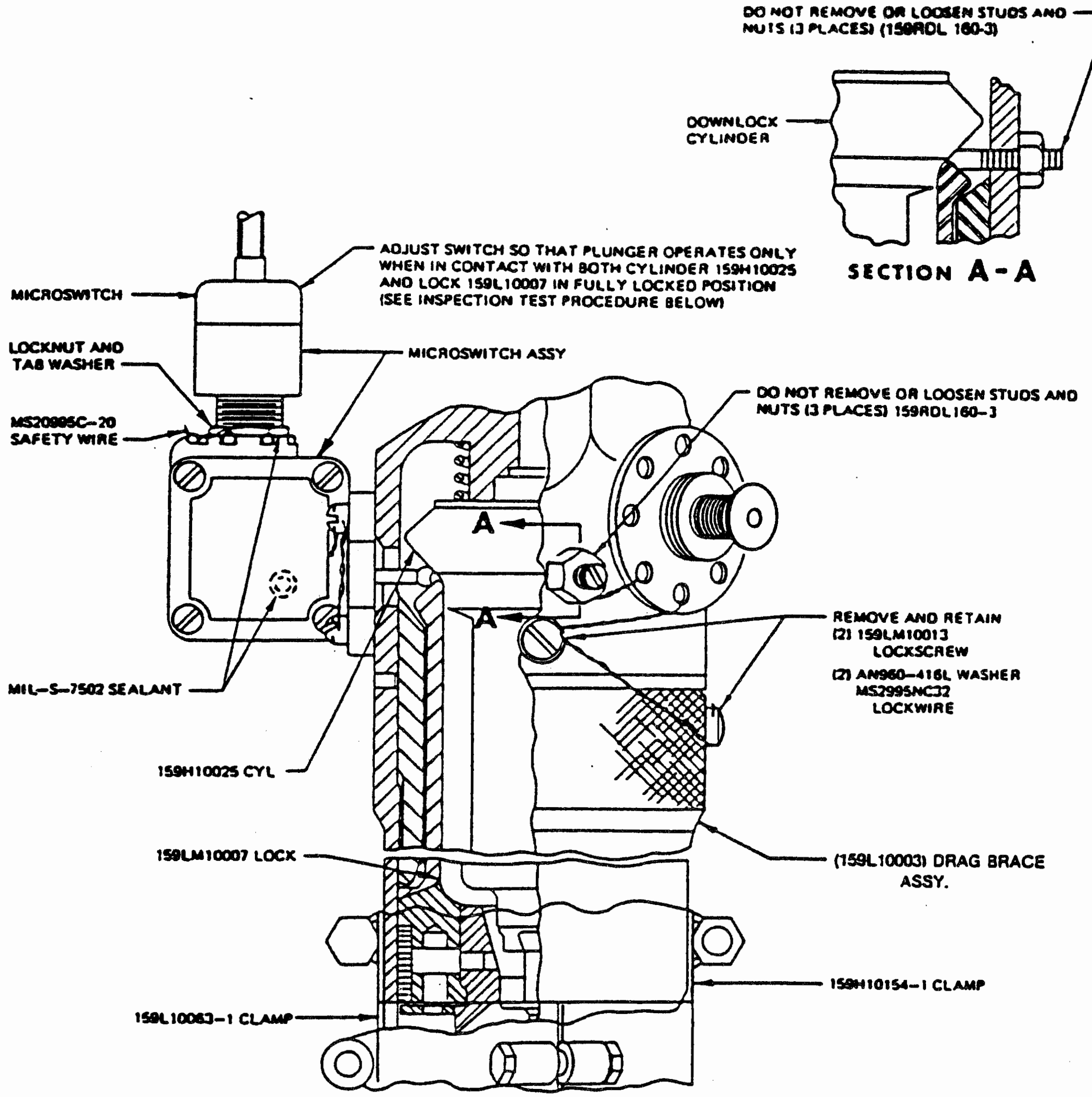
17. Main Gear Swivel Fitting — Lubrication

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: Clean all fittings prior to lubrication and wipe fitting and area clean of excess lubrication upon completion. Wipe all exposed piston rods of hydraulic actuating cylinders and shock struts with approved hydraulic fluid.



Main Drag Brace Installation
Figure 201.

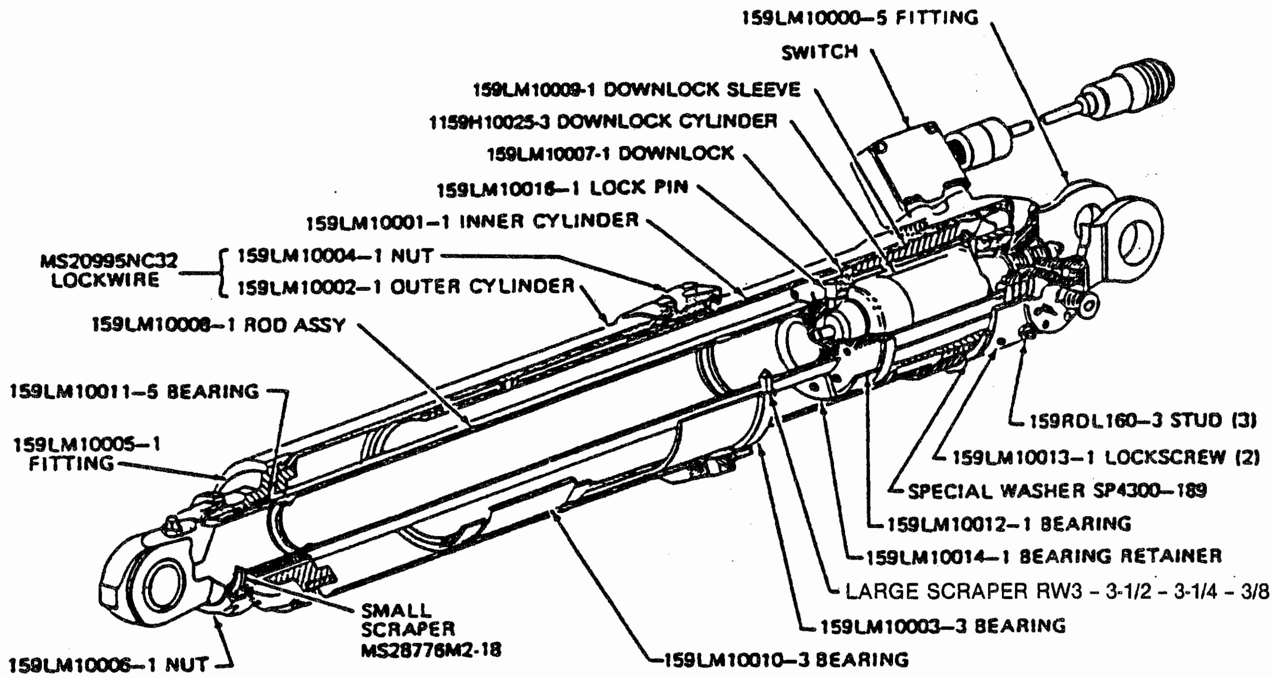


SWITCH INSPECTION TEST PROCEDURE

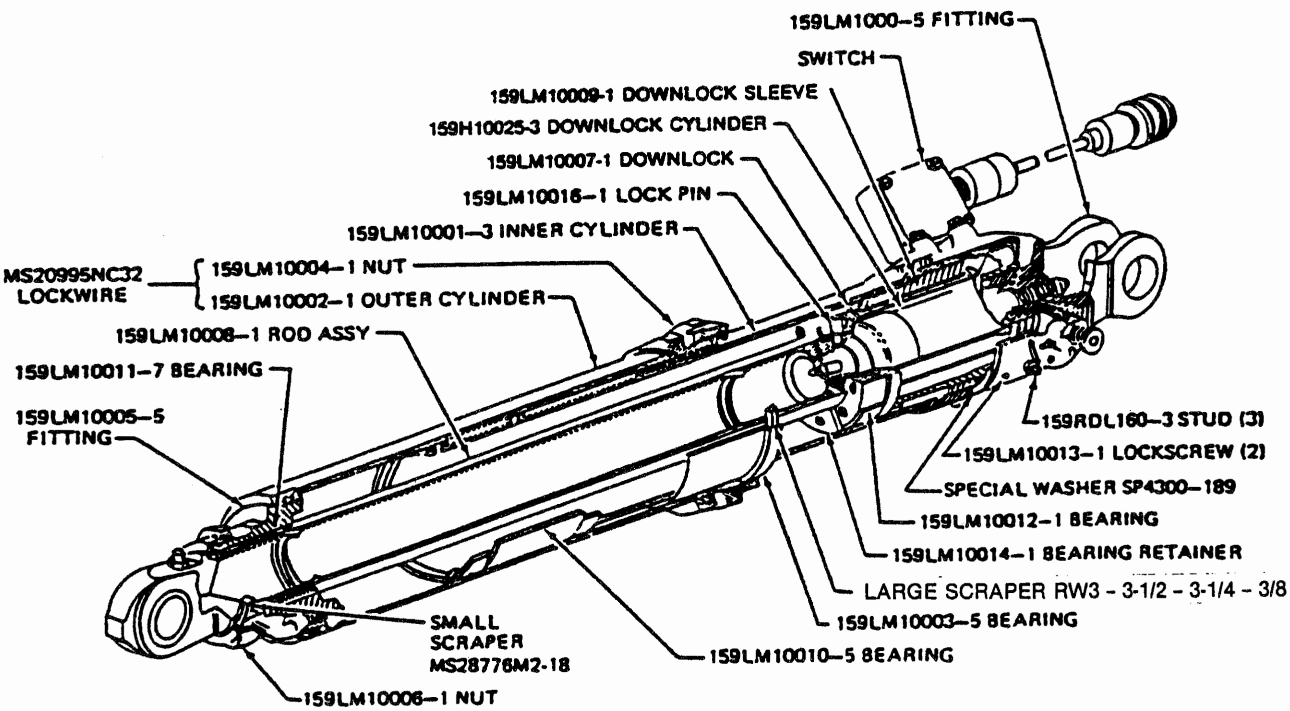
Each drag brace assembly 159L10003 must be inspected as follows for proper operation of microswitch:

(A) When cylinder 159H10025 is locked in full retracted position, switch circuit A - C should be complete.
(B) When cylinder is in other than locked position, switch circuit B - D should be complete.
(C) After switch is adjusted, tighten locknut and safety-wire to switch housing.

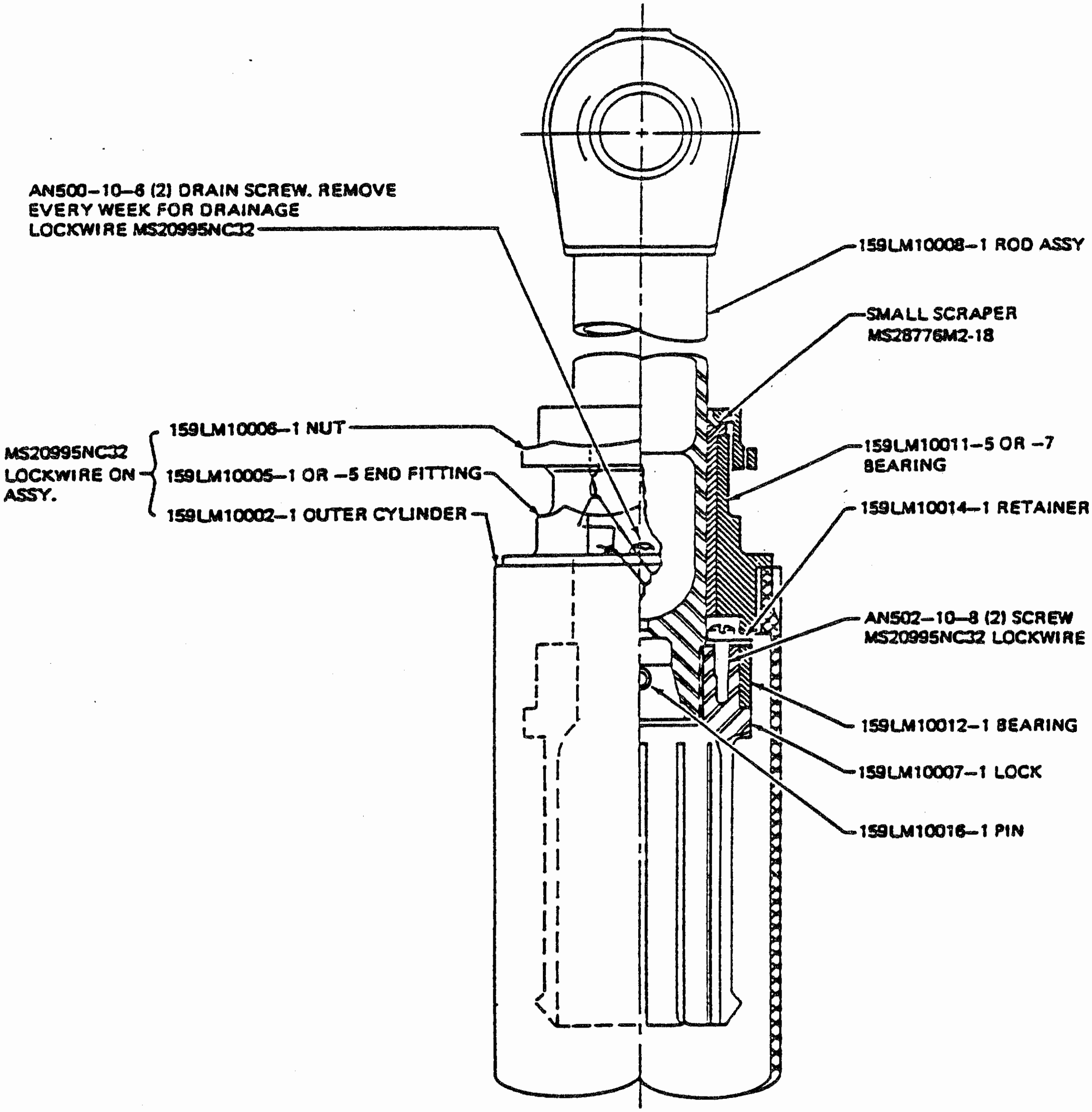
Main Drag Brace Switch Installation
Figure 202.



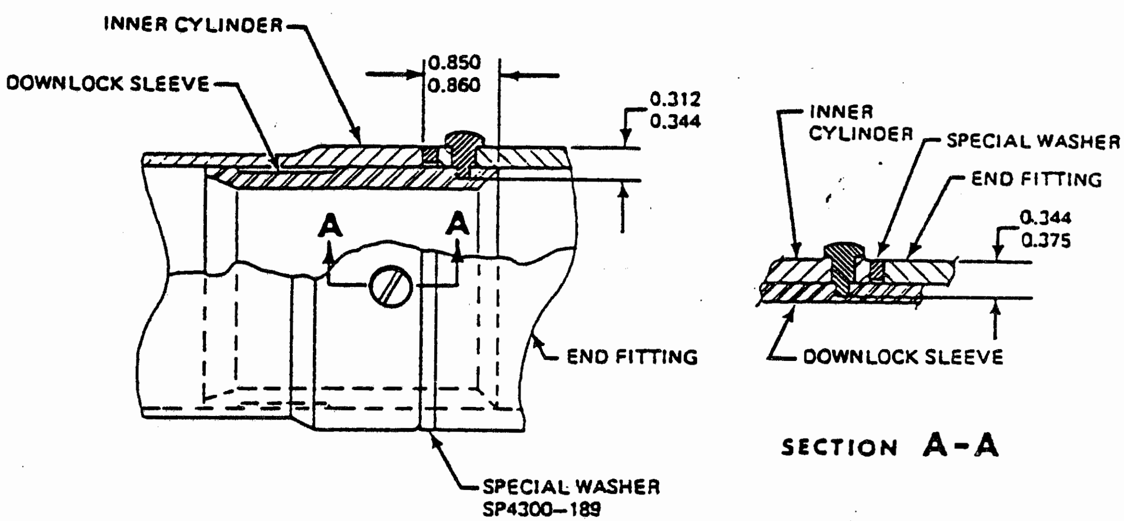
Drag Brace Assembly (159L10003-7)
Figure 203.



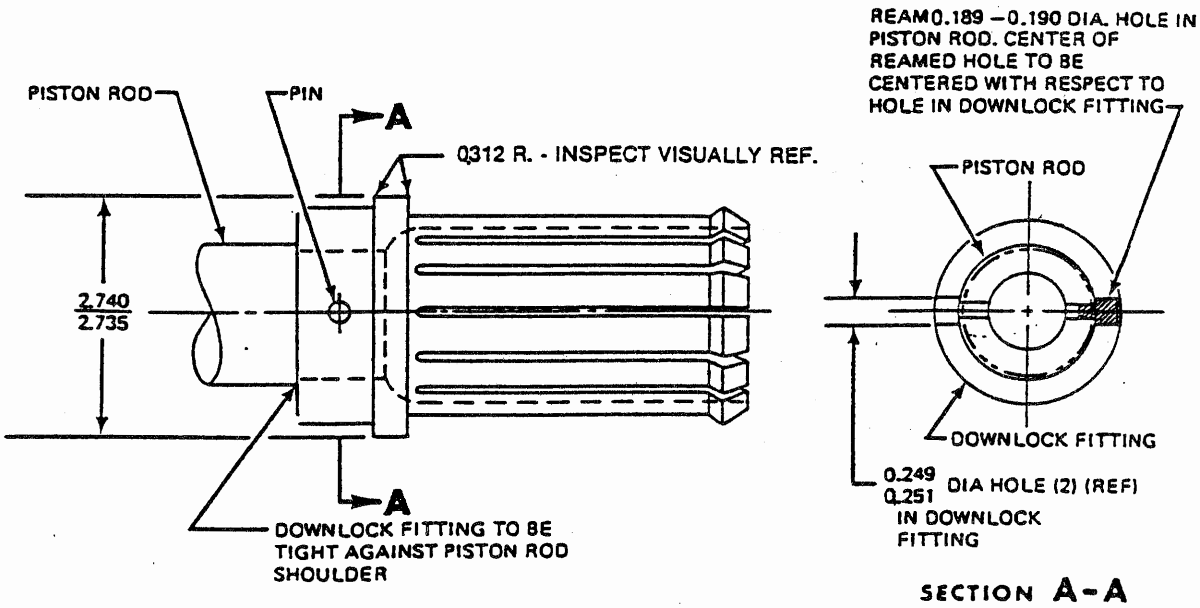
Drag Brace Assembly (159L10003-11)
Figure 204.



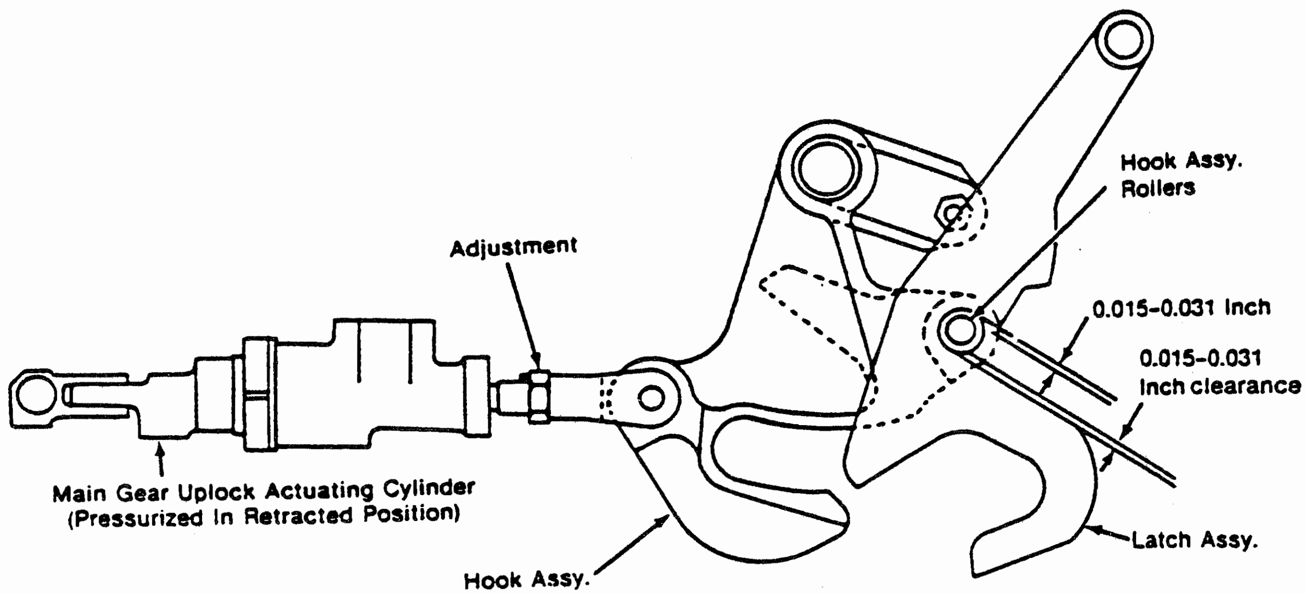
Main Gear Drag Brace Rod
Figure 205.



Inner Cylinder,Downlock Sleeve and End Fitting Assembly
Figure 206.

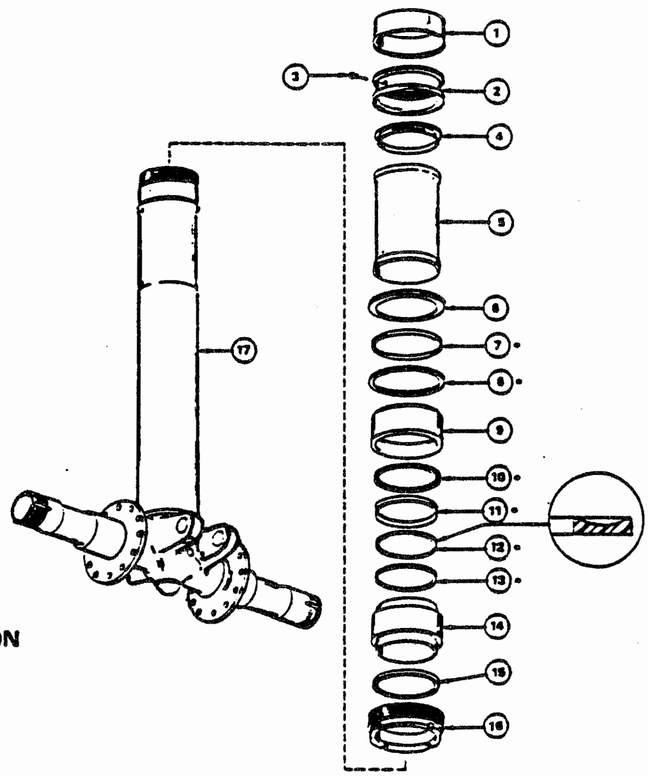


Piston Rod and Downlock Fitting Assembly
Figure 207.



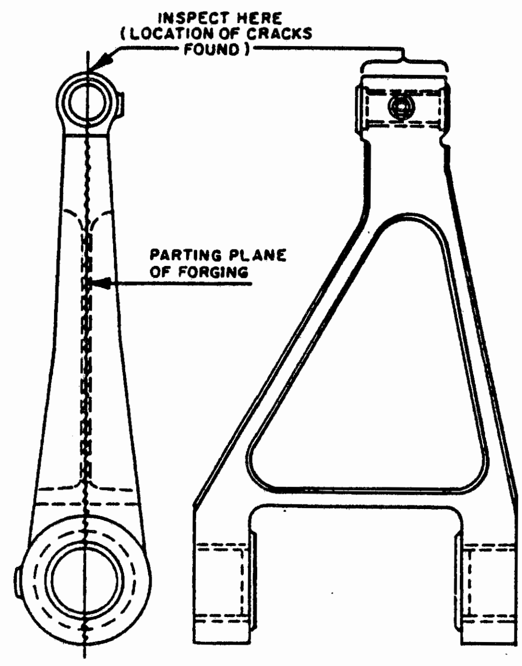
Main Gear Uplock Actuating Cylinder Installation / Adjustment
Figure 208.

1. Upper Bearing
2. Bearing Adapter
3. Locking Pin
4. Snubber Valve
5. Piston Extension Stop
6. Thrust Washer
7. Preformed Packing (O-ring)
8. Back-Up Ring
9. Packing Adapter
10. Back-Up Ring
11. Preformed Packing (O-ring)
12. Rubber Back-Up Ring
13. Back-Up Ring
14. Adapter - Lower Bearing
15. Wiper Ring
16. Packing Nut
17. Piston



• **CONTAINED IN MODIFICATION KIT, P/N 2572894**

Main Gear Shock Strut Seal Installation
Figure 209.



Main Landing Gear Torque Arm Assembly
Figure 210.

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NOSE LANDING GEAR — DESCRIPTION / OPERATION

1. **General**

The nose landing gear comprises two wheels and tires mounted on a conventional Bendix nitrogen-hydraulic shock strut housed in and attached to the fuselage structure. Further support is obtained from the drag brace which is attached to the midpoint of the shock strut and to fuselage structure. The gear retracts forward and up into the wheel well, extends aft and down. Before retraction starts, the nose wheel doors open, after completion of retraction, they close.

CAUTION: AS A SAFETY PRECAUTION, INSTALL NOSE LANDING GEAR GROUND SAFETY LOCK, 159GT1015, AND BOTH MAIN LANDING GEAR GROUND SAFETY LOCKS, 98GT1030, WHEN AIRCRAFT IS ON THE GROUND.

REMOVE ALL THREE GROUND SAFETY LOCKS BEFORE FLIGHT.

2. **Nose Gear Shock Strut**

The nose strut is a conventional nitrogen-hydraulic unit designed to absorb the shock of landing, takeoff and taxiing. It is essentially a steel piston riding in a cylinder and utilizing O-rings to prevent leakage between the piston and outer cylinder. The wheel axle is attached to the strut by two bolts. An air-oil filler valve is provided at the top of the strut and a placard on the nose wheel drag brace fairing indicates the correct strut pressures for given strut extensions. The strut is filled to overflow with 2 pints, 15 ounces of hydraulic fluid and pressurized with dry nitrogen to pressure specified on instruction plate attached to drag brace fairing. A jack point is provided at the base of the nose strut to permit removal of tires and wheels. Towing is accomplished by attaching tow bar 159GT1049 or equivalent to the axle. Mooring rings are provided for tie down.

NOTE: Use same type hydraulic fluid as is used in the aircraft hydraulic system.

3. **Nose Gear Drag Brace**

The nose gear drag brace connects to the shock strut at the lower end and to fuselage structure at the upper end. The drag brace restrains the shock strut from pivoting about its trunnion and aids in supporting the strut in the extended position. The nose gear drag brace, unlike the main gear drag braces, does not extend or retract. (Aircraft having ASC 226 use a two section drag brace which incorporates a mechanical fuse).

CAUTION: PROVISIONS FOR A PIP PIN ARE PROVIDED ON THE MECHANICAL FUSE. PIP PIN MUST BE REMOVED BEFORE FLIGHT.

4. **Nose Gear Selector Valve**

This valve is a manually controlled, two position, four way, rotary valve. It is located at the top of the nose wheel well and is controlled by the pilots and copilots landing gear control handle.

See Landing Gear Hydraulic System

The nose gear selector has two flow positions. In position 1, (landing gear down configuration) pressure from the main power system permits fluid to be directed to the cylinder 2 port, and allows pressure from the cylinder 1 port to return to the reservoir. In position 2, (landing gear up configuration) pressure from the main power system permits fluid to be directed to the cylinder 1 port, and permits pressure from cylinder 2 port to return to the reservoir.

5. **Nose Gear Strut Friction Damper**

(Aircraft Having ASC 168)

This Optional Service Change provides a friction damper for nose gear strut.

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NOSE LANDING GEAR — MAINTENANCE PRACTICES

1. Nose Gear Shock Strut — Service

The filling procedure is same as for main gear shock strut except that the nose strut is inflated to 121 psi in the fully extended position.

2. Nose Gear Uplock Roller Bolt — Removal / Installation

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Removal

(1) Jack aircraft.

(2) Remove wheels.

(3) Remove uplock roller and bolt assembly.

B. Installation

(1) Install parts on strut. Ensure lubricant fitting on stud is in accessible position for periodic lubrication.

CAUTION: ENSURE THAT ROLLER CAN BE TURNED FREELY AND THAT CLAMP-UP BUSHING AND BOLT DO NOT TURN.

(2) Torque nut to 95-100 inch-pounds.

(3) Install cotter pin in main and nose gear uplock roller bolts and nuts.

(4) Lubricate as required.

(5) Install wheels

3. Nose gear Uplock Roller Bolt — Inspection

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

A. Clean parts with cleaning solvent, ensure all grease passages are clear.

B. Inspect all parts for galling. Minor imperfections can be polished out with fine emery cloth.

C. Inspect clamp up bushings and roller bolt assembly for cracks. Replace if necessary.

D. Install parts on strut. Ensure lubricant fitting on stud is in accessible position for periodic lubrication.

4. Nose Gear Strut Actuator Mounting Adapter — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Disconnect hydraulic power and deplete hydraulic system pressure.

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(2) Jack aircraft.

(3) Disconnect nose gear actuator piston from mounting adapter at nose strut trunnion.

(4) Separate connecting link from retract lever.

(5) Remove attaching hardware and separate retract lever from actuator mounting adapter.

(6) Remove mounting adapter.

NOTE: On Aircraft 1 - 23 and 114, actuator adapter magnaflux should be done at this time if due.

B. Installation

(1) Apply a coating of grease on shaft of mounting adapter.

(2) Install adapter. Assemble retract lever and secure with attaching hardware.

(3) Attach connecting link to retract lever.

(4) Connect nose gear piston to mounting adapter.

(5) Lube all fittings with grease, MIL-G-23827.

(6) Lower jack.

5. Nose Gear Strut Actuator Mounting Adapter — NDT Inspection

NOTE: The following Inspection/magnaflux procedure applies to adapter P/N 171552, which can be used with nose gear shock/strut P/N 159SCL104-1 or -15. This procedure does NOT apply to adapter P/N 173524 which will be done at nose overhaul on strut P/N 159SCL104-5, -7, -11 or -15 (See Figure 201)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove Adapter. (See this section)

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

B. Thoroughly clean all removed part and adapter mounting boss with cleaning solvent.

C. Inspect all parts for wear and visual signs of cracks. Replace bushings if wear appears to be excessive.

D. Magnaflux mounting adapter. If either mounting adapter or retract lever are replaced, line ream existing 0.429-0.431 inch holes in new part(s) to 0.4365-0.4380 inch.

E. Install adapter. (See this section)

6. Nose Gear Trunnion and Shock Strut — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Disconnect electrical connector at downlock switch.

(3) Remove hydraulic power from aircraft.

(4) Disconnect nose gear steering cables and hydraulic lines; cap exposed lines and ports.

(5) Disconnect nose gear actuating cylinder and tie out of way.

(6) Disconnect drag brace fairing support rods and tie out of way.

(7) Disconnect drag brace at shock strut.

(8) Disconnect down lock mechanism from trunnion.

WARNING: IF STRUT IS BEING REPLACED, IT MUST BE DEFLATED BY BLEEDING PRESSURE FROM FILTER VALVE USING SAFE, RECOMMENDED PRACTICES.

CAUTION: DO NOT ALLOW STRUT TO SWING BEYOND DOWN AND LOCKED POSITION DURING REMOVAL. FORCE EXERTED ON LANDING GEAR DOOR CONTROL OPERATING LEVER WILL CAUSE OVERTRAVEL OF DOOR VALVE. THIS WILL RESULT IN SHEARING ITS STOP PIN. DISCONNECT LINKAGE TO LEVER ARM OF DOOR CONTROL VALVE.

(9) Support shock strut and remove bolt from shock strut/trunnion attach point.

(10) Remove shock strut.

(11) Remove trunnion pins after removal of trunnion bolts.

B. Installation

(1) Install trunnion (do not install shims at this time). Ensure that total end play between trunnion and fuselage bushing face does not exceed 0.024 inches. If play exceeds this limit, replace fuselage bushings.

(2) Shim between trunnion and fuselage bushing, to maintain an end play of 0.001 inch minimum to 0.003 inch maximum. (See Figure 202). (Use shims 159LM10135-1, -3 or -5 on right side only).

NOTE: On Aircraft 173 - 200 maintain 0.001 inch minimum to 0.003 inch maximum end play (use shims 159LM10135-1, -3, -5 or -7 on right side only). Trunnion pins must be free and permit rotation of trunnion swivel joint throughout its travel. Torque trunnion nut (159RDL180-11) 480 to 590 inch-pounds and safety.

(3) Install shock strut. Torque shock strut to trunnion, attach bolt to 4200 inch-pounds (350 foot-pounds) maximum.

(4) Attach drag brace to shock strut and attach rods to drag brace fairing.

(5) Connect nose gear actuating cylinder and connect linkage to lever arm of door control valve.

(6) Connect nose gear steering cables; with handle and wheels in neutral, tension cables to 20 +0 −5 pounds. Safety-wire turnbuckles.

NOTE: For every 10°F above 70°F, add 1.5 pounds of tension. For every 10°F below 70°F, subtract 1.5 pounds of tension.

(7) Connect hydraulic lines removed in Step A.(4) above.

(8) Install electrical connector at downlock switch.

(9) Adjust downlock switch.

(10) Lubricate nose strut fittings as required.

(11) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(12) Perform Nose Wheel Steering System — Operational Test. (See Section 32-11-0)

(13) Check for hydraulic fluid leakage and service hydraulic reservoir.

(14) Lower jacks.

7. Nose Gear Selector Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Connect external source of hydraulic pressure to aircraft.

(3) Locate nose gear door selector valve in nose wheel well upper bulkhead. Place a suitable container with a minimum capacity of five gallons near area where work is performed.

(4) Remove ground safety lock from nose gear. Unlock nose gear downlock manually. Raise nose gear until upper trunnion is no longer in contact with striker. Support nose gear in that position.

(5) Disconnect rod assembly from door control valve arm.

(6) Remove valve arm from valve.

CAUTION: SLOWLY REMOVE FILLER CAP FROM RESERVOIR TO BLEED OFF ANY RESIDUAL PRESSURE IN RESERVOIR.

(7) Disconnect hydraulic lines and fittings from valve and plug openings.

(8) Remove mounting screws and remove valve from mounting bracket.

B. Installation

(1) Install valve arm on valve shaft.

(2) Position valve on mounting bracket and install mounting screws. Install lockwire where required.

(3) Place nose gear uplock in open position before proceeding.

(4) Place rod assembly in position, bolt should fit freely without binding. If bolt binds, disconnect bungee rod at terminal end and adjust. Reconnect rod assembly to arm and bungee rod to striker.

(5) Remove plugs and install hydraulic lines and fittings to valve assembly.

(6) Remove support from nose gear and place gear in down and locked position. Door control valve should shift to the door close position.

(7) Check hydraulic reservoir fluid level and service as required.

(8) Remove all ground safety devices and perform a Landing Gear System — Functional Test. (See Section 32-0)

(9) Replace ground safety devices.

(10) Remove external source of hydraulic pressure from right engine nacelle.

(11) Lower jacks.

8. Nose Gear Strut Piston / Seals — Removal / Installation

(See Figure 204)

A. Removal

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Jack aircraft.

(2) Remove ground safety lock from nose gear strut.

(3) Retract strut to approximately 45° of full retraction using external hydraulic power or auxiliary hydraulic pump (if having ASC 109, auxiliary to main selector) Shutdown hydraulic power.

WARNING: DO NOT LOOSEN VALVE BODY FROM STRUT UNTIL ALL PRESSURE HAS BEEN RELEASED. SERIOUS INJURY OR LOSS OF LIFE CAN RESULT IF PERSONNEL ARE STRUCK BY VALVE ASSEMBLY EJECTED UNDER PRESSURE.

(4) To release pressure from strut as follows: Check 3/4 inch swivel nut for tightness, remove valve cap and loosen swivel nut slowly to bleed nitrogen from strut (maximum 3 1/2 turns). Allow all pressure to be released, then remove filler valve.

NOTE: Before any strut disassembly, take care to accomplish Steps (5) thru (8) below.

(5) Ensure that the area around Schrader valve port is clear of personnel as additional pressure/fluid may be expelled during the following steps.

(6) Remove cotter pin (19) from orifice support tube retaining nut (20).

(7) Using a socket wrench, slowly loosen the retaining nut.

CAUTION: WHEN RETAINING STEM SEPARATES FROM NUT, A BOLT WILL PROBABLY BE FELT AND PRESSURE/FLUID WILL BE EXPELLED THROUGH SCHRADER VALVE PORT AND THROUGH RETAINING STEM HOLE IN THE TOP OF THE STRUT.

NOTE: If firm nut turning resistance does not continue, nut will back off retaining stem until it disengages. Retaining stem will remain protruding through top of the strut.

If firm nut turning resistance continues, it may be an indication that pressure exists in cavity at the upper end of strut. If this pressure is present, the orifice support tube retaining stem (28) should retract inside nut until it separates from nut. Do not remove socket from nut until the nut disengages.

(8) Using a soft hammer and a soft punch, (i.e. brass or equivalent), tap the orifice support tube retaining stem into strut until the orifice support tube seal is below the Schrader valve port. Total movement of stem will be approximately 1 1/8 inches. If any pressure is present, it will be expelled at this time.

(9) Remove pip-pin and separate torque arm from steering unit.

(10) Remove lockplate at packing nut (13) on struts not having ASC 168. On struts having ASC 168, remove key (17).

(11) Provide a drain pan under strut assembly.

CAUTION: COVER WHEELS WITH HYDRAULIC FLUID RESISTANT COVER BEFORE DRAINING FLUID FROM STRUT.

(12) Remove packing nut (13) on struts not having ASC 168.

(13) On struts having ASC 168, remove gland nut (18), adapter (16), packing (15) and packing nut (14).

(14) Slowly withdraw piston assembly(with wheels installed from cylinder and drain fluid from piston.

(15) Remove bearing sleeve (1), pin (2), bearing adapter (3), then remove snubber (4), and upper and lower cam assemblies (5 and 6), then remove remainder of parts (7 thru 13).

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(16) Clean all metal parts and exposed piston surfaces with cleaning solvent and dry with clean filtered air.

B. Installation

NOTE: Install and secure orifice support tube in the following manner be fore installation of piston.

(1) Remove any sealant that may be on top of cylinder, orifice support tube retaining stem, nut and washer.

(2) Install a new packing into groove of orifice support tube and install a new backup ring above packing (22 and 23).

(3) Carefully insert orifice support tube into cylinder with stem extending through hole at top of cylinder.

(4) Install washer onto stem and secure with nut. Tighten nut and torque to 100 $\pm$ 10 inch-pounds. Back off nut to align nearest cotter pin hole.

NOTE: Do not apply any sealant to orifice support tube retaining stem, nut or washer.

(5) Install cotter pin.

(6) Before assembly apply a generous coating of grease, HI-LO MS No. 1 to all rubber rings and sleeve bearings (9 and 11)

(7) On struts not having ASC 168, carefully slide packing nut (13), wiper ring (12) and bearing adapter on piston. On struts having ASC 168, carefully slide packing nut (14), packing (15), adapter (16) and gland nut (18) on piston.

(8) Assemble O-ring (10) on sleeve bearing (9), then assemble backup ring (8) and O-ring (7), on outer diameter of sleeve bearing (9). Install assemblies on piston.

(9) Carefully install upper and lower cam assemblies (5 and 6) on piston. Seat upper cam on shoulder at outer diameter of piston just below threads. Then install snubber (4).

(10) Install bearing adapter (3), pins (2) and bearing sleeve (1).

(11) Slide lower cam assembly (6) on piston to engage with upper cam (5). Wedge lower cam in place with a long piece of micarta or similar material so that it can be withdrawn later.

(12) Carefully install piston into cylinder so that machine keys slide into keyways of outer cylinder. Remove micarta wedge and slide piston until it seats in outer cylinder.

(13) Install packing nut (13) and tighten. Then install lockplate with screws on strut not having ASC 168.On struts having ASC 168, install packing nut (14), packing (15), packing adapter (16) and gland nut (18), then install key (17) and safety-wire.

(14) Align torque arm of piston with steering unit and install pip pin.

(15) Extend strut to down and locked position and install ground safety lock.

(16) Service shock strut. If different type hydraulic fluid is used, revise data plate to reflect type used.

(17) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(18) Lower jacks.

9. Nose Strut Friction Damper Torque Check — Inspection

WARNING: PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR. BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Raise nose of aircraft with fuselage jack. Place nose strut jack under jack pad and grease pad and remove wheels.

WARNING: DO NOT LOOSEN VALVE BODY FROM STRUT UNTIL ALL PRESSURE HAS BEEN RELEASED. SERIOUS INJURY OR LOSS OF LIFE CAN RESULT IF VALVE ASSEMBLY IS EJECTED UNDER PRESSURE.

B. Deflate strut and close off air filler valve.

C. Place torque wrench on (GN10U-820) nut on jack pad and check friction torque of inner cylinder, 450-500 inch-pounds required to rotate inner cylinder within outer cylinder.

D. Adjust 159RDL169-7 adapter to obtain 450-500 inch-pounds torque wrench reading at jack pad. Refer to Figure 205.

E. Inflate strut slowly and check inflation pressure necessary to start and fully extend inner cylinder from

13.5 to 14.5 inch dimension "X" position. Pressure should be 40-55 psig. Inflate to operating pressure. Maintain 400-500 inch-pounds friction torque during servicing period.

NOTE: To prevent damage to key slots and threads when torque value is being obtained. Temporarily key nut, 159RDL169-19, and adapter 159RDL169-7 with key 159RDL169-17 and use four inch wrench.

F. Safety as required.

G. Service strut.

H. Install nose wheels.

I. Lower jacks and remove jacks and pads.

10. Nose Gear Strut Trunnion — NDT Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: This inspection applies to Aircraft with Bendix nose gear strut assembly 172240 (159SCL104-1) or 2570890 (159SCL104-13), trunnion P/N 171468 / 173513 / 171467.

A. Remove nose gear trunnion assembly. (See this section)

WARNING: THE APPROPRIATE PRECAUTIONS TO ENSURE THAT STRIPPER DOES NOT COME IN CONTACT WITH SKIN OR EYES. USE ONLY IN WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

B. Clean and strip paint from trunnion.

C. Inspect retract shaft attachment lugs on nose gear trunnion forward flanges for cracks in radii above and below area where attachment lugs blend into trunnion flanges; use dye penetrant or zyglo method (NDT) with a ten power or greater glass or approved equivalent.

NOTE: If cracks are found, replace trunnion assembly.

D. Apply apply approved alodine, primer and paint to assembly.

E. Install trunnion assembly. (See this section)

11. Nose Gear Outer Cylinder — Inspection

(See Figure 207)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: Inspection of the outer cylinder is recommended by use of Zyglo or an equivalent dye penetrant inspection procedure (NDT).

A. Mask area to be inspected.

WARNING: TAKE APPROPRIATE PRECAUTIONS TO ENSURE THAT STRIPPER DOES NOT COME IN CONTACT WITH SKIN OR EYES. USE ONLY IN WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(1) Strip epoxy paint from affected area by applying stripper by brush method.

(2) Perform dye penetrant inspection.

(3) Clean surface to be repainted with clean cloth saturated with toluene. Wipe area dry with clean cloth. It is imperative that all surfaces to be painted are free of contamination or poor paint adhesion will result.

(4) Prime surface with epoxy primer, allow to dry and apply epoxy white paint.

12. Nose Gear Torque Arm — Removal / Installation

A. Removal

(1) Remove steering unit pin and secure unit.

(2) Remove bolt connecting torque arm to strut.

(3) Remove torque arm.

B. Installation

(1) Install torque arm.

(2) Install bolt connecting torque arm to strut.

(3) Install steering unit pin.

13. Nose Gear Torque Arm — NDT Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove torque arm. (See this section)

WARNING: USE SOLVENT IN WELL VENTILATED AREA, AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

B. Clean and remove paint from entire torque arm with stripper (See Figure 208)

C. Perform dye check or Zyglo inspection (NDT) on torque arms (part number 171417 or 86336 and 86342 / 2580490) in areas as shown in Figure 208.

NOTE: If cracks are found replace assembly.

D. Upon completion of inspection (NDT). apply approved alodine, primer and paint to assembly.

E. Install torque arm (See this section)

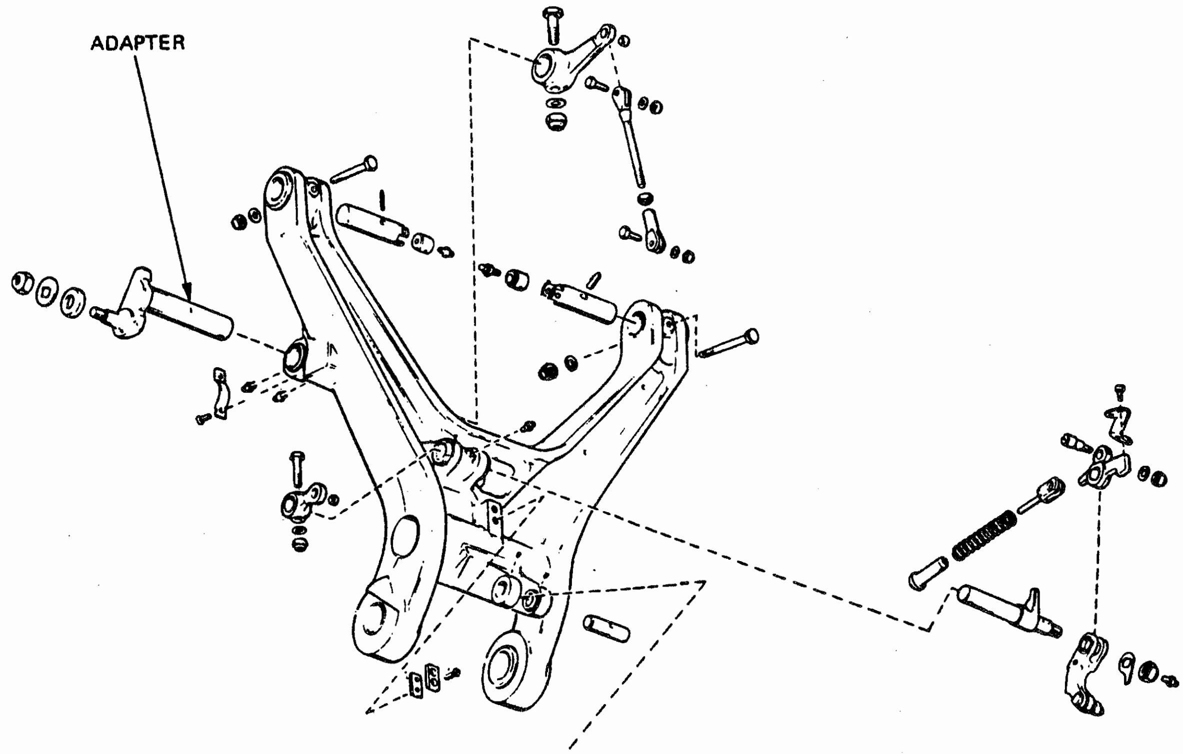
14. Nose Landing Gear Retract Cylinder Fuselage Attachment Structure — Inspection

A. Clean fuselage attachment structure.

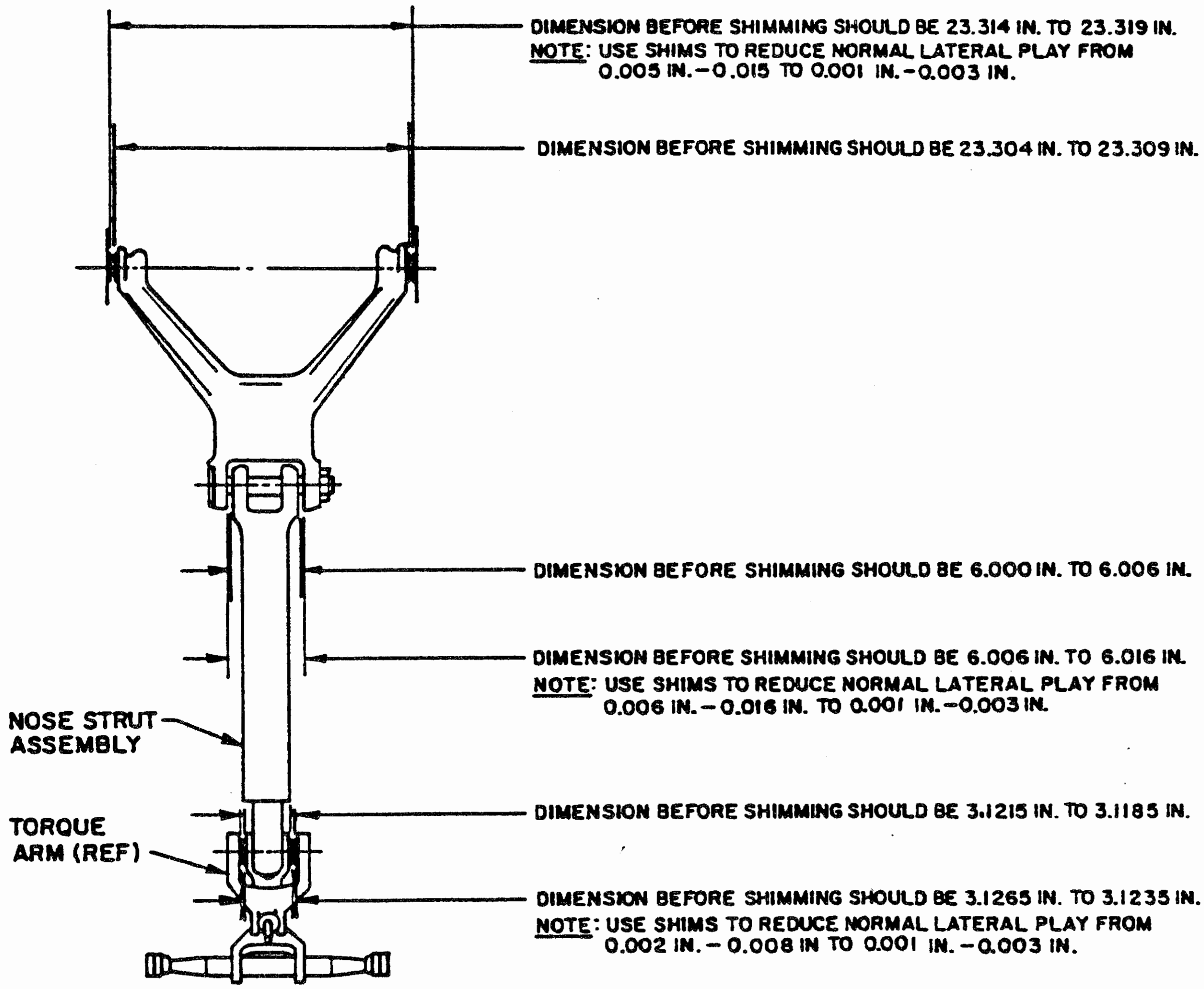
B. Visually inspect structure for cracks as shown in Figure 209.

C. If cracks are detected, replace parts and hardware within 10 flight hours.

D. Return aircraft to flight status.

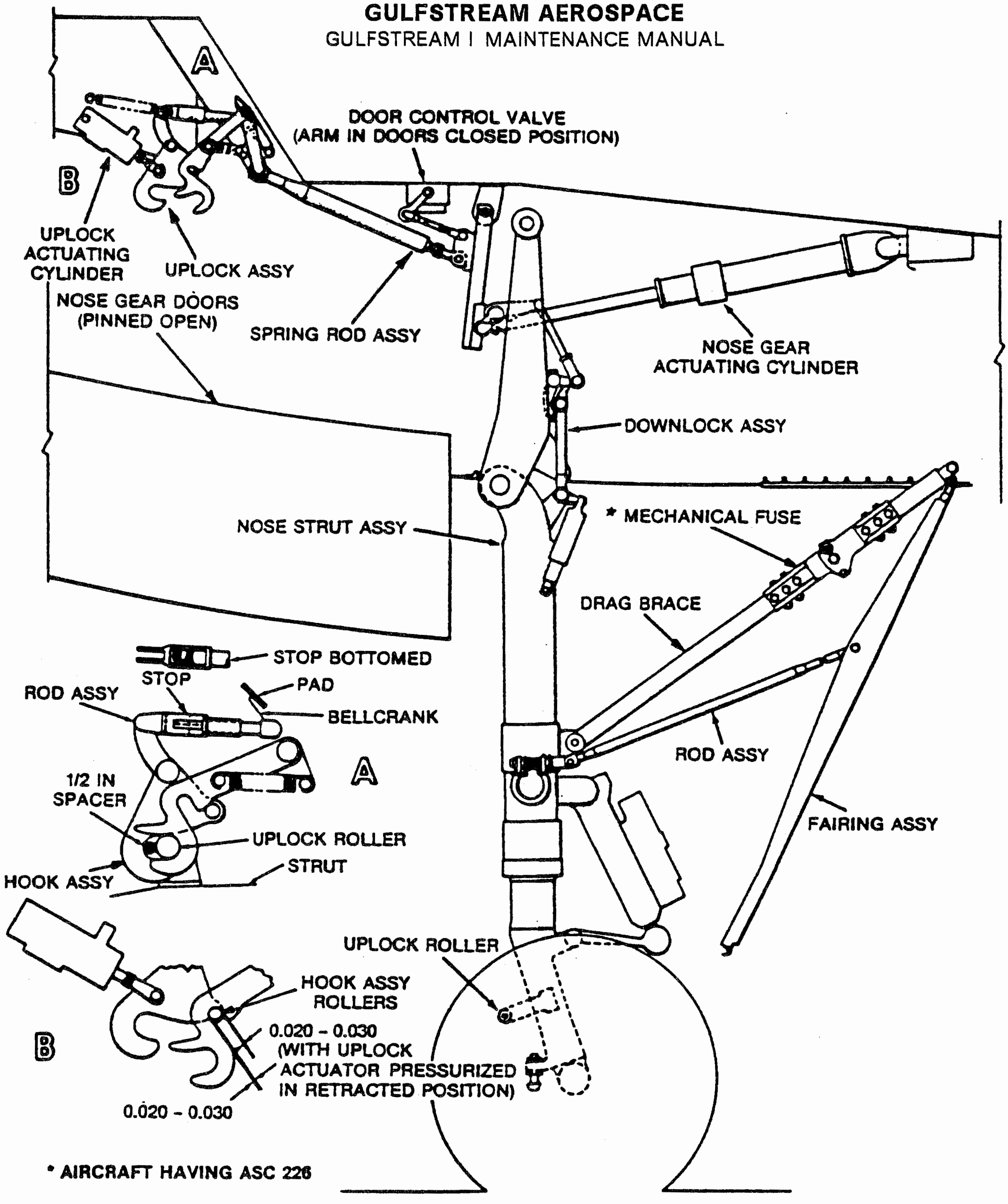


Nose Gear Strut Actuator Mounting Adapter
Figure 201.

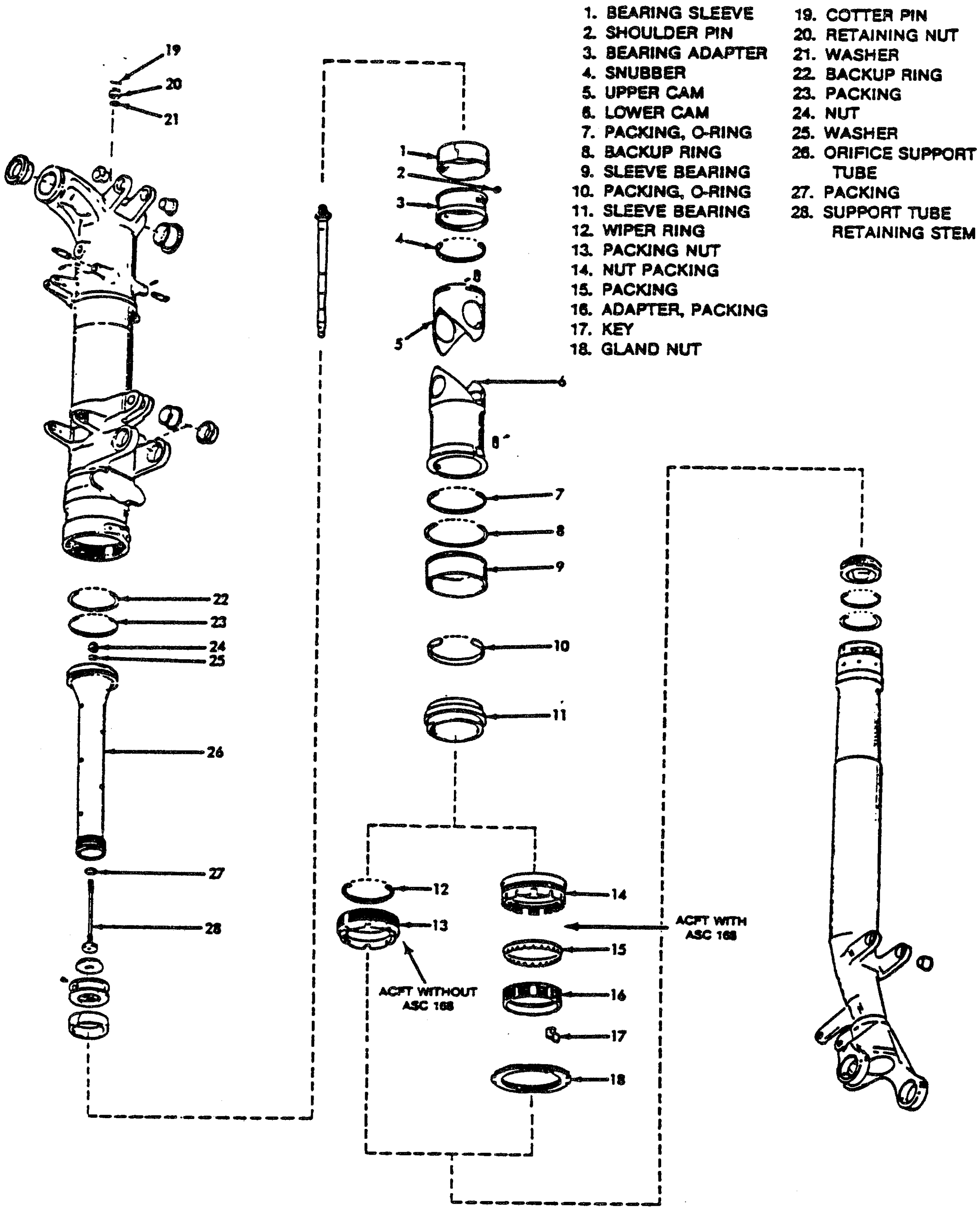


NOTE: ALL JOINTS MUST BE FREE TO ROTATE AFTER SHIMMING.

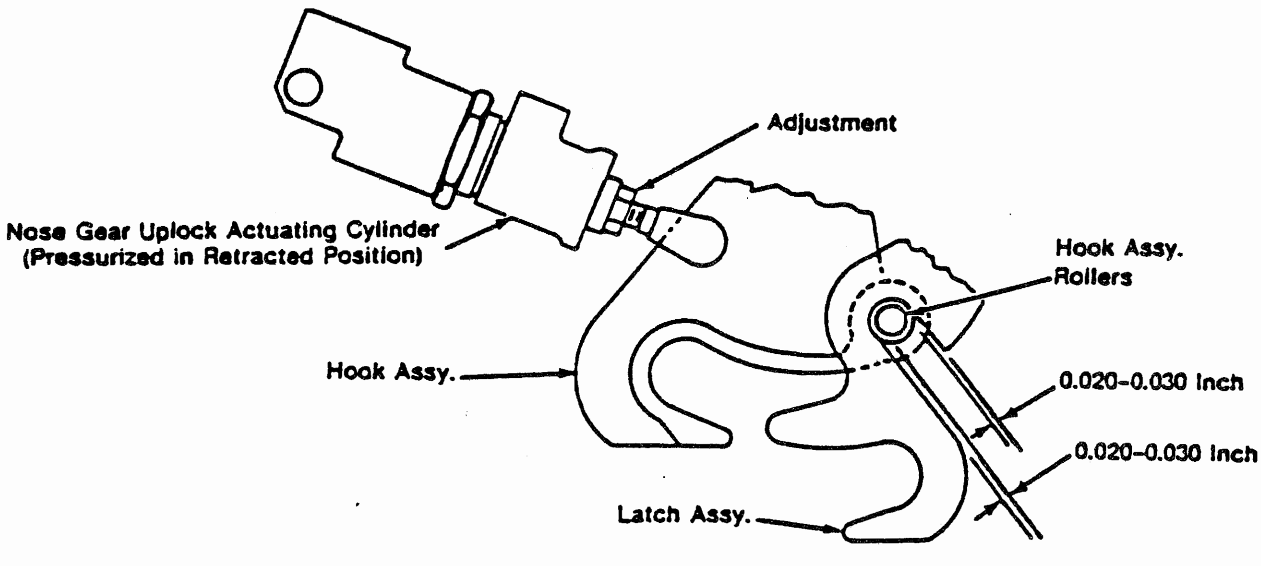
Nose Gear Trunnion and Shock Strut Shimming
Figure 202.



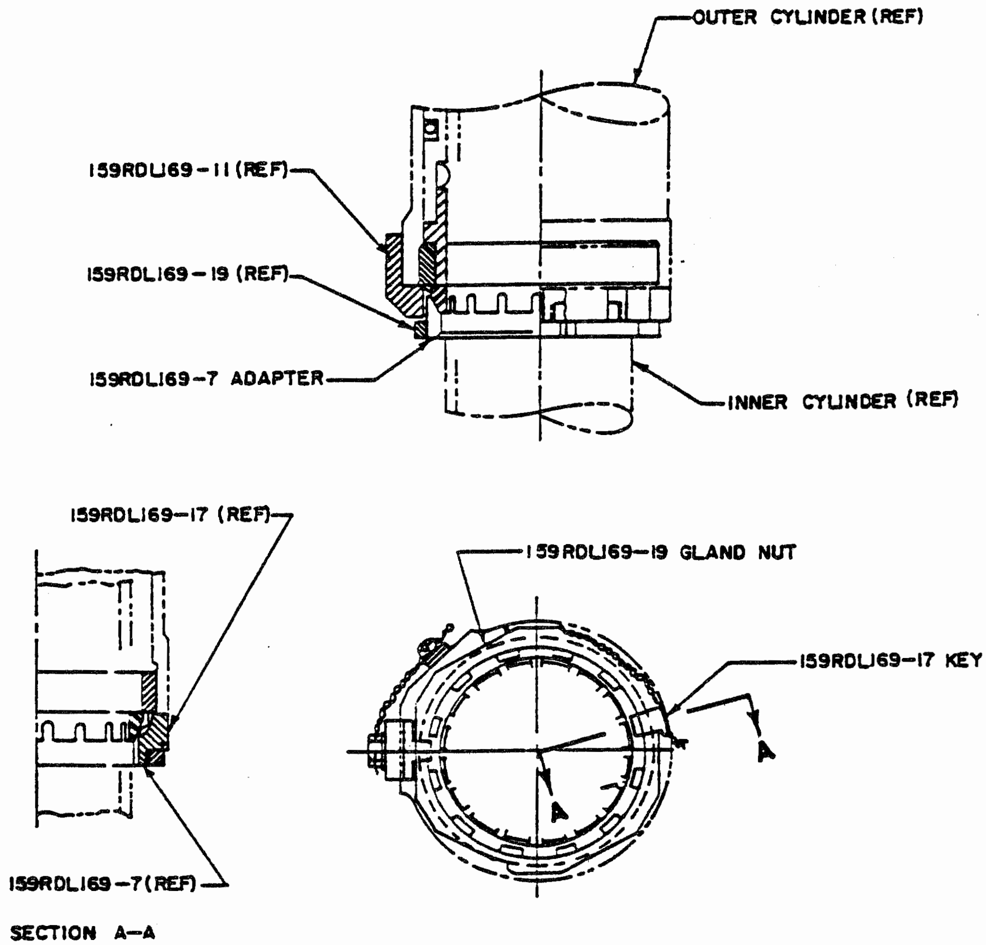
Nose Gear Uplock — Adjustment
Figure 203.



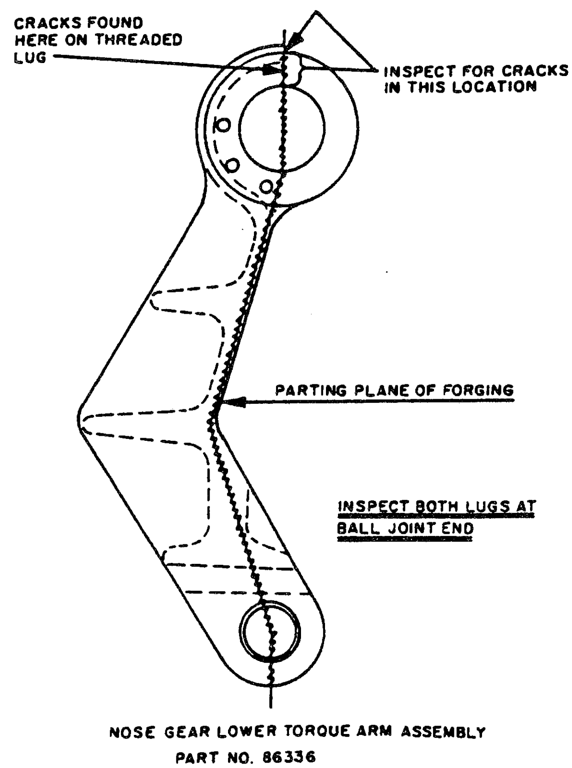
Nose Gear Strut
Figure 204.



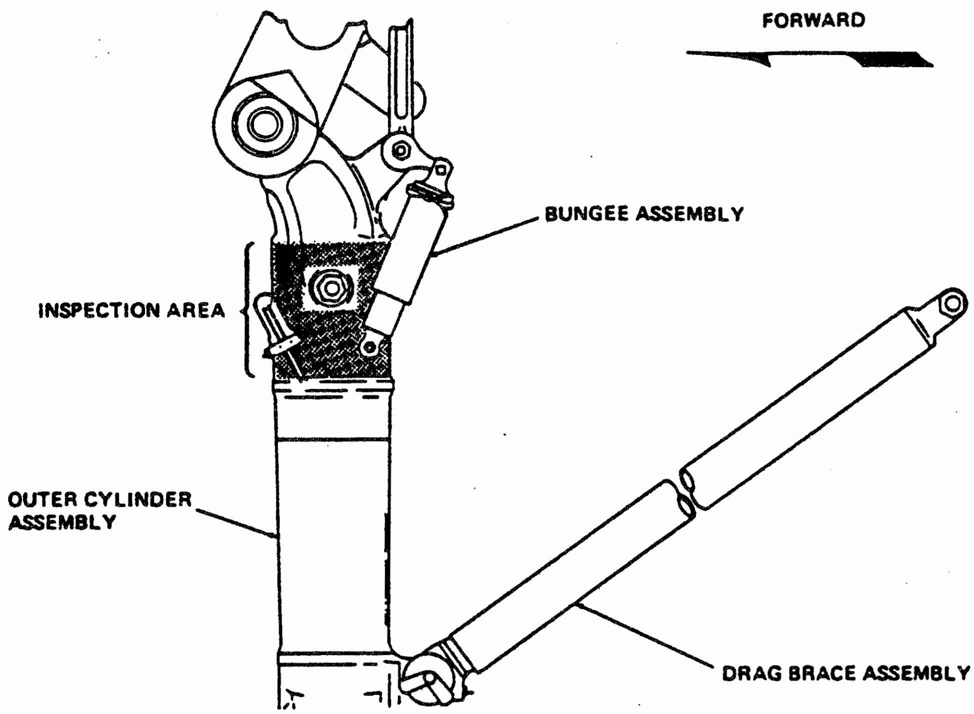
Nose Strut Friction Damper
Figure 205.



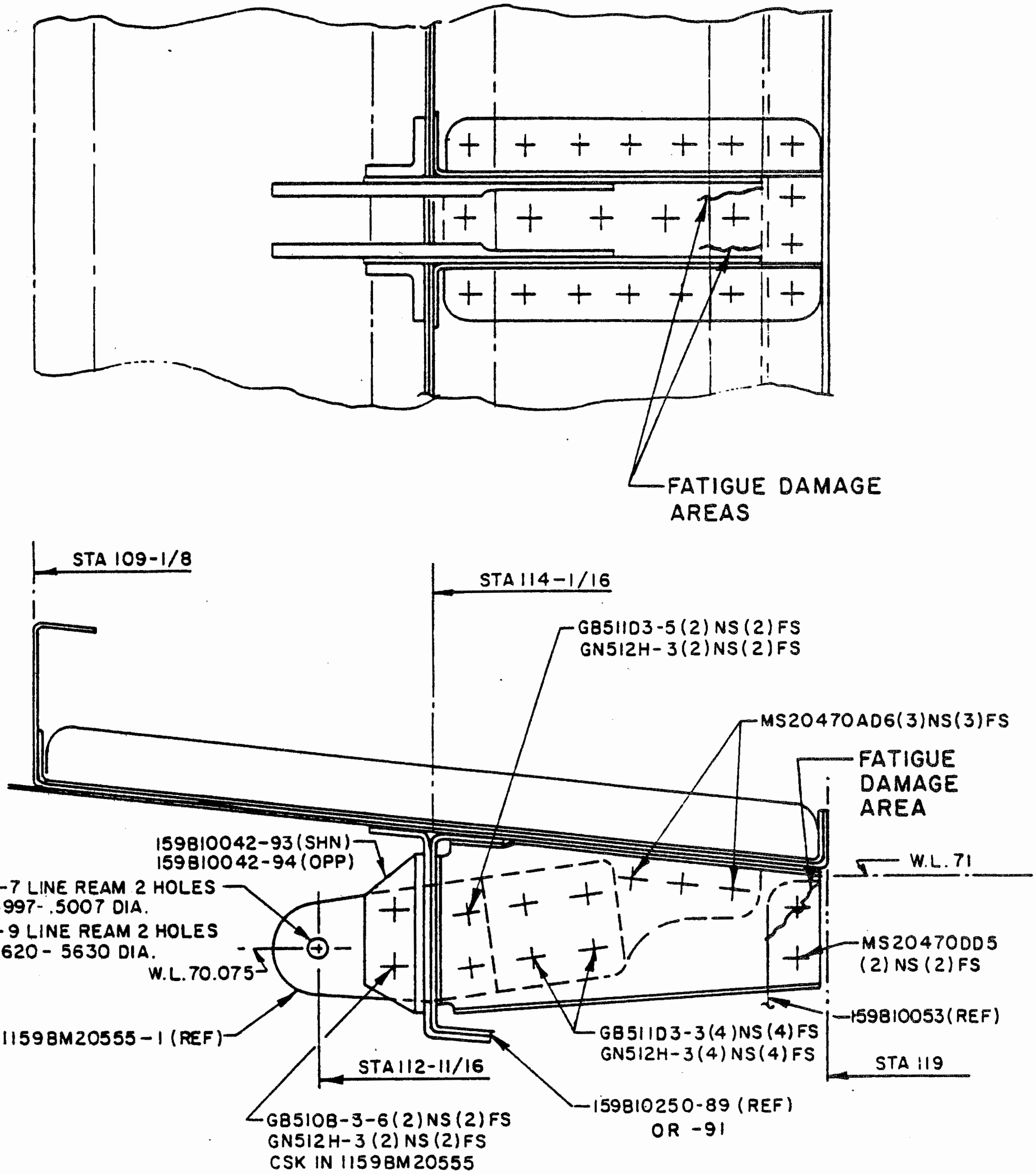
Nose Gear Uplock Actuating Cylinder Installation / Adjustment
Figure 206.



Nose Gear Lower Torque Arm Assembly
Figure 207.



Nose Gear Outer Cylinder — Inspection
Figure 208.



Nose Gear Retract Cylinder Fuselage Attachment Structure
Figure 209.

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LANDING GEAR DOORS — DESCRIPTION / OPERATION

1. Description

(See 32-0, Figure 1 and Figure 2)

Each main gear is enclosed by two forward (clamshell) and two aft doors and a fairing door attached to the drag strut. All are of rib and beam construction. Each door contains two hinges which mount the door to the nacelle. A hydraulic door cylinder actuates the clamshell doors. Others are operated by mechanical attachment to the strut.

The nose gear is enclosed by two clamshell doors and a faring door attached to the drag strut. All are of rib and beam construction. Each door contains two hinges which mount it to the fuselage. The faring door is attached to the aircraft with a single fitting. A hydraulic door cylinder actuates the clamshell doors. The faring door is mechanically attached to the strut.

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LANDING GEAR DOORS — MAINTENANCE PRACTICES

1. **Main Gear Door Trunnion / Swivel / Rod End — Inspection**

   (Aircraft 1 - 111 and 114 Not Having ASC 164)

   (See Figure 201)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

NOTE: The door swivel assembly and trunnion door rods should be inspected for 360° freedom of motion. If either swivel or door rod is binding, following steps should be accomplished.

A. Remove door rod at trunnion door and strut end.

B. Remove door swivel and strut swivel.

C. Remove rod end terminal from door rod (free strut end).

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

D. Clean all parts with approved solvent clean all grease passages in trunnion door, door rod and main gear strut (total five holes per rod assembly).

E. Inspect all parts for excessive wear or galling; replace defective parts.

F. Remove any paint between main strut swivel mounting lug surface and washer, this surface should be clean and smooth.

G. Assemble strut swivel, torque castle nut to 10-40 inch-pounds on strut, check swivel for complete 360° freedom of motion.

H. Check for minimum clearance of 0.002 inch between washer and strut when nut is torqued and cotter pin installed.

I. Assemble door swivel. Torque nut to 5-85 inch-pounds.

CAUTION: CENTER WASHER (159LM10073) ON SWIVEL BEFORE TORQUING. LATEST SWIVEL ASSEMBLY (159L10060-5) PROVIDES 0.005 INCH GAP MAXIMUM WHEN NUT TORQUED ON AIRCRAFT 24 - 200, 322 AND 323 EXCLUDING 114. EARLIER AIRCRAFT WITH SWIVEL (159LM10103-5) CAN BE SHIMMED UP ONLY IF RESULTANT GAP IS IN EXCESS OF 0.0028 INCH (SEE FIGURE 201)

NOTE: Trunnion door rattle can be minimized on Aircraft 1 - 23 and 114 by replacement of swivel assemblies 159L10060-10R-3 with -5 assemblies.

J. Check swivel for complete 360° freedom of movement.

K. Install door swivel by threading into door fitting.

L. Eliminate all end play between bushing, door and position lubricator by applying a torque of 100-650 inch-pounds on large hex bushing.

M. Check swivel for complete 360° freedom of motion.

N. Check rod end (159LM10028-3) by threading into door rod; there must be no binding of threads for 1 3/32 inch total travel, excessive wear on threads is cause for rejection.

NOTE: Rod ends that measure over 2.265 $\pm$ 0.010 inch from centerline of hole to end should be cut back to this dimension, chamfered 0.031 X 45°, then brush cadmium plated.

O. Ensure grease passages of lubricators on door rod end are free, apply grease to all fittings (five per door rod).

P. Disconnect both trunnion door rods from strut.

Q. Ensure that door safety struts are firmly installed.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

R. Jack aircraft.

S. Retract landing gear with external power source.

T. Attach either door rod to door fitting and main gear strut swivel.

NOTE: Rod end of assembly without nut or key must be attached to strut swivel.

U. Adjust rod assembly so that edge of door is flush with nacelle skin when door is closed.

V. Disconnect rod assembly from strut swivel and check that it is still possible to rotate face end (without locknut) 2 1/2 turns minimum in both directions.

NOTE: Check inspection hole to ensure proper thread engagement of rod end to rod after rotation check.

W. Adjust opposite rod assembly refer to Steps S. through V. above.

X. Extend landing gear and attach both rod assemblies to strut swivels and secure attaching hardware.

Y. Lubricate door rod and swivel fittings.

Z. Lower jacks, remove jacks and pads.

2. Main Gear Door Trunnion / Swivel / Rod End — Inspection (Aircraft 1 - 111 and 114 Having ASC 164 and Aircraft 112 - 200, 322 and 323, Excluding 114)

(See Figure 201)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

NOTE: The door swivel assembly and the trunnion door rods should be inspected for 360° freedom of motion. If either swivel or door rod is binding, accomplish the following:

A. Remove door rod at trunnion door and at strut end.

B. Remove door swivel assembly at door end after removing grease fitting.

C. Remove grease boot from ball rod end.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

D. Clean all parts with an approved solvent. Clean all grease passages in trunnion door terminal rod end and ball socket fitting assembly (total four holes per rod installation).

E. Inspect all parts for signs of excessive wear and galling, replace defective parts.

F. Check end play in swivel (0.005 inch maximum on -5 assembly); also check swivel for full 360° freedom of rotation.

G. Assemble swivel and torque nut to 5-85 inch-pounds.

H. Install swivel in door and torque hex bushing to 100-650 inch-pounds to eliminate end play and to position grease fitting. (Check swivel for freedom of motion.)

I. Install terminal rod end and ball rod end to door rod (check dimensions in Figure 202). Lockwire jam nut on ball rod end only (terminal rod is used for adjustments).

J. Install boot and secure door rod to socket fitting on strut.

K. Tighten ball rod end socket bushing. Then back off until ball nut is loose. Hand tighten bushing until ball joint binds in socket and again back off bushing to closest serration.

L. Insert screw and lockwire (See Figure 202).

M. Apply grease to all grease fittings.

N. Disconnect both trunnion door rods from strut.

O. Ensure that door safety struts are firmly installed.

WARNING: PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

P. Jack aircraft.

Q. Retract landing gear using external power source.

R. Attach one door rod to main gear strut fitting.

NOTE: Ball end of door rod has a preset dimension (1 5/8 inch), however it may be extended to minimum thread engagement if adjustment on opposite side is insufficient.

S. Adjust rod assembly until aft edge of door is flush with nacelle skin when door is closed.

T. Disconnect rod and adjust opposite rod assembly as outlined in Steps R and S above.

U. Extend landing gear and attach both rod end assemblies to strut fittings and secure with attaching hardware. Lubricate all fittings as required.

V. Retract landing gear and check adjustments of doors.

W. Extend landing gear and install ground safety locks.

X. Lower jacks.

3. Main Gear Door Selector Valve — Removal / Installation

A. Removal

(1) Connect an external source of hydraulic pressure to ground test connections in right engine nacelle.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(2) Jack aircraft.

(3) Locate main gear door selector valve in main wheel well upper bulkhead and place a suitable container with a minimum capacity of five gallons.

CAUTION: WHEN DISCONNECTING ROD ASSEMBLY FROM UPPER END OF CRANK, REMOVE BOLT SLOWLY SINCE CRANK IS UNDER A SPRING LOAD.

(4) Disconnect rod assembly from upper end of valve crank.

(5) Disconnect bungee rod from valve crank and remove valve crank from valve shaft.

CAUTION: SLOWLY REMOVE FILLER CAP FROM RESERVOIR TO BLEED ANY RESIDUAL PRESSURE THAT MAY BE TRAPPED IN RESERVOIR.

(6) Disconnect all hydraulic lines and fittings from valve assembly and plug openings.

(7) Remove mounting screws from valve and remove valve from mounting bracket.

B. Installation

NOTE: On Aircraft having ASC 94, ensure that filters are properly installed in valve.

(1) Install valve crank to valve shaft.

(2) Place valve in position on its mounting bracket and install mounting screws. Safety as required.

(3) Ensure main gear uplocks are in open position before proceeding.

(4) Place bungee rod in position and see that bolt slides freely between bungee rod and valve cranks. If not, adjust terminal end of bungee rod until bolt fits freely and install.

CAUTION: VALVE CRANK WILL BE UNDER SPRING TENSION.

(5) Connect upper end of valve crank to rod. Ensure crank moved approximately 22 1/2° from neutral position. If not, adjust rod until full travel is attained and installed.

(6) Remove plugs from openings and install hydraulic lines and fittings to valve assembly.

(7) Remove all ground safety devices and perform a Landing Gear System — Functional Test. (See Section 32-0)

(8) Replace ground safety devices as required.

4. Clamshell Door Manual Control Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Locate valve in nose wheel well right side, aft of nose wheel well junction box.

CAUTION: BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS, SLOWLY REMOVE FILLER CAP FROM RESERVOIR TO BLEED ANY RESIDUAL PRESSURE THAT MAY BE TRAPPED IN RESERVOIR.

(2) Disconnect lines and fittings from valve.

(3) Remove mounting screws that secure valve to valve mounting bracket and remove valve.

B. Installation

(1) Install valve to mounting bracket using mounting screws.

(2) Using all new O-rings, connect lines and fittings to valve.

(3) Remove jury struts from clamshell doors.

NOTE: To perform operational check of valve, pressurize auxiliary hydraulic system to open clamshell doors, and main hydraulic system to close doors. Clamshell door ground servicing valve can be used only to open doors.

(4) Connect 28V dc power source to aircraft. Place EXT PWR switch to ON.

WARNING: ENSURE ALL PERSONNEL AND EQUIPMENT ARE CLEAR OF DOORS.

(5) Pressurize auxiliary hydraulic system. Place clamshell door ground servicing valve to DOORS OPEN position and hold until doors reach full open position.

(6) Turn auxiliary pump off and return clamshell door ground servicing valve to flight position.

(7) Pressurize main hydraulic system. Clamshell doors will close.

(8) Repeat Steps (5) thru (7) above six times or until system is bled.

(9) Check fluid level in hydraulic reservoir and service if required.

(10) Inspect work area for leakage.

5. Nose Gear Drag Brace Fairing — Adjustment

WARNING: BEFORE WORKING IN ANY WHEEL WELL,, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Jack nose gear.

WARNING: SERIOUS INJURY OR LOSS OF LIFE COULD RESULT WHILE WORKING IN THIS AREA WITH HYDRAULIC PRESSURE APPLIED. REMOVE HYDRAULIC POWER.

B. Disconnect nose gear door rods (159L10085) and tie rods clear of gear. Do not alter adjustment of these rods.

C. Disconnect drag brace faring door rods (159L10066-1) at forward end.

D. Loosen jam nut at aft end of rod.

E. Install an external source of hydraulic power to quick disconnect fittings at right engine hydraulic pump.

F. With hydraulic pressure applied, raise nose gear until it engages its uplock. Place fairing door in the closed position (hard hold) and adjust faring door rod by rotating shaft so that bolts may be inserted freely.

G. Lower nose gear, secure all door rods and tighten jam nuts, raise nose gear until it engages its uplock. With all hydraulic pressure off, ensure that the nose wheel well doors have a light clamp-up effect on drag brace fairing door.

H. Lower nose gear to down and locked position.

I. Lower jack.

6. Trunnion Door — Adjustment (Aircraft 1 - 111 and 114 not having ASC 164)

A. Disconnect both trunnion door rods from strut.

B. Ensure door safety struts are firmly installed.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

C. Jack aircraft.

D. Retract landing gear with external power source.

E. Attach either door rod to door fitting and main gear strut swivel.

NOTE: Rod end of assembly without nut or key must be attached to strut swivel.

F. Adjust rod assembly so that edge of door is flush with nacelle skin when door is closed.

G. Disconnect rod assembly from strut swivel and check that it is still possible to rotate face end (without locknut) 2 1/2 turns minimum in both directions.

NOTE: Check inspection hole to ensure proper thread engagement of rod end to rod after rotation check.

H. Adjust opposite rod assembly, refer to Steps D. thru G. above.

Extend landing gear and attach both rod assemblies to strut swivels and secure attaching hardware.

I. Lubricate door rod and swivel fittings.

J. Lower jacks, remove jacks and pads.

7. Trunnion Door — Adjustment (Aircraft 1 - 111 and 114 Having ASC 164 and Aircraft 112 - 200, 322 and 323)

A. Disconnect both trunnion door rods from strut.

B. Ensure door safety struts are firmly installed.

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

C. Jack aircraft.

D. Retract landing gear using external power source.

E. Attach one door rod to main gear strut fitting.

NOTE: Ball end of door rod has a preset dimension (1 5/8 inch), however it may be extended to minimum thread engagement if adjustment on opposite side is insufficient.

F. Adjust rod assembly until aft edge of door is flush with nacelle skin when door is closed.

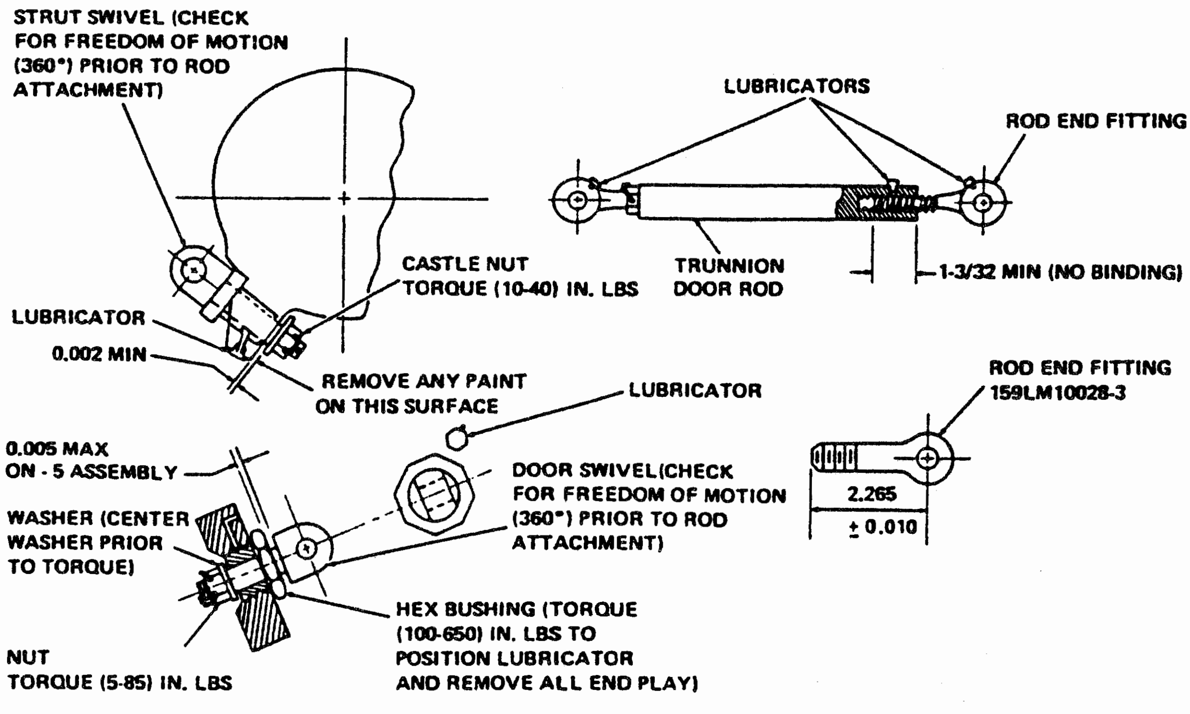
G. Disconnect rod and adjust opposite rod assembly as outlined in Steps E. and F. above.

H. Extend landing gear and attach both rod end assemblies to strut fittings and secure with attaching hardware. Lubricate all fittings as required.

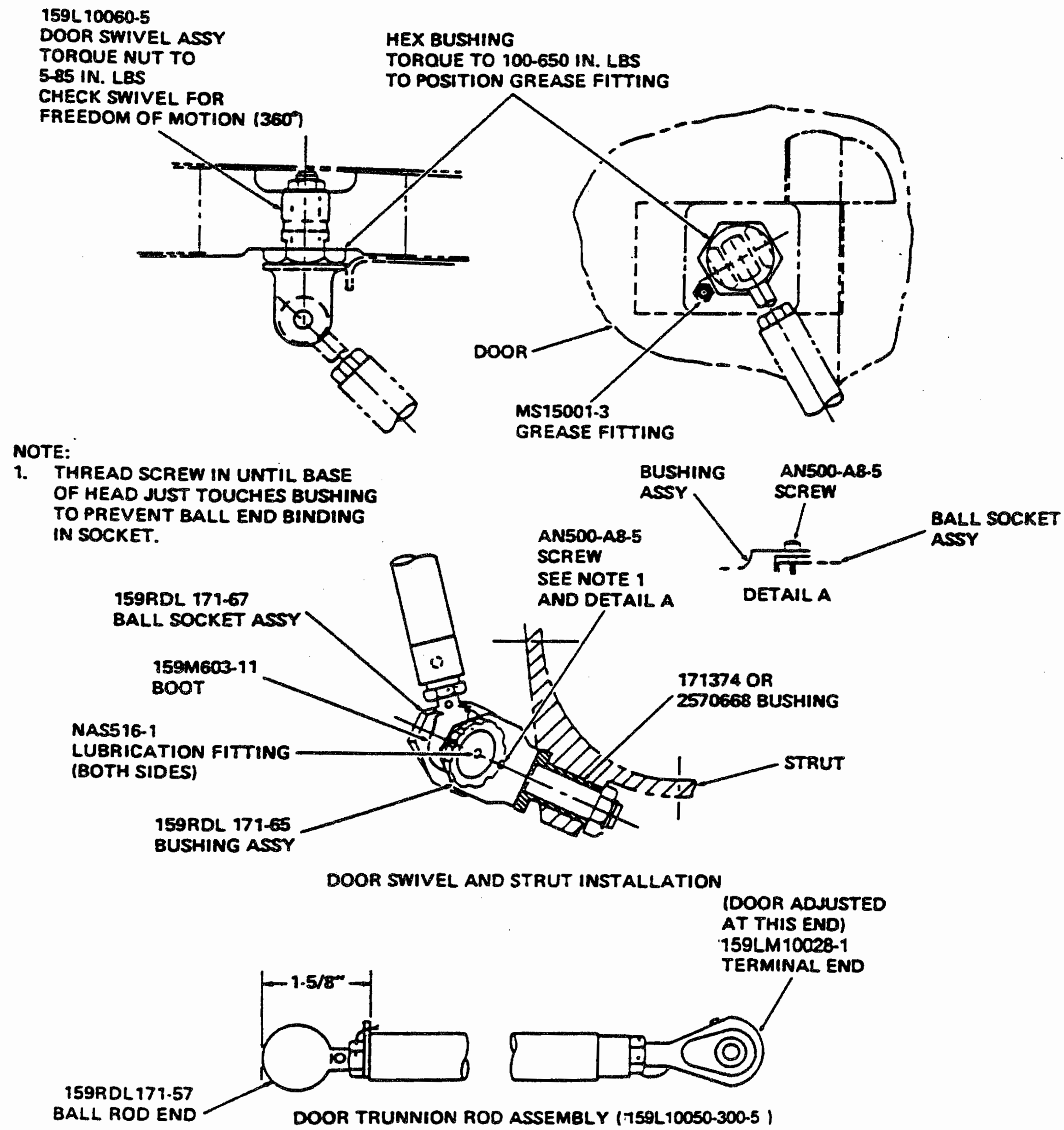
I. Retract landing gear and check adjustments of doors.

J. Extend landing gear and install ground safety locks.

K. Lower jacks, remove jacks and pads.



Trunnion Door Rod End Dimensions and Installation
(Aircraft 1 - 111 and 114 Not Having ASC 164)
Figure 201.



Trunnion Door Rod End Dimensions and Installation
(Aircraft 1 - 111 and 114 Not Having ASC 164, Aircraft 112 - 200, 322 and 323)
Figure 202.

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LANDING GEAR DOOR CONTROL VALVES — DESCRIPTION / OPERATION

1. General

The door control valves are three position, four way poppet selector valves which are provided to control hydraulic pressure to the door actuating cylinder. Line filters are provided at the pressure and return ports of each door control valve to prevent foreign particles from contaminating the poppet seats. During the up cycle of the gear, pressure is ported through the valve to the open port of the door cylinder, thereby opening the doors. As the landing gear retracts the door control valve assumes a neutral position shutting off both pressure and return through the valve. When the gear goes into the uplock the valve is then repositioned by mechanical linkage, reversing the flow to the close side of the door cylinder, thereby closing the doors.

The nose gear door control valve has a total throw of 60° (30° each side of neutral). The main gear door control valve has a total throw of 45° (22 1/2° each side of neutral).

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LANDING GEAR DOOR CONTROL VALVES — MAINTENANCE PRACTICES

1. **Main Gear Door Control Valve — Adjustment**

(See Figure 201)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Each main gear door control valve may be adjusted as follows (gear down and locked):

B. Disconnect rod assembly from crank assembly.

C. Install neutral pin between crank assembly and bracket.

D. Disconnect spring rod assembly from crank assembly. Also disconnect both springs.

E. Position 1/2 inch diameter uplock hook roller to make contact with latch assembly between flat and ramp (Insert B).

F. Adjust stop bolt (Fuselage Station 277) until striker on uplock begins to make contact with roller (Insert A).

G. Adjust and install spring rod assembly to maintain the relationship and lockwire spring rod assembly and stop bolt.

H. Install both springs and remove neutral pin.

I. Install rod assembly.

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

J. Jack aircraft.

K. Pressurize hydraulic system and operate gear. Check for proper sequence of door operation and gear operation.

NOTE: During hydraulic checkout of main gear sequence, the rod assembly may be adjusted to permit maximum valve opening within valve travel limitations.

L. Lower jacks.

2. **Nose Gear Door Control Valve — Adjustment**

(See Figure 202)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

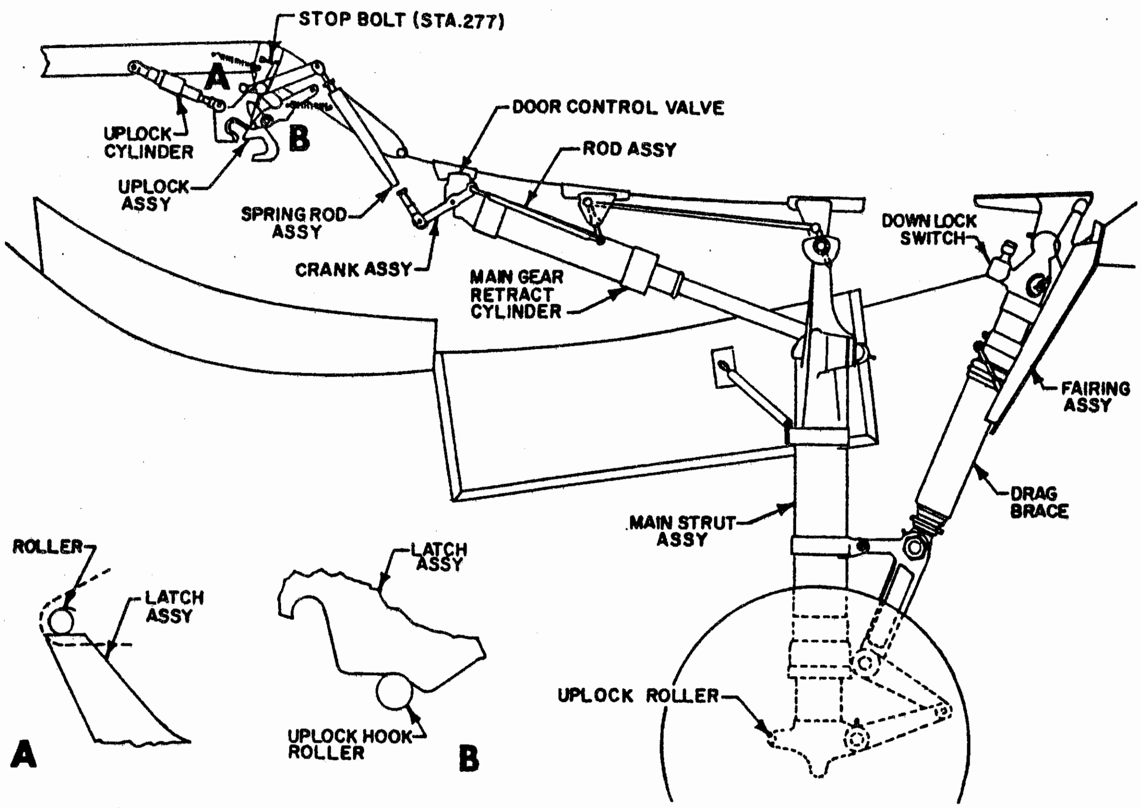
A. Jack aircraft.

B. Using hydraulic hand pump, unlock nose gear. Brace nose gear keeping it free from making contact with arm assembly.

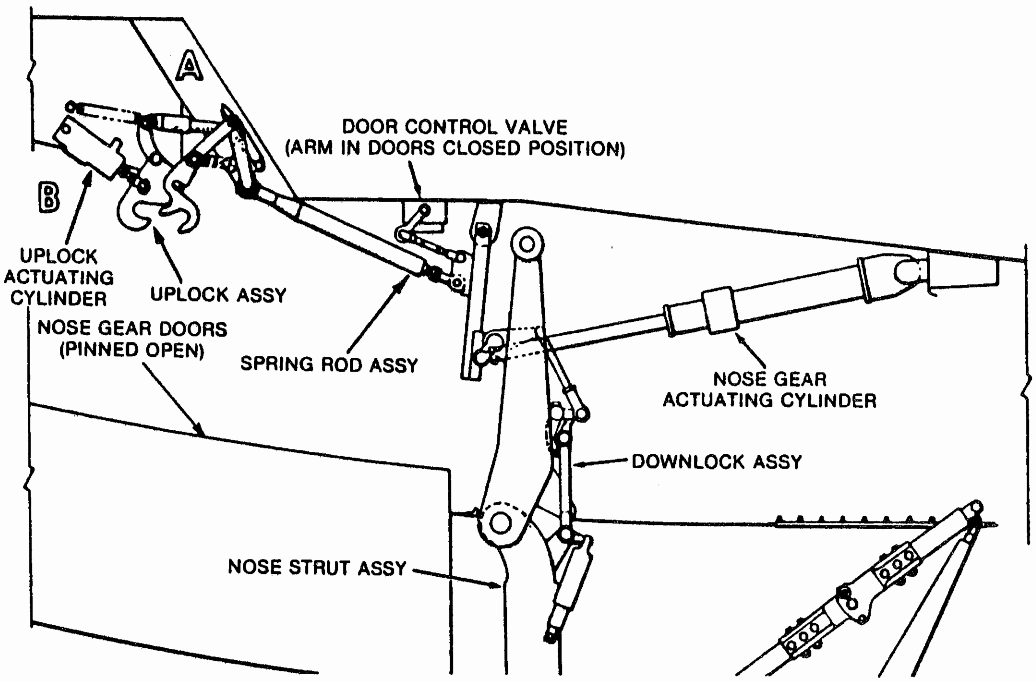
C. With door control valve in neutral position, adjust spring rod assembly to pick up arm assembly and torque shaft assembly (nose gear uplock unlocked).

D. Pressurize hydraulic system and operate gear. Check for proper sequence of door operation and gear operation.

E. Lower jacks.



Main Gear Door Control Valve — Adjustment
Figure 201.



Nose Gear Door Control Valve — Adjustment
Figure 202.

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LANDING GEAR DOOR ACTUATING CYLINDERS — DESCRIPTION / OPERATION

1. Description

(See Section 32-5-0, Figure 1)

A. Main Gear Door Actuating Cylinder

Each set of main landing gear doors is retracted and extended by means of a hydraulic actuating cylinder which is attached to inboard and outboard door fittings. An internal lock, incorporated in the cylinder, holds the piston in the extended position, thus keeping the doors closed. The extended length of the cylinder is 26.2 inches, and the retracted length is 17.375 inches, measured between the center lines of the two cylinder attaching points. The tubing at the head-end port and at the normal pressure port of the shuttle valve end is 3/8 inch, the tubing at the emergency port of the shuttle valve is 1/4 inch. The terminal at the rod end may be adjusted $\pm$ 3/16 inch. One lubricating hole near the rod end permits oiling of the felt wiper. See Lubrication Chart (Chapter 12).

B. Nose Gear Door Actuating Cylinder

The nose gear doors are retracted and extended by means of a hydraulic actuating cylinder which is attached to inboard and outboard door fittings. The extended length of the cylinder is 28.625 inches, and the retracted length is 18 inches, measured between center lines of the two cylinder attaching points. The ports accommodate 1/4 inch tubing. The terminal at the rod end may be adjusted $\pm$ 0.250 inch. One lubricating hole near the rod end permits oiling the felt wiper. See Lubrication Chart (Chapter 12).

2. Operation

(See Landing Gear Hydraulic System, Section 32-5-0)

A. Main Gear Door Actuating Cylinder

During the landing gear up cycle, pressure through the shuttle valve at the open port of the door cylinder unlocks the ball locks, causing the piston to retract and the doors to open. As the opening cycle of the doors is completed, mechanical linkage, attached to the aft end of the inboard door, depresses two timer valves. As the lower timer valve operates to bring the gear up, the roller on the forward portion of the gear strikes the up lock hook. This action releases the hook so that it can engage the roller. The door control valve, which directs pressure to the head end of the cylinder, is repositioned, causing the piston to extend and the main gear doors to close.

During the landing gear down cycle, pressure through the shuttle valve at the open port of the door cylinder unlocks the ball locks, causing the piston to retract and the doors to open. After the gear come down and after pressure, directed to the head end of the cylinder, extends the piston and closes the main gear doors, the door control valve is repositioned. A shuttle valve, incorporated in the main gear door actuating cylinder at the rod end, permits automatic switchover from hydraulic to pneumatic pressure only for the emergency door-opening cycle. The doors remain open after emergency extension of the gear.

WARNING: THE SHUTTLE VALVES MUST BE REPOSITIONED AFTER EMERGENCY EXTENSION OF THE LANDING GEAR. SEE RETURNING LANDING GEAR TO NORMAL OPERATION.

B. Nose Gear Door Actuating Cylinder

During either the nose gear up or the gear down cycle, normal hydraulic pressure enters the open port at the rod end of the door cylinder, retracts the piston and opens the doors.

Whether the nose gear extends or retracts, the nose gear door control valve reverses the flow and pressure of the fluid at the close port, causing the piston to extend and the doors to close. A shuttle valve, at the rod end of the nose gear door actuating cylinder, permits automatic switchover from hydraulic to pneumatic pressure only for the emergency door opening cycle. The doors remain open after emergency extension of the gear.

WARNING: THE SHUTTLE VALVE MUST BE REPOSITIONED AFTER EMERGENCY EXTENSION OF THE LANDING GEAR. SEE RETURNING LANDING GEAR TO NORMAL OPERATION.

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LANDING GEAR DOOR ACTUATING CYLINDERS — MAINTENANCE PRACTICES

1. **Main Gear Door Actuating Cylinder — Removal / Installation**

   A. Removal

   (1) Install ground safety locks in all three gear and jury struts in appropriate main wheel well.

   CAUTION: SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO RELEASE RESIDUAL PRESSURE THAT MAY BE TRAPPED IN RESERVOIR.

   (2) Remove crossover rod from door cranks and retain hardware.

   (3) Disconnect hydraulic lines and fitting from actuator.

   (4) Disconnect door actuator from door cranks and retain hardware.

   B. Installation

   (1) Fully bottom terminal end of actuator and install between door cranks.

   (2) Connect hydraulic hand pump to actuator and close doors gradually. Open main gear doors and adjust cylinder terminal rod end so that both doors are flush with contour of nacelle when fully closed.

   (3) With main gear door fully open, install crossover rod between door cranks, adjusting length so that attaching bolts can be freely inserted.

   (4) Cycle doors fully closed and adjust crossover rod by rotating rod (if necessary) so that both doors mate evenly. Readjust actuating cylinder terminal rod end, if necessary so that both doors when properly mated will be flush with contour of nacelle. When pressure is removed from cylinder, the doors should move slightly toward open position.

   (5) Remove hand pump and connect to aircraft hydraulic system.

   (6) Operate doors 6 times to bleed system.

   (7) Check hydraulic reservoir for capacity. Service as required.

   (8) Remove ground safety devices as required.

2. **Nose Gear Door Actuating Cylinder — Removal / Installation**

   A. Removal

   (1) Install all ground safety locks.

   WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

   (2) Jack aircraft.

   (3) Open clamshell doors using door ground servicing valve.

   (4) Remove both door rod assemblies, tie rod assembly, retaining attaching hardware.

   (5) Disconnect hydraulic and pneumatic lines from door actuator and cap all exposed lines and fittings.

   (6) Disconnect actuator from door cranks and retain attaching hardware. Carefully remove actuator.

   B. Installation

   (1) With door cranks in door closed position, insert neutral pins through neutral pin holes in cracks.

   (2) Adjust length of tie rod assembly until attaching bolts can be freely inserted and install.

   (3) Attach body end of actuator to right door crank.

   (4) Install pressure source to extend actuator until it is fully bottomed and adjust rod end until it aligns with left door crank and attaching bolt can be freely inserted. Install rod end of actuator.

(5) Install door rod to clamshell doors.

(6) Remove neutral pins from door cranks.

(7) With cylinder fully bottomed in closed position, manually hold door closed and adjust door rod so that attaching bolt can be freely inserted through door crank. Remove bolt and repeat procedure for other door.

(8) With door rod assemblies installed, ensure there is no pinch with doors closed and that they are flush with fuselage contour. Readjust as required.

(9) Install nose gear clamshell door jury strut.

(10) Remove external pressure source form actuator and connect aircraft hydraulic and pneumatic lines to actuator.

(11) Connect an external source of pressure to main hydraulic system.

(12) Remove all ground safety locks and door jury struts.

(13) Cycle gear six times to bleed air out of system.

NOTE: If auxiliary pump is to be used, install ground safety locks in main gear.

(14) Replace all ground safety locks and lower aircraft from jacks using all necessary precautions.

3. Main Gear Door Actuating Cylinder — Adjustment

A. Install door cylinder with terminal rod end adjustment fully bottomed.

B. Using hydraulic hand pump, close doors gradually. Open main gear doors and adjust cylinder terminal rod end so that both doors are flush with contour of nacelle when fully closed. Repeat until doors are flush.

C. With main gear doors open install crossover tie rod, adjusting length so that attaching bolts can be freely inserted.

D. Cycle doors fully closed and adjust cross tie rod (by rotating rod), if necessary, so that both doors when properly mated will be flush with contour of nacelle. When pressure is removed from cylinder, the doors should move slightly towards the open position.

4. Nose Gear Door Actuating Cylinder — Adjustment

A. Remove both door rod assemblies, tie rod assembly and door cylinder.

B. With door cranks in door closed position, insert neutral pins through neutral pin holes. Install the rod assembly, adjusting length to permit attaching bolts to be freely inserted.

C. With neutral pins still in place install door cylinder, adjust terminal rod end so that with cylinder bottomed in fully extended position (using hydraulic hand pump), the attaching bolts can be freely inserted.

D. Remove neutral pins. Use hand pump to fully bottom cylinder and readjust terminal rod end, if necessary, so that neutral pins will fit freely.

E. Remove neutral pins and attach door rod at door end. With cylinder fully bottomed (using hydraulic hand pump), manually hold door in the closed position and adjust rod length so that attaching bolt at the crank can be freely inserted. Remove bolt and repeat same procedure for opposite door.

CAUTION: TO PREVENT FAILURE OF DOOR ROD ASSEMBLIES, DO NOT SHORTEN RODS IN EXCESS OF AMOUNT REQUIRED FOR STEP F.

F. With both door rod assemblies installed, adjust rods gradually to remove door pinch, and ensure doors mate and close flush with the fuselage contour.

LANDING GEAR CONTROL SYSTEM — DESCRIPTION / OPERATION

1. Description

The landing gear control system consists of the following: the pilots and the copilots landing gear handles (placarded LANDING GEAR HANDLE) located to the left and the right of the center pedestal console near the trim tab wheels; the nose gear selector valve located at the to p of the nose wheel well; the main gear and speedbrake selector valve located along the center line at the top of the nose wheel well; mechanical linkage; the emergency landing gear extension handle (placarded EMER LDG GEAR) located on the copilots side console; and mechanical linkage to the air release valve. (See Figure 1) The landing gear control handles, used for normal extension or retraction of the landing gear, have two detent positions, UP and DOWN. They are mechanically connected to the nose gear selector valve and the main gear and speed brake selector valve.

2. Operation

A. Normal

With the selection of UP, main hydraulic power system pressure is directed through the selector valves to the up system, and the return flow is routed to the hydraulic reservoir. With the selection of DOWN, main hydraulic power system pressure is directed through the selector valves to the down system and return flow is routed to the hydraulic reservoir.

B. Emergency

The EMER LDG GEAR handle, used for emergency extension of the landing gear, is mechanically connected to both the air release valve and to the emergency extension system dump valve. Air pressure operates another dump valve (the manual reset dump valve), shutting off main system pressure and releasing emergency air pressure into the down system, opening the nose gear and main gear doors, unlocking the nose gear and main gear uplocks, and bringing the nose gear down with air pressure and the main gear down by gravity. Simultaneously, the emergency extension system dump valve opens the up system of the landing gear to return fluid to the hydraulic reservoir.

3. Detail Component Operation

A. Proper Operation of Landing Gear Handle Transfer Switches.

With aircraft on jacks, operate the landing gear handles. Ensure that the transfer switches (LANDING GEAR HANDLE switches No. 1 and No. 3), mounted on the bracket located on the cockpit floor at Fuselage Station 85.375, are being actuated.

B. Proper Operation of Landing Gear Handles in Unison.

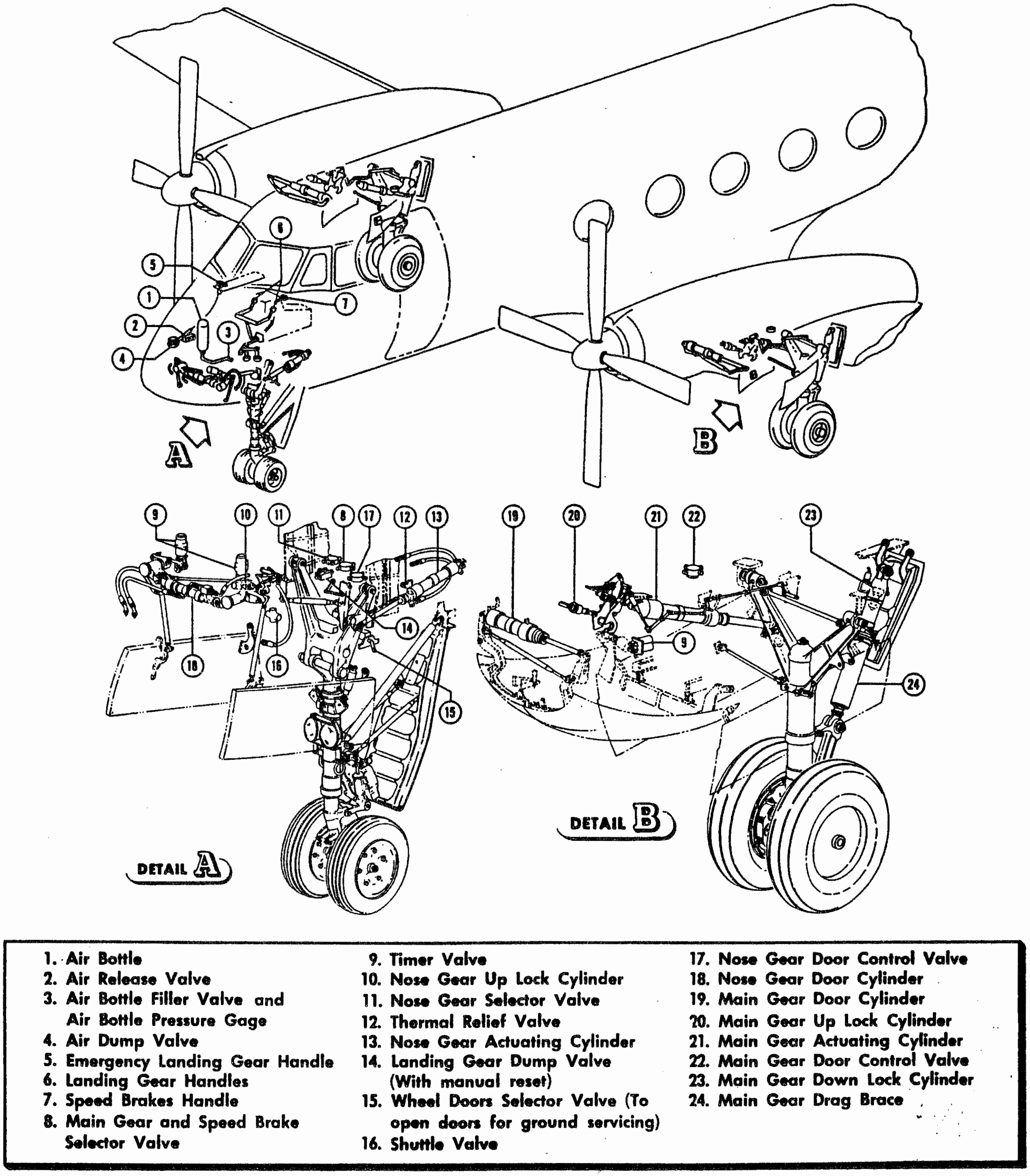
With aircraft on jacks, operate the landing gear handles. They should operate in unison. If one handle sticks in a detent, or otherwise fails to operate in conjunction. with the other handle, adjust the linkage between them.

C. Proper Operation of Ground Solenoid Lock On Copilots Landing Gear Handle.

With aircraft on jacks, operate the pilots or copilots landing gear handle. With no electrical power, the spring should push out the latch located in the ground solenoid lock on the handle preventing movement of the handle to UP. If this does not occur, the spring is defective and the ground solenoid lock should be replaced.

With electrical power on and aircraft on jacks, the latch spring will be overridden by solenoid action, permitting the landing gear handles to be moved freely.

WARNING: KEEP AIRCRAFT ON JACKS WHEN REPLACING SOLENOID.



Landing Gear Control System
Figure 1.

LANDING GEAR HYDRAULIC SYSTEM — DESCRIPTION / OPERATION

1. General

A. Main Gear — Up Cycle

Main hydraulic system pressure enters the check valve, then passes through the manual reset dump valve (See Figure 1). From the dump valve, the flow goes to the nose gear selector valve and the main gear and speedbrake selector valve. With landing gear selected UP, hydraulic pressure is diverted from the P1 port of the main gear and speedbrake selector valve to the C2 port. The flow of fluid is now directed to the unlocking port of the drag brace down lock, to the P port of the up system timer valve (this timer valve is closed at this time, thus no hydraulic pressure can reach the rod end of the main gear cylinder) to the head of the uplock cylinder; and to the P port of the door control valve. The uplock cylinder cannot extend, as a mechanical lock connected to the uplock will hold the uplock hook. The hydraulic pressure at the P port of the door control valve is now directed out the C2 port on the left gear or C1 port on the right gear, and through the shuttle valve at the rod end of the main gear door cylinder, unlocking the ball locks and retracting the cylinder, which opens the main gear doors. The final opening motion of the doors operates a mechanical linkage attached to the aft end of the inboard main gear door, depressing the two timer valves. This directs pressure from the C port of the up system timer valve to the rod end of the main gear cylinder, retracting the piston and bringing the main gear up. When the roller on the forward portion of the main gear strut strikes the uplock latch, it releases the uplock hook, allowing the uplock cylinder piston to extend and engage the roller; and repositions the door control valve, reversing the flow so that the pressure will leave the door control valve at the C1 port on the left gear or C2 port on the right gear. Pressure is directed to the head end of the main gear door cylinder, extending the piston and closing the main gear doors.

B. Main Gear — Down Cycle

With landing gear selected DOWN, the main hydraulic pressure at the P1 port of the main gear and speedbrake selector leaves the C1 port, to be directed to the locking port of the drag brace main gear down lock, the O port of the down system timer valve, the R port of the door control valve, and the head end of the main gear cylinder. Pressure going to the drag brace prepares the drag brace lock to lock the fingers in the drag brace when the main gear is fully extended. Although pressure is at the head end of the main gear cylinder, the cylinder cannot extend, as the uplock cylinder is not pressurized and the gear remains held in the uplocks. The uplock cylinder is pressurized when the down system timer valve is opened by the mechanical connection to the main gear door. Hydraulic pressure at the R port of the door control valve is now directed out the C2 port on left gear or C1 port on right gear. It goes through the shuttle valve at the rod end of the main gear door cylinder, unlocks the ball lock, retracting the piston and opening the main gear doors. The final opening motion of the main gear doors operates a mechanical linkage at the aft end of the inboard door, depressing the two timer valves. Pressure at the down system timer valve is now permitted to leave the C port of the timer valve, and go directly to the uplock cylinder through the shuttle valve, retracting the piston and releasing the roller from the uplock. Pressure at the head end of the main gear cylinder now extends the piston and brings the main gear down. When the main gear strut reaches the full down position, a mechanical linkage attached to the top of the main gear strut repositions the door control valve, directing the pressure to leave the door control valve at the C1 port on the left gear or C2 port on the right gear. Pressure goes to the head end of the main gear door cylinder, extending the piston and closing the main gear doors.

Aircraft 88 - 200, 322 and 323, excluding 114 and Aircraft having ASC 140 have a pressure relief valve connected between the rod end of the actuating cylinder of each main gear and the timer and check valves for that gear. The pressure relief provides a means for relieving any excessive hydraulic pressure resulting from aerodynamic loads during speedbrake extensions at high airspeeds. The relief valve has a cracking pressure of 1750 psi and a reseat pressure of 1525 psi.

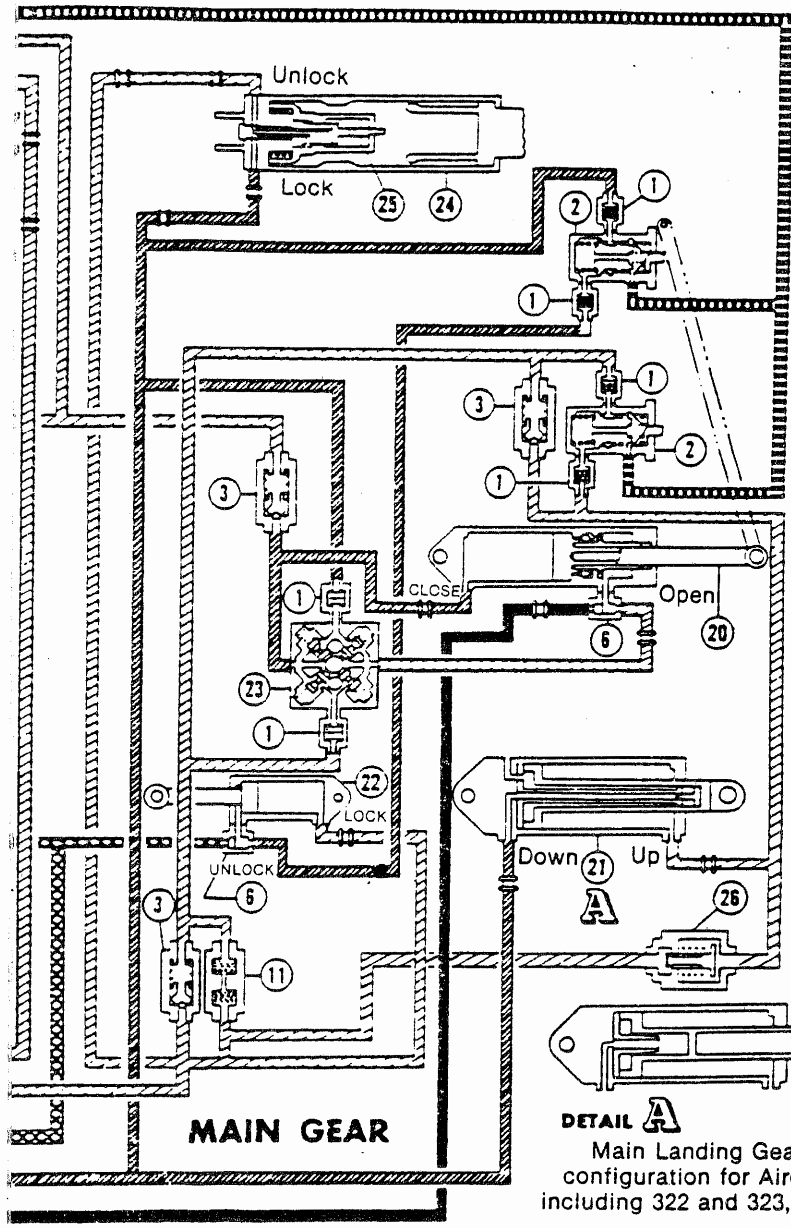
C. Nose Gear — Up Cycle

The main hydraulic system pressure enters the check valve, then passes through the dump valve. From the dump valve, the flow goes to two points: the nose gear selector valve, and the main gear and speed. brake selector valve. With landing gear selected UP, main hydraulic system pressure goes to the PRESS

port of the nose gear selector valve. The UP selection directs hydraulic pressure from the PRESS port of the nose gear selector valve to the CYL 1 port of the valve. From this point, hydraulic pressure is directed through the check valve to the ground servicing valve. With the ground servicing valve in the flight position, the pressure from the nose gear selector valve is blocked. The flow upstream of the check valve goes to three points, through a restrictor to the P port of the nose gear door control valve, to the head end of the uplock cylinder to extend the piston; and to the P port of the up system timer valve. This timer valve is closed at this time; thus, no hydraulic pressure can reach the rod end of the nose gear cylinder. The uplock cylinder is retracted in a position to engage the roller (on the forward part of the nose gear strut) when the nose gear strut retracts. The door control valve is ported at this time to permit hydraulic pressure to pass from the P port to the C1 port, thus hydraulic pressure enters the rod end of the nose gear door cylinder, retracting the piston and opening the doors. The final retracting motion of the door cylinder operates a mechanical linkage to open the two timer valves. The up system timer valve directs hydraulic pressure to the rod end of the nose gear cylinder, retracting the piston and bringing the nose gear strut up. When the roller on the forward end of the strut strikes the latch, it performs two functions: it releases uplock hook, permitting the uplock cylinder piston to extend and engage the roller, and it repositions the nose gear door control valve, reversing the flow so that the pressure leaves the nose gear door control valve at the C2 port. This pressure is directed to the head end of the nose gear door cylinder, extending the piston and closing the nose gear doors. This completes the up cycle of the nose gear.

D. Nose Gear — Down Cycle

With landing gear selected DOWN, main hydraulic system pressure is at the bottom of PRESS port of the nose gear selector valve. The DOWN selection directs hydraulic pressure to the CYL 2 port of the valve. From this point hydraulic pressure is directed to three points, through the shuttle valve leading to the head end of the nose gear cylinder, to the R port of the door control valve, and to the P port of the down system timer valve. The down system timer valve at, this time is closed thus, no hydraulic pressure can pass through the timer valve to reach the shuttle valve end of the uplock cylinder. The uplock cylinder retains the nose gear strut in the up and locked position, even through hydraulic pressure is at the head end of the nose gear cylinder. The door control valve is ported at this time to permit hydraulic pressure to pass from the R port to the C1 port. This directs hydraulic pressure to the rod end of the nose gear door cylinder through the shuttle valve, retracting the piston and opening the nose gear doors. The final retracting motion of the door cylinder operates and opens the two timer valves, permitting hydraulic pressure to pass through the down system timer valve to the rod end of the uplock cylinder, through the shuttle valve. The uplock cylinder retracts and releases the roller on the forward portion of the nose gear strut, and the pressure at the head end of the nose gear cylinder extends the piston and brings the nose gear strut down. The final motion of the nose gear strut operates a mechanical linkage to reverse the selection of the nose gear door control valve. This action directs the hydraulic pressure at the R port of the door control valve through the C2 port, to the head end of the nose gear door cylinder, extending the piston and closing the doors, thus completing the down cycle of the nose gear.



Main Landing Gear Cylinder configuration for Aircraft 25 - 200 including 322 and 323, excluding 114

. Nose Gear Selector Valve
. Restrictor
. Nose Gear Door Control Valve
. Air Release Valve
. Air Bottle
. Air Bottle Pressure Gage
. Air Bottle Filler Valve
. Air Dump Valve
. Landing Gear Dump Valve (With manual reset)

19. Main Gear and Speedbrake Selector Valve
20. Main Gear Door Cylinder
21. Main Gear Actuating Cylinder
22. Main Gear Uplock Cylinder
23. Main Gear Door Control Valve
24. Main Gear Drag Brace
25. Main Gear Downlock Cylinder
*26. Pressure Relief Valve

Landing Gear Hydraulic System — Schematic
Figure 1

LANDING GEAR HYDRAULIC SYSTEM — MAINTENANCE PRACTICES

1. **Landing Gear Thermal Relief Valve — Removal / Installation**

   A. Removal

   WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

   (1) Open clamshell doors and install jury brace on nose gear clamshell doors.

   CAUTION: OPEN FILLER CAP ON HYDRAULIC RESERVOIR TO BLEED RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

   (2) Place a suitable container beneath valve.

   (3) Disconnect hydraulic lines and remove fittings from valve. Discard O-rings and retain fittings.

   (4) Remove valve.

   B. Installation

   (1) Using new O-rings install fittings into valve.

   (2) Place valve in position and connect hydraulic lines to valve. Tighten fittings and lines.

   (3) Check and service reservoir as required.

2. **Landing Gear Hydraulic Dump Valve — Removal / Installation**

   A. Removal

   WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

   (1) Install ground safety locks in gear, and jury struts on clamshell doors.

   WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

   (2) Jack aircraft.

   CAUTION: BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS, SLOWLY REMOVE FILLER CAP FROM RESERVOIR TO BLEED ANY RESIDUAL PRESSURE THAT MAY BE TRAPPED IN RESERVOIR.

   (3) Locate valve in nose wheel well, upper right corner at approximately Fuselage Station 50.

   (4) Disconnect lines and fittings from valve.

   (5) Disconnect valve arm from gang rod.

   (6) Remove valve from mounting bracket.

   B. Installation

   (1) Remove valve arm from old valve and install on new valve shaft in same position.

   (2) Install valve on mounting bracket.

   (3) Connect valve arm to gang rod.

   (4) Connect all lines and fittings to valve.

   (5) Perform a Landing Gear System — Functional Test. (See Section 32-0)

   (6) Lower aircraft.

   (7) Remove ground safety devices as required.

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LANDING GEAR ACTUATING CYLINDERS — DESCRIPTION / OPERATION

1. **General**

   A. Main Gear Actuating Cylinder

   Each main landing gear is retracted and extended by means of a hydraulic actuating cylinder which is attached to the shock strut at the rod end, and to a fitting on the wing structure in the nacelle at the head end. An internal dashpot, incorporated in the cylinder, slows the action during the final phase of the retraction stroke. In addition, a feed tube is installed on the first 25 aircraft, which reduces the area and volume during the piston extension. Extended length of the cylinder is 38.703 inches, measured between, the center lines of the two cylinder attaching points. The retracted length is 24.050 inches. The ports accommodate 1/2 inch flexible tubing. The terminal at the rod end can be adjusted $\pm$ 0.250 inch in length. Two lubricating holes near the rod end permit oiling of the felt wiper. (See Lubrication Chart, Chapter 12). Up pressure to the cylinder forces the piston to retract and permits fluid restricted by the dashpot at the end of the retraction stroke to return to the reservoir from the down port at the head end. Down pressure to the cylinder, entering at the head end, forces the piston to extend and allows fluid to return to the reservoir from the port at the rod end.

   B. Nose Gear Actuating Cylinder

   The nose gear is retracted and extended by means of a hydraulic actuating cylinder which is attached to the shock strut yoke at the rod end and to the fuselage structure in the nose wheel well at the head end. The extended length is 30 inches and the retracted length is 19.125 inches, measured between the centerlines of the two cylinder actuating points. The ports accommodate 3/8 inch flexible tubing. The terminal at the rod end can be adjusted $\pm$ 0.250 inch in length. After installation, adjustment may be necessary to ensure that the cylinder has not bottomed in the gear down and locked or up and locked position. The lubricating holes near the rod end permit oiling of the fed wiper. (See Lubrication Chart, Chapter 12). Up pressure to the cylinder forces the piston to retract, and permits fluid to return to the reservoir from the down port at the head end. Down pressure to the cylinder, entering at the head end, forces the piston to extend and permits fluid to return to the reservoir from the port at the rod end.

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LANDING GEAR ACTUATING CYLINDERS — MAINTENANCE PRACTICES

1. Main Gear Actuator — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Connect external source of hydraulic pressure to ground test connections in right engine nacelle.

CAUTION: WHEN DISCONNECTING HYDRAULIC LINES, DO NOT DAMAGE LINES BY TWISTING.

(3) Deplete hydraulic system pressure and disconnect hydraulic lines from actuator. Cap all exposed fittings and connections.

(4) Disconnect drag brace fairing door.

(5) Remove attaching hardware from upper and lower end of actuator and retain. Remove actuator.

B. Installation

(1) Install upper end of actuator using existing hardware.

(2) Before connecting actuator lower end, connect hand rig to OPEN port of uplock cylinder then pressurize actuator with a hand pump until piston is bottomed in extended position.

(3) With the piston extended and bottomed, measure and record length of chrome showing on piston.

NOTE: Distance from outer cylinder to end of chrome is approximately 16 inches.

(4) Remove pressure from actuator and retract piston approximately 1/8 inch from fully extended position. Connect actuator lower end to gear. Do not tighten nut.

(5) Retract landing gear slowly with a hand pump to up and locked position.

(6) Secure gear in up position with a beam from floor to lower portion of strut.

(7) Disconnect actuator lower end from gear and with hydraulic pressure applied fully bottom piston in retracted position.

(8) Measure the amount of chrome showing on piston (approximately 7/8 inch) in the fully retracted position.

(9) Carefully extend piston from fully retracted position until amount of chrome showing is 1/8 $\pm$ 1/32 inch greater than measurement made in Step above (fully retracted). Adjusted rod end until it is aligned with gear adapter.

(10) Connect and secure actuator lower end to gear. If nut is 5/8 inch, torque to 270-300 inch-pounds and if nut is 3/4 inch, torque to 480-600 inch-pounds.

(11) Open uplock with hand pump.

(12) Remove support beam and extend gear.

(13) With gear in down and locked position, ensure that amount of chrome showing on piston is less than amount recorded in Step (3) above (piston fully extended).

(14) Connect landing gear actuator and uplock cylinder.

(15) Apply external hydraulic pressure.

(16) Remove ground safety locks.

(17) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(18) Install ground safety locks and lower jacks.

2. Main Gear Actuator — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Jack aircraft.

B. Deplete hydraulic system pressure.

C. Remove retract cylinder terminal end fitting from retract cylinder mounting adapter on strut.

D. Count threads showing from piston on terminal end fitting so that it may be replaced in same relative position, then remove end fitting from piston.

E. Inspect terminal end fitting threads, piston threads and interior of piston for rust and corrosion damage.

NOTE: Terminal end fitting will corrode before piston starts to corrode.

F. If rust or corrosion is not evident (discoloration is acceptable as long as there is no scaling or pitting), clean inspected areas and coat with corrosion preventative compound.

NOTE: Replace terminal end fitting if severely corroded. Coat new fitting with corrosion preventative compound.

Actuating cylinders 159H10026-5 and 159H10026-9 use terminal end fitting 159HM10157-1. Actuating cylinder 159H10026-7 uses terminal end fitting 159HM10153-1.

G. If piston is severely corroded, replace actuating cylinder assembly.

H. Install terminal end fitting after coating threaded portion outside and inside with corrosion preventative compound. Ensure that same number of threads shown as when removed in Step D.

NOTE: If a new actuator assembly is to be installed, remove end fitting and coat with corrosion preventative compound.

I. Install actuator end fitting to retract cylinder mounting adapter on strut.

J. Remove ground safety devices.

K. Using external source of hydraulic pressure, cycle landing gear and check for proper operation.

L. Lower jacks.

3. Nose Gear Actuating Cylinder — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND BEFORE LOOSENING HYDRAULIC FITTINGS OR LINES, SLOWLY REMOVE HYDRAULIC RESERVOIR FILLER CAP TO RELIEVE ANY RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

WHEN DISCONNECTING HYDRAULIC LINES, DO NOT DAMAGE LINES BY TWISTING.

(2) Disconnect hydraulic lines from actuator and cap all exposed fittings and connections.

(3) Remove hardware from lower end of actuator.

(4) Remove hardware from upper end of actuator and carefully remove actuator.

B. Installation

(1) Install upper end of actuator.

(2) Adjust rod end of actuator before installation so that actuator will not bottom in either gear up or gear down position.

(3) Install rod end of actuator.

(4) Connect hydraulic lines.

(5) Check hydraulic reservoir for fluid level and service.

(6) Remove ground safety devices.

(7) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(8) Install ground safety devices.

(9) Lower jacks.

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LANDING GEAR UPLOCK CYLINDERS — DESCRIPTION / OPERATION

1. Description

(See Section 32-5-0, Figure 1)

The uplock cylinders actuate the uplock hooks which retain main and nose gears in up and locked position. Conventional poppet type shuttle valves are incorporated which permit automatic switchover from hydraulic to pneumatic landing gear extension. Uplock cylinders are attached at rod end to the lock; at head end to the aircraft structure. Extended length of the nose gear uplock cylinder is 7 inches, and retracted length is 6 inches, measured between the center lines of the cylinder attaching points. A $\pm$ 0.125 inch adjustment in length can be made at the rod end of the nose gear uplock cylinder. Extended length of each main gear uplock cylinder is 9.562 inches, and the retracted length is 7.812 inches. A $\pm$ 0.187 inch adjustment in length can be made at the rod end of each main gear uplock cylinder, See Maintenance Practices this section for installation adjustments. A lubrication hole is provided in each of the cylinders to permit oiling of the felt wiper. See Lubrication Chart, Chapter 12-0. .

2. Operation

Refer to Landing Gear Hydraulic System, Section 32-5-0.

A. Main Gear Uplock Cylinders.

Up pressure enters the lock port at head end of main gear uplock cylinder. The piston cannot extend, since a mechanical lock connected to uplock hook holds the uplock hook against hydraulic pressure. As gear retracts, the roller on the forward portion of strut strikes the latch. This action permits the uplock piston to extend, locking the gear up.

Normal down pressure enters the unlock port at the rod end of each main gear uplock cylinder, causing piston to retract and releasing the roller from the uplock hook, thus permitting strut to extend.

Emergency air pressure for extension of main gear enters the unlock port at the rod end of main gear uplock cylinder, after shifting the poppet in the shuttle valve. Thereafter, action in uplock cylinder is normal.

WARNING: SHUTTLE VALVES MUST BE REPOSITIONED AFTER EMERGENCY EXTENSION OF LANDING GEAR. REFER TO RETURNING LANDING GEAR TO NORMAL OPERATION, NORMAL / EMERGENCY LANDING GEAR - OPERATIONAL TEST. (SEE SECTION 32-0)

B. Nose Gear Uplock Cylinder

Up pressure enters lock port at head end of nose gear uplock cylinder. Uplock hook is in a position to receive the roller on the forward portion of nose gear strut from previous extension cycle. When roller engages uplock, the uplock cylinder extends, locking gear in the up position.

Normal down pressure enters unlock port at rod end of nose gear uplock cylinder, causing piston to retract and releasing the roller on the forward portion of nose gear strut from engaging hook in uplock assembly, thus permitting strut to extend.

Emergency air pressure for extension of nose gear enters the unlock port at the rod end of nose gear uplock cylinder after shifting the poppet in the shuttle valve. Thereafter, action in uplock cylinder is normal.

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LANDING GEAR UPLOCK CYLINDERS — MAINTENANCE PRACTICES

1. Main Gear Uplock Actuating Cylinder — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Deplete hydraulic system pressure.

(3) Disconnect hydraulic lines and pneumatic line from uplock actuator and cap all exposed lines and fittings.

(4) Remove and discard attaching hardware and carefully remove actuator.

B. Installation

CAUTION: ENSURE THAT ONLY PROPER HARDWARE IS USED ON INSTALLATION OF UPLOCK ACTUATING CYLINDER AT THE THREE ATTACHING BOLTS.

(1) Install cylinder end of actuator using new replacement hardware.

(2) Connect a source of hydraulic pressure to unlock port of actuator and slowly pressurize until actuator bottoms.

(3) Adjust rod end of actuator under pressure to obtain a 0.015 inch to 0.031 inch clearance between latch assembly slots and 1/2 inch rollers on hook assembly. See Figure 201. Connect rod end of actuator to hook assembly using new replacement hardware.

(4) Disconnect external source of pressure from actuator, connect all hydraulic lines and air line to actuator.

(5) Remove ground safety devices.

NOTE: Auxiliary pump may be used. Leave ground safety locks in opposite main gear and in nose gear and operate affected gear only.

(6) Cycle gear six times to bleed air from system and check for leaks.

(7) Replace ground safety devices and lower jacks.

2. Nose Gear Uplock Actuating Cylinder — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Deplete hydraulic system pressure.

(3) Disconnect hydraulic lines and pneumatic line from uplock actuator and cap all exposed lines and fittings.

(4) Remove and discard attaching hardware and carefully remove actuator.

B. Installation

(1) Install actuator body using new replacement hardware.

(2) Connect a source of hydraulic pressure to unlock port of actuator and slowly pressurize until actuator bottoms. Adjust rod end of actuator under pressure to obtain a 0.020 inch to 0.030 inch clearance between hook, rollers and associated slots in latch assembly. See Figure 202. Connect rod end of actuator to hook assembly using new replacement hardware.

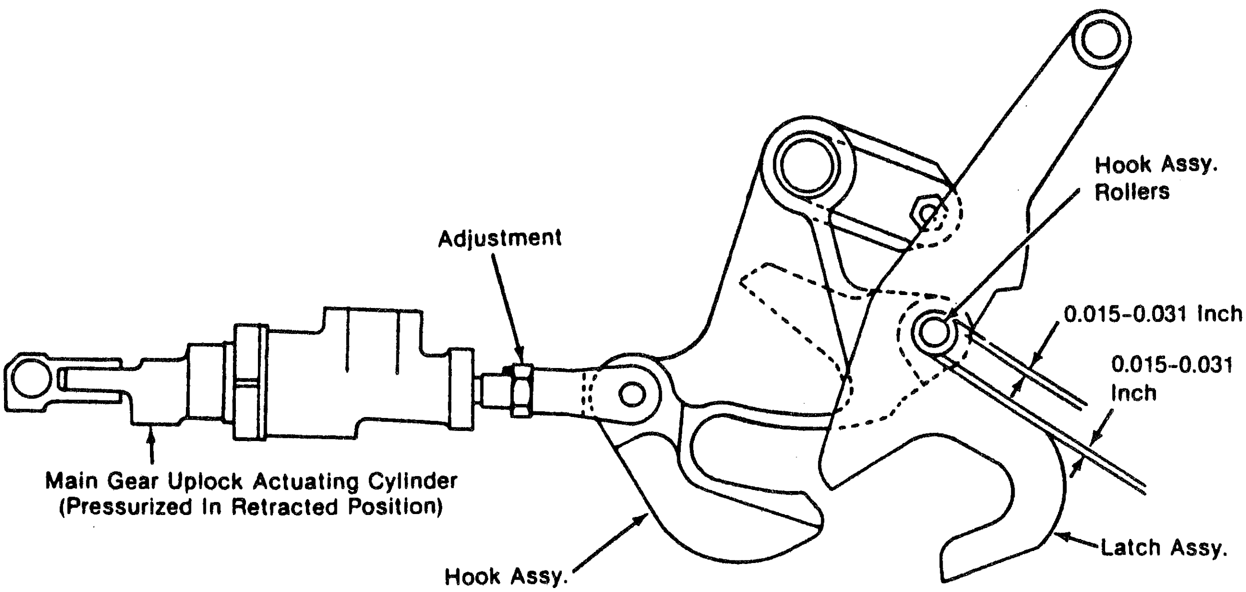
(3) Disconnect external source of pressure from actuator connect all hydraulic lines and air line to actuator.

(4) Remove ground safety devices.

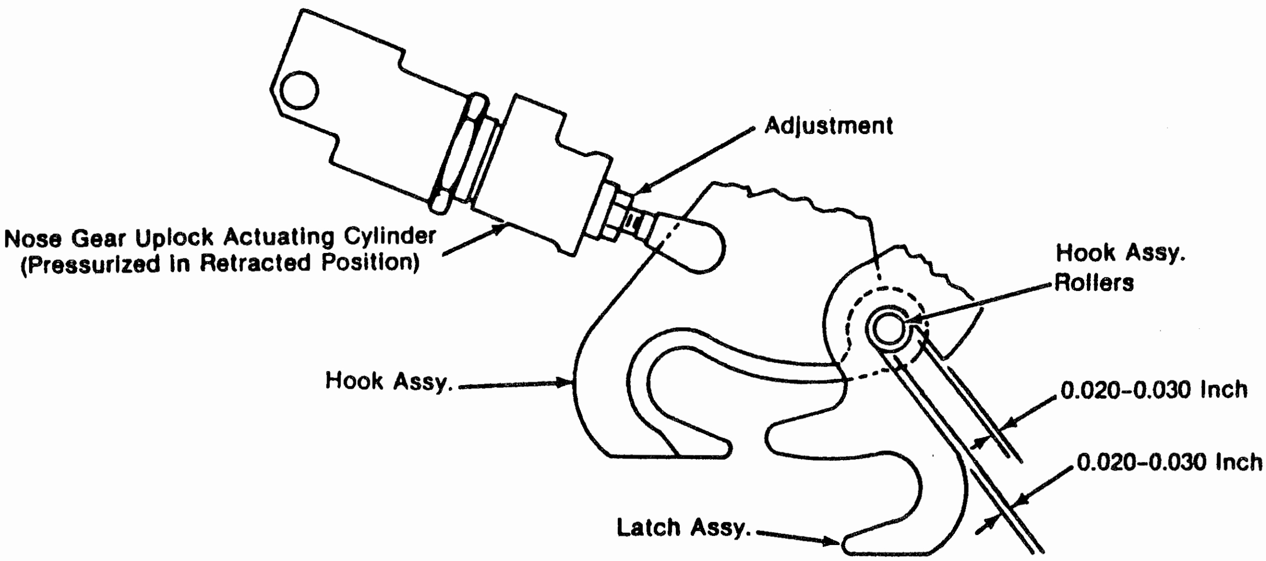
NOTE: Auxiliary pump may be used. Leave ground safety locks on both main gear and operate nose gear only.

(5) Cycle gear six times to bleed air from system and check for leaks.

(6) Replace ground safety devices and lower jacks.



Main Gear Uplock Actuating Cylinder Installation / Adjustment
Figure 201.



Nose Gear Uplock Actuating Cylinder Installation / Adjustment
Figure 202.

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LANDING GEAR CHECK VALVES, FILTERS, SHUTTLE VALVES AND RESTRICTORS — DESCRIPTION / OPERATION

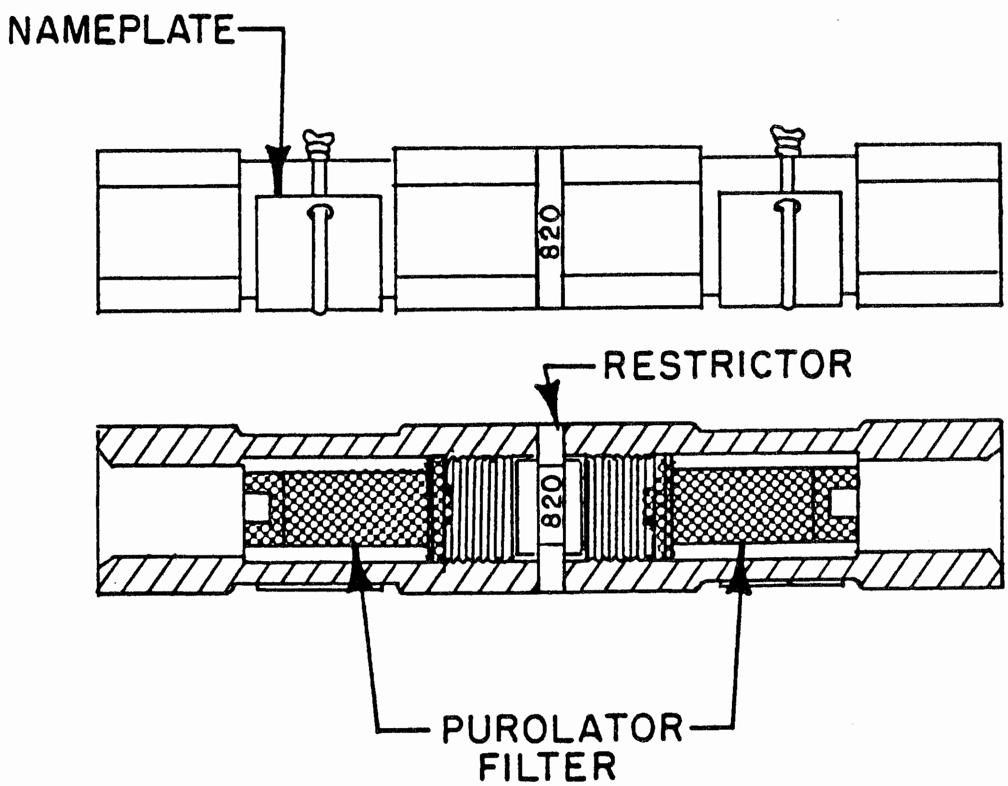
1. **Description**

All of the check valves in the landing gear system are similar in construction and operation to those described in Chapter 29.

The filters in the landing gear system are similar in construction, operation and maintenance to those described in Chapter 29.

The shuttle valves in the landing gear system are described in Landing Gear Hydraulic System, see Section 32-5-0.

The restrictors installed in the landing gear system provide the proper operating speed of the units in the system. Two filters incorporated in each restrictor should be removed and cleaned as necessary, or at overhaul periods. (See Figure 1)



Restrictor Assembly — Protected Type
Figure 1.

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LANDING GEAR CHECK VALVES, FILTERS, SHUTTLE VALVES AND RESTRICTORS — MAINTENANCE PRACTICES

1. Main / Nose Gear Door Shuttle Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Gain access to shuttle valve mounted on door cylinder and place a suitable container with minimum capacity of five gallons in area of valve.

(2) Pull DOOR CONT circuit breaker and placard.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO BLEED OFF ANY RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

(3) Disconnect hydraulic lines from shuttle valve.

(4) Remove Banjo fittings and discard O-rings.

(5) Remove two mount bolts from shuttle valve and remove valve from aircraft.

B. Installation

(1) Place valve in position and install Banjo fittings with new O-rings. Do not tighten until after mount bolts have been installed. Then lockwire.

(2) Connect hydraulic and pneumatic lines to valve.

(3) Depress DOOR CONT circuit breaker and remove placard.

(4) Remove all three jury struts from clamshell doors.

(5) Close doors using hydraulic pressure.

NOTE: Doors can be closed by rotating either propeller. On aircraft having ASC 109, the auxiliary pump may be used to pressurize the main hydraulic system.

(6) Open doors by pressurizing auxiliary hydraulic system and place clamshell door ground servicing valve in the DOORS OPEN position.

(7) Repeat Steps(5) and (6) five or six cycles to bleed air out of system. Check for leaks at all fittings.

(8) Check fluid level in hydraulic reservoir and service.

2. Nose Gear Actuating Cylinder Shuttle Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Connect external source of hydraulic power to ground test connections in right engine nacelle.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS, SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO RELIEVE RESIDUAL PRESSURE THAT MAY BE TRAPPED IN RESERVOIR.

(3) Locate valve in nose wheel well on upper bulkhead in vicinity of nose gear actuator and place clean, metal container under valve.

WARNING: PLACARD EMERGENCY LANDING GEAR HANDLE TO PRECLUDE OPERATION WHILE WORK IS BEING PERFORMED.

(4) Disconnect all lines and fittings from shuttle valve and plug openings.

(5) Remove mount bolts and remove valve from aircraft.

B. Installation

(1) Place valve in position and install mount bolts.

(2) Connect lines and fittings to valve.

(3) Remove all landing gear safety devices.

(4) Remove safety placard from emergency landing gear handle.

(5) Perform Emergency Operational Test part of Landing Gear System — Functional Test. (See Section 32-0) and check for leaks at air line.

(6) Lockwire emergency landing gear handle to stowed position with soft breakaway wire.

(7) Service pneumatic bottle to 1950 $\pm$ 50 psi with nitrogen or dry compressed air.

(8) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(9) Install landing gear safety devices.

(10) Check fluid level in hydraulic reservoir and service.

(11) Remove external hydraulic pressure source from ground test connections in right nacelle.

(12) Lower and remove jacks.

3. Main Gear Upline Restrictor — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Gain access to main wheel well and place container below area of restrictor.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS SLOWLY REMOVE HYDRAULIC RESERVOIR FILLER CAP TO RELIEVE ANY RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

(3) Disconnect lines and fittings necessary to remove restrictor from aircraft.

B. Installation

(1) Install restrictor in main landing gear system upline and torque fittings 50-70 inch-pounds. Ensure not to overtorque fittings.

(2) Remove ground safety devices from gear.

(3) Pressurize main hydraulic system using external source of pressure.

NOTE: If portable hydraulic unit is not available, auxiliary pump may be used as an alternate method of pressurizing main hydraulic system.

(4) Cycle gear five to six times to bleed air from system and check for leaks.

(5) Check fluid level in hydraulic reservoir and service.

(6) Lower jacks.

4. Main Gear Upline Restrictor Filter Element — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove restrictor.

B. Remove filter element from restrictor.

C. Inspect filter.

D. Clean filter or obtain serviceable filter.

E. Install filter using new O-rings.

F. Install restrictor.

5. Nose Gear Upline Restrictor — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Provide a container suitable for hydraulic fluid.

CAUTION: BEFORE LOOSENING HYDRAULIC FITTINGS OR LINES, REMOVE RESERVOIR FILLER CAP SLOWLY TO BLEED ANY RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

(3) Remove filler cap from hydraulic reservoir.

(4) Disconnect lines and fittings, remove restrictor from aircraft.

B. Installation

CAUTION: DO NOT OVERTORQUE FITTINGS.

(1) Install restrictor and torque fittings 50-70 inch-pounds.

(2) Remove ground safety locks from gear. Remove jury braces from doors.

(3) Pressurize main hydraulic system.

(4) Cycle gear six times to bleed air from system and check for leaks.

(5) Check hydraulic system reservoir and service as required.

(6) Lower aircraft and remove jacks.

6. Nose Gear Upline Restrictor Filter Element — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove restrictor. (See this section)

B. Remove filter element from restrictor.

C. Inspect filter.

D. Clean filter or obtain serviceable filter.

E. Install filter using new O-rings.

F. Install restrictor. (See this section)

7. Main / Nose Gear Uplock Shuttle Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Connect a source of external hydraulic pressure to ground test connections in right engine nacelle.

CAUTION: DEPLETE SYSTEM PRESSURE AND SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO RELIEVE ANY RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

(3) Gain access to shuttle valve.

WARNING: PLACARD EMERGENCY LANDING GEAR HANDLE TO PREVENT OPERATION WHILE WORK IS BEING PERFORMED.

NOTE: Each shuttle valve is located on its respective uplock actuator.

(4) Disconnect all lines and fittings from valve and plug openings.

(5) Remove mount bolts and remove valve from aircraft.

B. Installation

(1) Place valve in position and install Banjo fitting using new O-rings do not tighten until after mount bolts have been installed, then safety wire.

(2) Connect hydraulic pneumatic lines and fittings to valve.

(3) Remove ground safety devices.

(4) Remove safety placard from emergency landing gear handle.

(5) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(6) Install ground safety devices as required.

(7) Check for leakage at all connections.

(8) Safety wire emergency landing gear handle to stowed position using soft, breakaway wire.

(9) Service pneumatic bottle to 1950 $\pm$ 50 psi with nitrogen or dry compressed air.

(10) Check fluid level in hydraulic reservoir and service. Remove external hydraulic pressure source.

(11) Lower jacks and move from aircraft.

LANDING GEAR MANUAL RESET DUMP VALVE — DESCRIPTION / OPERATION

1. **Description**

This is a pressure operated dump valve which is reset manually. The valve blocks the pressure line to the landing gear and speedbrake selector valve and the nose gear selector valve. It relieves all pressure in the landing gear system lines, and thereby prevents pressure from interfering with the extension of the landing gear during emergency operation. The valve is located in the nose wheel well directly above the nose wheel well junction box on the right beam.

On Aircraft 87 - 200, 322 and 323 and Aircraft 1 - 86 and 114 having ASC 116, a manual control has been provided on the copilots side of the center pedestal. This control provides the copilot with the means to reset the dump valve for normal operation of the landing gear after emergency extension of the gear has been employed.

2. **Operation**

Refer to Landing Gear Emergency Extension Air System, Section 32-7-0.

Emergency pressure from the air bottle (actuated by the EMER LDG GEAR handle) enters the dump valve at the port marked PILOT, actuates a piston which blocks the main (normal) pressure to both main and nose gear selector valves at the port marked P, and opens the selector valves at the port marked CYL, through the port marked R for return to the reservoir.

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LANDING GEAR MANUAL RESET DUMP VALVE — MAINTENANCE PRACTICES

1. **Landing Gear Manual Reset Dump Valve — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS, SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO RELIEVE ANY RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

(2) Locate manual reset dump valve in nose wheel well. Valve is directly above wheel well junction box.

(3) Disconnect all hydraulic and pneumatic lines from valve.

NOTE: On Aircraft 1 - 86 and 114 having ASC 116 and Aircraft 87 - 200, 322 and 323, a manual control has been installed on the copilot side of the center pedestal.

(4) Remove mount bolts securing valve to bulkhead, disconnect mechanical reset linkage. Remove valve from aircraft.

B. Installation

(1) Install dump valve and reset linkage using three mount bolts.

(2) Measure distance between reset button and reset crank. Distance should be 5/16 inch. If it is 5/16 inch, go to Step (4). If measure is not 5/16 of an inch, continue to next step.

NOTE: When measuring, button should be in reset position.

(3) Compress spring on telescopic unit and adjust end fitting so that the bellcrank assembly is 5/16 of an inch from the dump valve reset button in the reset (normal) position. Check that flex cable is visible through inspection hole and tighten plug in end fitting. (See Figure 201)

(4) Connect all hydraulic and pneumatic lines.

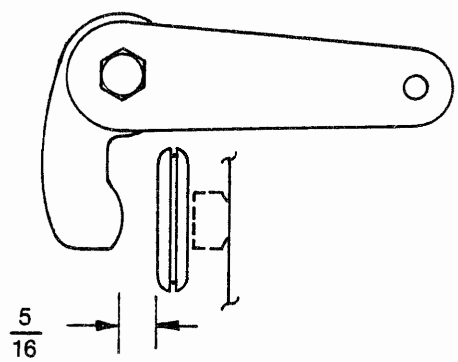
WARNING: DO NOT PULL EMERGENCY LANDING GEAR HANDLE BY MISTAKE DURING NEXT STEP.

(5) Pull manual reset control handle on center console and note that reset crank is touching reset button on valve.

(6) Perform a Landing Gear System — Functional Test. (See Section 32-0)

(7) Safety wire manual reset control handle with breakaway wire.

(8) Lower jacks and remove from aircraft.



View Looking Outboard RH Side

Landing Gear Dump Valve Actuating Cable
Figure 201.

LANDING GEAR TIMER VALVE — DESCRIPTION / OPERATION

1. General

Spring-loaded, mechanically-actuated, poppet-type timer valves are installed in the system to control hydraulic pressure which actuates the units of the landing gear system in their proper sequence. When the valve plunger is pushed in by the opening of the gear doors, the plunger unseats the poppet, which permits fluid to flow. When the plunger is extended, the spring seats the poppet and blocks fluid flow. A valve port opens to return to prevent a build-up of pressure, in the event of a leak at the poppet seat on the cylinder side of the valve. This prevents inadvertent operation of the cylinders. Filters in the lines to the pressure and cylinder ports of the valve require normal maintenance.

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LANDING GEAR TIMER VALVE — MAINTENANCE PRACTICES

1. Main / Nose Gear Timer Valve — Functional Test

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

NOTE: This procedure does not apply to aircraft that have had timer valves (159H10035-3) replaced by GII type valves (1159H20265-3).

A. Jack aircraft.

B. At nose gear doors, disconnect vertical door rods at upper ends.

C. At both main wheel doors, disconnect bungee assemblies at rod terminal end. Retain attaching hardware.

D. Disconnect and cap hydraulic line or fitting at port marked R on three down timer valves (one in each wheel well).

CAUTION: THE FOLLOWING PROCEDURE IS TO BE PERFORMED WITH ONE VALVE AT A TIME. TWO OTHER VALVES IN DOWN SYSTEM MUST HAVE R PORTS PLUGGED WHILE PERFORMING THIS CHECK.

NOTE: Left main wheel well upper valve, right main wheel well lower valve and right valve in nose wheel well are down timer valves.

E. Attach a length of flex hose to port R. Place its opposite end above a metal five gallon container.

F. Attach an external source of hydraulic pressure to ground test connection in right engine nacelle.

NOTE: Aircraft auxiliary hydraulic pump may be used as an alternate source by connecting auxiliary to main hydraulic system.

G. Pressurize main hydraulic system to 1500 psi.

H. With plunger of timer valve fully extended, observe fluid flow, if any, at open end of flex hose. If leakage rate can be measured in drops per minute, valve may be considered acceptable. If leakage is a steady stream, flow rate shall be measured. Allow flow for a one minute period, then direct flow into a one quart container for an accurate one minute period. If flow rate is greater than 60cc per minute (1/8 pint per minute) leakage shall be cause for replacement. Reduce system pressure to zero.

I. Repeat Step H. at each of two remaining timer valves in down system.

J. In nose wheel well, disconnect hydraulic line 159H10002-682LGU-4 at nose gear door control valve and plug line. With aircraft on jacks, place landing gear control handle in UP position.

NOTE: Left main wheel well lower valve, right main wheel well upper valve and nose wheel well valve are up timer valves.

K. Perform Steps D. through H. on timer valve in up system.

NOTE: Upon completion of leakage test, manually depress external valve plunger 3/16 inch to 1/4 inch several times for assurance of valve actuation. Remove and replace valves that leak beyond serviceable limits, or if valve plunger does not return to extended position.

L. Remove flex line used during test.

M. Connect all hydraulic lines and fittings to return port of each timer valve.

N. Connect hydraulic line 159H10002-682LGU-4 to door control valve.

O. Connect bungees to both main landing gear doors.

P. Connect nose gear door rods to their original positions.

Q. Perform a Landing Gear System — Functional Test. (See Section 32-0)

R. Check fluid level in hydraulic reservoir and service.

S. Remove external hydraulic pressure source.

T. Lower jacks and remove ground safety devices as required.

2. Main Gear Up / Down Timer Valve — Removal / Installation

A. Removal

WARNING: PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Jack aircraft.

(2) Connect a source of external hydraulic pressure to ground test connections in right engine nacelle.

CAUTION: DEPLETE SYSTEM PRESSURE AND BEFORE LOOSENING ANY LINES OR FITTINGS, SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO BLEED ANY RESIDUAL PRESSURE TRAPPED IN RESERVOIR.

(3) Disconnect bungee assembly at door end and retain hardware.

NOTE: Left main wheel well upper valve and right main wheel well lower valve are down timers. Opposite valves will be up timers.

(4) Locate valve to be removed and disconnect all hydraulic lines and fittings and discard O-rings. Cap open ports and lines.

(5) Remove mount bolts from valve and remove valve from aircraft.

B. Installation

NOTE: The GII type timer valve (1159H20265-3) is recommended as a replacement valve for timer valves installed in the GI (159H10035-3).

When installing timer valve (1159H20265-3) it is necessary to enlarge hole in support angle from 23/64 inch to 13/32 inch to provide sufficient clearance for timer valve plunger.

(1) Install valve on mounting bracket using mount bolts.

(2) Install and adjust bungee to depress plunger on mounting bracket with door in full open position.

(3) Connect hydraulic lines and fittings to valve using new O-rings.

(4) Perform Normal and Emergency part of Landing Gear System — Functional Test. (See Section 32-0) and check for leaks.

(5) Disconnect external hydraulic unit. Lower jacks and remove ground safety devices.

3. Main Gear Timer Valve Bungee Linkage — Inspection

(See Figure 201)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove hardware attaching link (2) to the bungee (1) and to the bellcrank assembly (3).

B. Inspect all mating surfaces for excessive wear and bolt holes for elongation.

C. Remove bolts from actuator assemblies (4) and remove assemblies.

D. Remove plunger and spring (5 and 6) and inspect for corrosion or contamination.

E. Clean plunger, spring and housing assemblies. Apply a light coat of general purpose grease MIL-G-23827.

F. Install actuator assemblies.

G. Before installing attaching hardware, apply a coating of grease to bolts and mating surfaces of bungee, link and bellcrank.

CAUTION: ENSURE LINKAGE NUTS ARE NO TIGHTER THAN FINGER TIGHT AND BOLTS ARE FREE FROM BINDING.

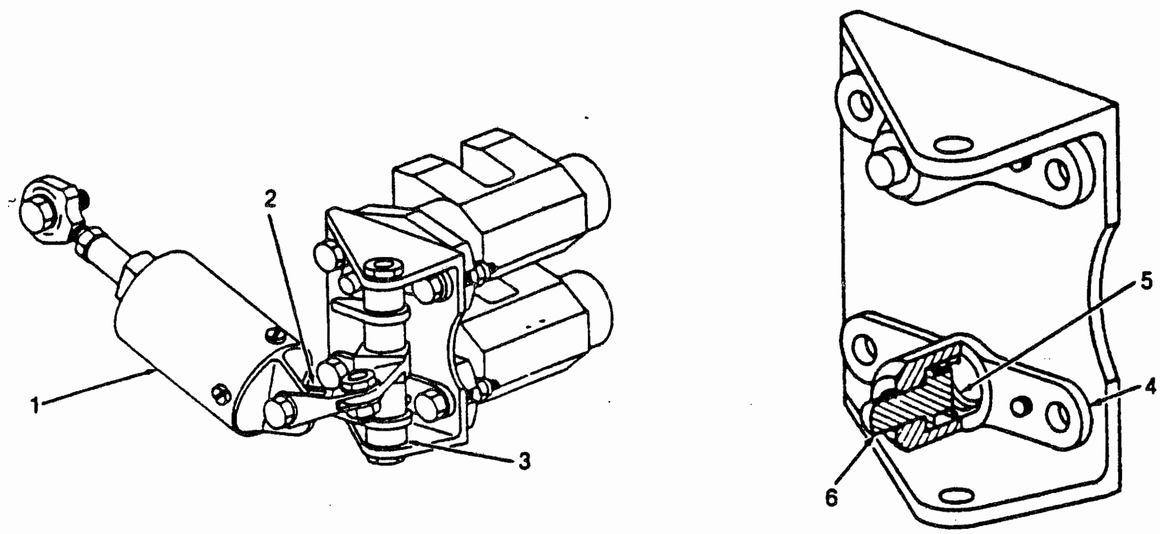
H. Install bolts, washers and nuts in proper sequence and tighten nuts finger tight, back off to first slot and install cotter pins.

I. Perform a Landing Gear System — Functional Test. (See Section 32-0)

4. Main / Nose Gear Timer Valves — Adjustment

A. Adjust each main gear timer valve so length of bungee assembly keeps plunger fully depressed when doors are fully open.

B. Adjust nose gear timer valves so that they are fully actuated when gear doors are in open position.



Main Gear Timer Bungee Linkage Assembly
Figure 201.

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SPEEDBRAKES SYSTEM — DESCRIPTION / OPERATION

1. General

The main gear is extended to act as speedbrakes. Speedbrakes are operated by pulling out the speedbrake control handle, located on the aft left side of center pedestal console. The handle controls only the main gear and speedbrake selector valve. It does not control the nose gear selector valve. Thus, only the main gear comes down. The speedbrake control handle also returns the main gear to the up position. Refer to Landing Gear — Description/Operation section, this chapter, for information on extension of main landing gear. With main gear down as speedbrakes, selection of landing gear down with pilots or copilots landing gear control handle will bring the nose gear down. Main gear will remain down. Mechanical linkage between nose gear selector valve and main gear and speedbrake selector valve keeps the latter in the down position when landing gear handle is moved to down, thus placing all gear in the same sequence.

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LANDING GEAR EMERGENCY EXTENSION AIR SYSTEM — DESCRIPTION / OPERATION

1. General

Refer to Landing Gear Control System — Description/Operation, Section, 32-4-0 for a general description of the emergency extension of the landing gear.

NOTE: The emergency air system can only extend the landing gear. There are no provisions for emergency retraction.

The air release valve and dump valve are mechanically linked together, and are connected by a teleflex cable to the emergency gear extension handle. Pulling the emergency gear extension handle out operates the air release valve and emergency extension system dump valve simultaneously. The dump valve opens up side of landing gear system to return. Air pressure operates a second dump valve (manual reset dump valve); this closes the pressure supply to both nose gear and main gear selector valves, and opens the selector valves to return. The air release valve releases the air supply in the air bottle (1950 psi) which shifts landing gear system shuttle valve and enters both gear down systems.

A. Main Gear — Emergency Extension

Emergency extension air enters the shuttle valve on the rod end of the main gear door cylinder, which shifts and permits air pressure to unlock the ball-lock in the door cylinder, retracting piston, and opening main gear doors. Through another path, air pressure travels to the shuttle valve at the rod (unlock) end of the uplock cylinder, retracting piston and releasing main gear uplock hooks. When main gear struts are freed, they will drop into the airstream, and airstream impact will force them aft and down. When main gear struts reach the limit of their travel, the drag brace locks will engage and secure main gear in down and locked configuration, thus completing the emergency extension of main landing gear. Main gear doors will remain open.

B. Nose Gear — Emergency Extension

Emergency extension air enters the shuttle valve on the rod end of the nose gear door cylinder, which shifts and permits door cylinder to retract and open nose gear doors. Fluid in the head end of door cylinder will pass through the door control valve, then travel to the down port of nose gear selector valve and return to the reservoir. It should be noted that return fluid coming from the head end of the nose gear door cylinder has two ways of reaching return. It can go through the check valve above the door control valve and reach return through the dump valve that was operated simultaneously with the air release valve, or it can pass through the door control valve, reaching the down side of the nose gear selector valve. If selector valve was in the UP position, flow will leave the valve at the pressure port and return to reservoir. If selector valve was in the DOWN position, flow will leave the valve at the return port, pass through the second dump valve and return to reservoir. The air pressure also travels to the dump valve that supplies hydraulic pressure to nose gear selector valve, shifts spool in the valve, cutting off pressure supply to the nose gear selector valve, and opening up a return for the system. Through interconnecting lines, air pressure also goes to the shuttle valve at the rod end of the uplock cylinder and to the shuttle valve near the head end of the nose gear cylinder. Uplock cylinder retracts and releases roller on the forward portion of the nose gear strut, and nose gear cylinder will extend bringing nose gear down. In emergency extension of the nose gear, nose gear doors will remain open.

On Aircraft 1 - 86, and 114 having ASC 116 and Aircraft 87 - 200, 322 and 323, manual resetting of the dump valve can be accomplished from a control located at the copilots position. This resetting operation permits normal hydraulic operation of the landing gear after emergency extension has been employed.

2. **Emergency Extension System Dump Valve**

The emergency extension system dump valve is a manually-operated, four-way, two-position, rotary selector valve actuated by the EMER LDG GEAR handle. It functions during emergency extension of the main landing gear to open up and down sides of the landing gear system to return. The valve has two positions and four ports, one of which is always plugged.

Actuation of the EMER LDG GEAR handle operates the air release valve and the emergency dump valve simultaneously through interconnected linkage. In position 1 (normal), the dump valve blocks pressure to all ports. In position 2 (dump), the up and down lines are open to the return port for return to the reservoir.

3. **Emergency Extension System Air Release Valve**

The emergency extension system air release valve is a pneumatic, three-port control and vent valve, located forward of air bottle on the right side of nose wheel well. The three ports are: inlet, from the air bottle through a conventional check valve (Aircraft not having ASC 215); outlet, to the landing gear emergency extension system; and vent to the atmosphere.

A lever on the valve is mechanically linked to the landing gear emergency extension dump valve and is connected by a teleflex cable to the EMER LDG GEAR handle located on the copilots side console. When EMER LDG GEAR handle is actuated, air release valve lever closes the vent port and opens the outlet port, discharging air bottle into landing gear down system. A minimum of 8° travel of the lever closes the vent (bleed) port. A maximum of 36° travel of the lever to full open is provided, and there is a minimum of 54° full travel.

4. **Emergency Extension System Air Bottle**

The air bottle (Kidde Part No. 891104) is located on forward right side of nose wheel well. It functions in conjunction with landing gear emergency air release valve when the EMER LDG GEAR handle is actuated. Air or nitrogen pressure in the bottle is directed to landing gear emergency extension system. Recharge air bottle if air gage for bottle reads below 1900 psi. Use an air-nitrogen (100 - 3000 psi) trailer.

5. **Emergency Extension System Air Gage and Filler Valve**

The air gage is attached to air bottle which is located on the right side of nose wheel well. It is similar to the accumulator air gage, and to main and auxiliary hydraulic pressure gages located in the cockpit. The dial of the gage is calibrated in pounds per square inch and reads from 0 to 2000. This gage records the pressure of the charge (either dry compressed air or nitrogen) in the bottle. The gage should read 1900 - 2000 when the bottle is fully charged.

The air filler valve (Schrader 798A) is attached to the air bottle located on the right side of the nose wheel well.

LANDING GEAR EMERGENCY EXTENSION AIR SYSTEM — MAINTENANCE PRACTICES

1. Returning Landing Gear to Normal Operation

To return the Landing Gear System to normal operation, See Section 32-0 Landing Gear Normal / Emergency — Operational Test, part of Landing Gear System — Functional Test.

2. Landing Gear Emergency Air Bottle — Removal / Installation

A. Removal

WARNING: LOOSEN BLEEDER VALVE ONLY ENOUGH TO ALLOW PRESSURE TO DISCHARGE SLOWLY. DO NOT ATTEMPT TO REMOVE BLEEDER VALVE WHILE PRESSURE REMAINS IN BOTTLE.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Gain access to air bottle located in right side of nose wheel well. Slowly open valve on servicing port and allow pressure to drop to zero psi before continuing further.

(2) Disconnect air lines and remove fittings from air bottle.

(3) Ensure all moisture accumulation is drained from bottle.

(4) Remove nuts from upper and lower clamps and remove bottle from aircraft.

(5) If bottle is removed for hydrostatic test, send to certified facility.

B. Installation

(1) Ensure that cylinder to be installed has current hydrostatic test date.

(2) Place bottle in position and reinstall clamps securing bottle to bulkhead.

(3) Install fittings using new O-rings.

(4) Connect lines to fittings.

(5) Service bottle with nitrogen or dry compressed air to 1950 $\pm$ 50 psi. Check for leaks with soapy solution.

(6) Remove safety devices.

(7) Perform Emergency Landing Gear — Operational Test, part of Landing Gear System — Functional Test, see Section 32-0.

3. Landing Gear Emergency Air Bottle — Hydrostatic Test

A. Remove landing gear air bottle. (see this section)

B. Send bottle to certified facility for hydrostatic test.

NOTE: Ensure that bottle to be installed has current hydrostatic test date.

C. Install air bottle. (See this section)

4. Landing Gear Emergency Air Bottle — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

DO NOT REMOVE BLEEDER VALVE WHEN AIR BOTTLE IS PRESSURIZED.

A. Ensure cylinder has current hydrostatic test date.

B. Loosen bleeder valve at bottom of emergency air bottle two turns and bleed nitrogen from bottle.

C. After nitrogen is bled from bottle, ensure all moisture accumulation is allowed to drain.

D. Inspect exterior of bottle for corrosion and damage. If bottle is found to be defective, remove and send to overhaul station for detailed inspection.

E. Install bleeder valve and torque 80 inch-pounds.

F. Remove cap from filler valve in nose wheel well connect nitrogen filler hose to filler valve. Loosen swivel nut on filler valve slowly (approximately 2 1/2 turns).

G. Slowly fill emergency air bottle with nitrogen until pressure gage indicates 1900-2000 psi.

H. Torque swivel nut on filler valve 50-70 inch-pounds.

I. Turn off nitrogen supply and remove supply hose.

J. Install valve cap.

5. Landing Gear Air Release Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF THE AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

WARNING: DO NOT REMOVE BLEEDER VALVE WHEN AIR BOTTLE IS PRESSURIZED.

(2) Slowly open valve on pneumatic servicing port located in nose wheel well and allow pressure in landing gear pneumatic system to drop to zero.

(3) Gain access to valve in nose wheel well forward upper right bulkhead and disconnect all lines and fittings.

(4) Disconnect rod from valve arm and retain hardware.

(5) Remove mount bolts and remove valve from mounting bracket.

B. Installation

(1) Position valve on mounting bracket and install mounting bolts.

(2) Connect rod to actuating arm on valve.

(3) Reconnect all lines and fittings to valve.

(4) Service pneumatic bottle to 1950 $\pm$ 50 psi with nitrogen or dry compressed air.

(5) Remove safety devices from landing gear and doors.

(6) Perform Emergency Landing Gear — Operational Test, part of Landing Gear System — Functional Test, see Section 32-0.

(7) Check for leakage at fittings.

(8) Operate gear five or six cycles to bleed air out of system.

(9) Lower jacks and remove jacks from aircraft.

6. Landing Gear Air Bottle Pressure Gage — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

DO NOT REMOVE BLEEDER VALVE WHEN AIR BOTTLE IS PRESSURIZED.

(1) Slowly open valve on air bottle servicing port located next to air bottle pressure gage on right side of nose wheel well. Allow pressure to drop to zero before continuing.

(2) Disconnect air pressure line from back of gage.

(3) Remove mounting screws securing gage to mounting bracket. Remove gage from aircraft.

B. Installation

(1) Place gage in position and install mount screws.

(2) Connect air pressure line to back of gage and tighten.

(3) Service bottle with nitrogen or dry compressed air to 1950 $\pm$ 50 psi.

(4) Perform leak check on fitting using soapy solution. Wipe off residue at completion of test.

(5) Remove safety devices as required.

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GROUND SERVICING SYSTEM — DESCRIPTION / OPERATION

1. Description

Since main and nose gear doors are normally closed with landing gear down and locked, provision is made for manually opening the three sets of doors for maintenance purposes.

A ground servicing valve with attached manual control lever is located on the right side aft end of the nose wheel well. The lever has two positions: FLIGHT, which is aft; and DOOR OPEN, which is forward. The lever is spring.loaded aft and locked in the flight position by a spring-loaded knurled pin. To open doors the knurled knob is pulled, unlocking lever, the lever is then moved forward and locked in place with a ground safety lock and red streamer (159GT1007).

Auxiliary hydraulic system pressure is required to open doors for ground service. During this operation the normal system door control valve is not disturbed. Its last position, with gear down and locked, ports main system pressure to the close side of the wheel well door cylinder. Applying main hydraulic system pressure closes doors. Because the ground control valve in flight position traps hydraulic fluid in UP and OPEN portion of the system, a thermal relief valve is incorporated to relieve pressure in excess of 2050 psi caused by thermal expansion.

2. Operation

NOTE: For certain servicing and preflight inspections, it is necessary to open the wheel well doors, ground servicing control valve is located on the right side aft end of the nose wheel well. Control lever has two positions; forward marked DOOR OPEN position and aft marked FLIGHT POSITION. The lever is spring-loaded aft and locked in this position by a spring-loaded knurled knobbed pin. To open the doors the knurled knob is pulled, unlocking the lever, the lever is then moved forward and is locked in this position with a separate ground lockpin with red streamer (159GT1007). Main dc power and auxiliary hydraulic pump are utilized in this procedure.

A. Wheel Well Door — Opening Procedure

To open wheel well doors from outside aircraft utilizing outside battery switch proceed as follows:

(1) Connect left battery.

(2) Open outside battery switch door (outboard side left nacelle).

(3) Place outside battery switch on.

(4) In nose wheel well, rotate ground servicing valve lever to door OPEN position and secure lever in place using ground safety lockpin and red streamer flag.

(5) If airstair door is open continue as follows:

(a) Set out side door switch (on nose wheel junction box - nose wheel well) to OPEN position and hold until all doors are fully open (auxiliary hydraulic pump will operate).

(b) Release switch to OFF position when doors are fully open.

(6) If airstair door is closed, proceed as follows:

(a) Unlatch and unlock airstair door using outside controls (Do not open door).

(b) Toggle OUTSIDE DOOR SWITCH (Nose wheel well junction box to CLOSE position) - switch will stay in this position. Auxiliary hydraulic pump will energize and wheel well doors will open.

(c) When doors are fully open, return OUTSIDE DOOR SWITCH to OFF.

(d) Lock and latch airstair door.

(7) Install the following ground safety struts to prevent inadvertent closing of doors and possible injury to personnel: Gulfstream Tool 159GT1011 (1 required) for nose gear doors and 159GT1041 (2 required) for main gear doors.

(8) Turn off OUTSIDE BATTERY switch and close access door. Disconnect left battery if desired.

B. Clamshell Door — Opening Procedure (Utilizing Auxiliary Hydraulic Pump Switches.)

NOTE: If power is available on aircraft and cockpit is accessible, the cockpit AUXILIARY HYDRAULIC PUMP switches (pilots or copilots) may be used instead of the outside door switch as follows:

(1) Place ground servicing valve lever in nose wheel well to DOOR OPEN position and lock lever with Grumman Tool 159GT1007.

(2) Place pilots or copilots auxiliary hydraulic pump switch in cockpit to ON position until doors are fully open.

(3) Auxiliary hydraulic pump switch to OFF position.

(4) Install nose and main gear door safety struts (159GT1011 and 159GT1014).

NOTE: Auxiliary hydraulic system pressure is required to open the doors for ground service. During this operation the normal system door control valve was not disturbed. Its last position with gear down ported the main system pressure to the close side of the wheel well door cylinder. Applying main hydraulic system pressure will, therefore, close the doors.

C. Wheel Well Doors — Closing Procedure

(1) Remove wheel well door ground safety struts 159GT1011 (1) and 159GT1014 (2).

(2) Remove lockpin 159GT1007 from ground service control valve, and return valve to FLIGHT.

WARNING: CLEAR ALL PERSONNEL FROM WHEEL WELLS BEFORE RETURNING VALVE TO FLIGHT POSITION AND BEFORE APPLY MAIN SYSTEM HYDRAULIC PRESSURE.

(3) Apply main system pressure by either of the following:

(a) Rotate either left or right propeller by hand in direction of rotation until all wheel doors are fully closed.

(b) Cranking cycle of either left or right engine.

(c) Starting cycle of either left or right engine.

(d) Attaching an external hydraulic rig to attachment fittings on right engine hydraulic pump.

(e) For Aircraft having ASC 109, in aft section of nose wheel well; rotate auxiliary to main selector valve from FLIGHT to MAIN ON position and lock in place using Gulfstream Tool 159GT1007.

(f) Energize auxiliary hydraulic pump by following the same procedures as outlined in door opening procedure. Remove tool 159GT1007 at completion of closing doors and ensure lever has returned to FLIGHT position.

LANDING GEAR POSITION INDICATING AND WARNING SYSTEM — DESCRIPTION / OPERATION

1. **General**

The position indicating and warning system provides positive indication of the position of the landing gear. A malfunction of any of the electrical components of this system does not affect the operation of the landing gear. Moving either of the interconnected landing gear handles operates the landing gear hydraulically. An internal red light, provided in each handle, comes on when the gear position does not correspond to the handle position.

A landing gear warning horn is provided which sounds whenever throttles are retarded to the 11,000 rpm mark (approximately 30° of quadrant travel), and all wheels are not locked down. A horn cutoff switch is provided which shuts the horn off and turns on the NO HORN warning light.

The main landing gear is used as speedbrakes in this aircraft, and therefore, the landing gear indication and warning system is involved during speedbrake descents to inform crew of the status of landing gear.

A solenoid locking device, known as the ground safety lock, is provided to prevent inadvertent movement of either landing gear handle to the UP position until airborne. It is equipped with a manual override in event of an electrical malfunction. The landing gear handle ground safety lock is operated from the left nutcracker switch only.

The nutcracker system provides ground or airborne information to certain systems requiring this data. The nutcracker system operates from both left and right main gear nutcracker switches through appropriate relays.

The intensity of all landing gear warning lights is controlled by a WHEEL LIGHTS INTENSITY switch, allowing selection of either BRIGHT or DIM mode of intensity. All landing gear lights except the landing gear handles red lights are tested through the warning lights test system.

Three configurations of landing gear indication and warning systems are installed in various aircraft. They will be covered separately and identified as follows:

IDENTIFICATION	DESCRIPTION COVERED IN SECTION
Not Having ASC 108A	32-9-1
Having ASC 108A Part II (Flap Gearbox Switch Only)	32-9-2
Having ASC 108A Part I (Complete modification)	32-9-3

NOTE: ASC 108A also indicates a Part III and IV. These parts provide corrections or improvements to the basic ASC 108. Parts III and IV will be disregarded, as all information is incorporated in Parts I and II, of ASC 108A in the text.

In utilizing the information which is contained in this section, ensure which system is installed in the aircraft and follow description of that particular configuration. Check logbook entries.

For Aircraft 1 - 94 and 114, use the following chart to determine extent of modification.

Steps:

A. Ensure aircraft effectivity (production) number and select category.

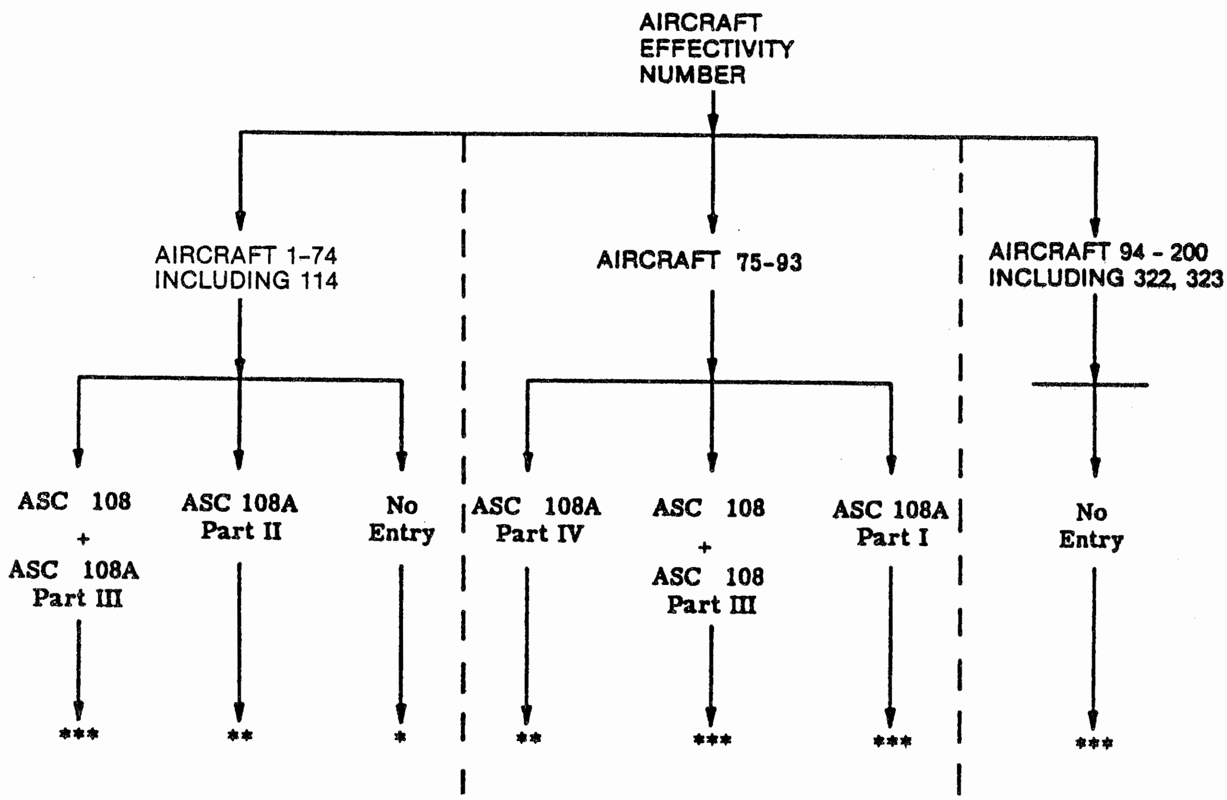
B. Check logbook entries pertaining to ASC 108 or ASC 108A.

C. Identify system utilizing the following key:

* Not having ASC 108A (See 32-9-1)

** Having ASC 108A Part II (Gearbox Switch only) (See 32-9-2)

*** Having ASC 108A Part I (Complete Modification) (See 32-9-3)



EXAMPLE: AIRCRAFT 46
LOG BOOK ENTRIES: ASC 108
ASC 108A PART III

STEP 1. Aircraft 46 falls into the category of 1-74

STEP 2. Logbook Entries: ASC 108 and ASC 108A Part III (under 1-74 Category), arrow indicates identity of system by ***.

STEP 3. ***Under KEY indicates that this aircraft has been modified to ASC 108A Part (complete modification). In order to get the landing gear warning system information pertaining to this aircraft refer to all pages in this chapter identified as 32-9-3. (Disregard 32-9-1 and 32-9-2 as they are for other configurations).

Aircraft Effectivity
Figure 1.

LANDING GEAR POSITION INDICATING AND WARNING SYSTEM — MAINTENANCE PRACTICES

1. **Gear Indication and Warning Switch — Adjustments**

A. Main Gear Downlock Switch

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Loosen cover of switch actuator housing and check for trapped moisture. Dry out if necessary.

(3) Tighten cover.

(4) Aircraft landing gear must be fully locked down.

(5) Break safety-wires at switch locknut and at electrical connector and remove switch wiring clamps.

(6) Loosen locknut and disconnect connector.

(7) Arrange circuit tester to monitor continuity between pins A and C of electrical plug.

(8) Rotate body of switch counterclockwise to remove from housing. Stop when circuit tester shows open circuit.

(9) Coat switch threads with sealing compound, MIL-S-7502; then cover with a 1012 finish, to protect compound from hydraulic fluid.

(10) Carefully screw switch body into housing, noting point at which test circuit closes.

(11) Screw switch body 3/4 turn into housing. Tighten locknut, connect electrical plug, and safety-wire both.

(12) Replace wiring clamps.

(13) Perform landing gear retraction cycle to check operation.

B. Main Gear Door Up Switch

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Break safety-wire and loosen switch locking nuts.

(3) Measure from lower face of bracket to end of switch stem threads, and adjust nut until this dimension 11/32 inch.

(4) Tighten nuts. There should be six threads showing between lower adjusting nut and end of stem.

(5) Perform landing gear retraction cycle to check operation.

(6) Safety-wire switch adjusting nuts.

NOTE: Should breathing of landing gear doors during flight cause landing gear handle lights to come on, the door up switch can be repositioned so that the door strikes the switch sooner. This increases the number of threads seen in Step (4) above.

C. Nose Downlock Switch

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Landing gear must be fully locked down.

(3) Break safety-wire at switch locking nuts and at electrical connector. Disconnect connector.

(4) Arrange tester to check electrical continuity between pins A and B of electrical plug.

(5) Loosen locknuts and permit switch to back away from striker until continuity tester shows open circuit.

(6) Note switch position at which continuity is just barely made.

(7) Position switch closer to striker by 2 full turns of nut. If switch plunger bottoms, back off nuts not more than 1 turn to obtain overtravel.

(8) Tighten locknuts, connect electrical plug, and safety-wire both.

(9) Perform landing gear retraction check to verify switch setting.

D. Nose Wheel Door Up Switch

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Jack aircraft.

(2) Disconnect nose wheel doors from push rods. Do not disturb length adjustment of these rods.

(3) In nose wheel well junction box, remove wire 2 from terminal 103-1. Arrange tester to monitor electrical continuity from this loose wire to aircraft structure.

(4) Break safety-wire and loosen switch adjusting nuts.

(5) Retract gear. Guide door push rods so that they do not foul structure, and keep doors pulled open so that they do not contact moving parts.

(6) Ensure strut is locked. Ensure landing gear is not moved while work in wheel well is in progress.

(7) Adjust switch until circuit tester just begins to show continuity.

(8) Move switch closer to striker by 2 full turns of adjusting nuts. Tighten nuts. Roller should roll when striker actuates. Reposition switch if necessary, and safety-wire clip. If switch plunger bottoms, back off nuts not more than 1 full turn to obtain overtravel.

(9) Extend landing gear. Observe precautions mentioned in Step (5) above.

(10) Safety-wire switch adjusting nuts.

(11) Reconnect wire 2 to terminal 103-1. Secure junction box cover.

(12) Perform landing gear retraction check to verify switch setting.

LANDING GEAR POSITION INDICATING AND WARNING SYSTEM NOT HAVING ASC 108A — DESCRIPTION / OPERATION

1. **Description**

For landing and on ground, the safety requirement is for gear to be down and locked. In flight, the safety requirements are for gear to be up and locked. The landing gear clamshell doors must be closed for a clean configuration. These doors cannot close unless gear is up and locked. Therefore, the indication circuit is arranged so that gear downlock switches provide down indication and door up switches provide up indication. (See Figure 1) When landing gear handle is moved to DOWN position, and gear extend, the corresponding green indicator light comes on as each gear locks down. As soon as the handle is moved, red handle lights will come on. As the last gear locks down, these red lights will go out.

Circuits for the red handle lights are completed through the landing gear handle transfer switch. The green indicator lights are completed through each of the downlock switches. All landing gear lights are dimmable by a WHEEL LIGHT INTENSITY switch located adjacent to the lights on the copilots left skirt panel. This switch operates a relay which cuts in individual resistances for dimming each light. Closing of the doors after the gear is up is used in the up indication circuit. For each set of gear, a switch attached to the structure is actuated when the clamshell doors close. If the handle is in the UP position, and one set of doors does not close, the red handle lights do not go out. This circuit is completed through the landing gear handle transfer switch and through each of the door uplock switches.

The landing gear warning horn sounds whenever the throttles are retarded to 11,000 rpm (approximately 30° of quadrant travel), and all the gear are not locked down.

Power for the landing gear indication and warning system is provided from the essential and main dc buses.

A. Component Location

Listed below are the components of this system, and their number and locations:

Unit	No. Per A/C	Location
Landing Gear Handle Transfer Switch No. 1	1	Cockpit floor, forward of pedestal.
Landing Gear Handle Warning Lights	2	In each landing gear handle.
Wheel Lights Intensity Switch	1	Copilots left skirt panel.
Warning Lights Test Relay D	1	Landing gear lights resistor box, left side of fuselage at Fuselage Station 97.
Right Wheel Downlock Switch	1	On right main gear drag brace.
Left Wheel Downlock Switch	1	On left main gear drag brace.
Nose Wheel Downlock Switch	1	Nose gear shock strut assembly.
Right Door Up Switch	1	Aft side of firewall of right main wheel well.
Left Door Up Switch	1	Aft side of firewall of left main wheel well.
Nose Wheel Door Up Switch	1	Right forward side of nose wheel.
Landing Gear Warning Horn	1	Cockpit overhead.
No-Horn Warning Light	1	Copilots left skirt panel.
Horn Cutoff Switch	1	Copilots left skirt panel.
Horn Cutout Relay	1	Left Fuselage Station 133 relay box.
Right Throttle Lever Switch	1	Actuated by right throttle lever in center ped-estal console.

Unit	No. Per A/C	Location
Left Throttle Lever Switch	1	Actuated by the left throttle lever in center pedestal console.
Landing Gear Handle Solenoid (Ground Lock)	1	Behind copilots left skirt panel.
Right Nutcracker Switch	1	On right main gear shock strut.
Left Nutcracker Switch	1	On left main gear shock strut.
Nutcracker Relay No. 1	1	Mid relay box floorboard No. 8.
Nutcracker Relay No. 2	1	Mid relay box floorboard No. 8.
Landing Gear Down Lights (L.G. DOWN LTS.) Circuit Breaker (5-amp.)	1	Left circuit breaker panel in cockpit.
Landing Gear Warning Lights (L.G. WARN LTS.) Circuit Breaker (5-amp.)	1	Left circuit breaker panel in cockpit.
Landing Gear Warning Horn (L.G. HORN) Circuit Breaker (5-amp.)	1	Left circuit breaker panel in cockpit.
Left Landing Gear Down Light	1	Copilots left skirt panel.
Right Landing Gear Down Light	1	Copilots left skirt panel.
Nose Landing Gear Down Light	1	Copilots left skirt panel.
Landing Gear Lights Dimming Resistors	6	Landing gear resistor box left side of Fuselage Station 97.
Landing Gear Lights Dimming Relay	1	Landing gear resistor box left side of Fuselage Station 97.

2. Operation

When the landing gear is set to DOWN and the gear is extending, a circuit through the LG WARN LTS 5 ampere circuit breaker causes the red handle landing gear warning lights to come on. (See Figure 2) As the last gear locks down, these red lights go out. As each individual gear locks down, its green landing gear down light will come on. The circuit to the red handle lights is completed through the landing gear handle transfer switch located on the cockpit floor, forward of the pedestal and the nose, left and right wheel downlock limit switches located on the nose gear shock strut and the left and right main gear drag braces.

The green down lights circuit is completed directly through each downlock switch.

When the handle is set to UP, if one of the three wheel doors fails to close after the gear is up, a circuit through the 5 ampere LG WARN LTS circuit breaker, the landing gear handle transfer switch and any one of the three door up switches will keep the red handle light on.

The three door switches, left, right and nose door up, are attached to structure and are actuated when the doors close.

A landing gear warning horn, located on the cockpit overhead, sounds whenever both throttle levers are retarded to 11,000 rpm or below (approximately 30° quadrant position) and all the wheels are not down and locked. The horn circuit receives power from main dc bus through the L.G. HORN 5 ampere circuit breaker through the left and right throttle switches and the de-energized contacts of the horn cutout relay, located on the left side of the Fuselage Station 133 relay box. The HORN CUTOFF switch, located on the copilots left skirt panel, if depressed, derives power through the same circuit breaker and the energized contacts of the horn cutout relay, stops the horn from sounding and lights the red NO HORN warning light on the copilots left skirt panel.

When the speedbrakes are extended, the following is the sequence of indication, (starting with gear up and locked and all doors closed, landing gear handles in the UP position)

A. Speedbrake handle is pulled.

B. Landing gear handle red lights come on as soon as first clamshell door starts to open. Both main gear clamshell doors open, and when open, both main gear uplocks release the gear.

C. Both gear are powered to down position and as each gear locks down the appropriate green gear down indication light signifies this fact through the downlock switches.

D. As each gear reaches down and locked, its clamshell doors reclose.

E. As the last clamshell doors close, the red landing gear handle lights go out (since all doors are closed).

F. If the throttle levers are retarded below 11,000 rpm (30° quadrant position) the landing gear warning horn will sound and the HORN CUTOFF switch may be utilized to silence to horn in the event of a power retarded let-down with speedbrakes extended. In the event that the horn is cut off by the switch, a red NO HORN light comes on, indicating the fact that all three gears are not down and locked and the horn is silenced. A check on this unsafe condition is confirmed by the fact that only two of the three green gear downlock lights are on. (NOSE is out).

When the crew selects gear down before landing, having made the first part of the descent in speedbrake configuration, the following is the indication sequence:

A. Gear handle DOWN. Red gear handle lights come on immediately since the transfer switch changes over the handle lights to the down lock switches, and the nose gear is not down and locked. (Speedbrake handle automatically resets to the in position by mechanical linkage).

B. The nose gear clamshell doors open fully, the nose gear uplock releases the nose gear. The nose gear is powered to down and locked. The green nose gear downlock light comes on when the nose gear locks down, and simultaneously the red landing gear handle lights go out, since all three gear are now down and locked. The nose gear clamshell doors then reclose to complete the cycle. Thus in the three down and locked configuration, three green gear down lock lights (left, right and nose) are on, and the gear handle red lights are off, signifying that the aircraft is safe to land. Retardation of throttle levers in this configuration will not sound the horn since all three gears are down and locked. However, if the NO HORN light has been turning on during the speedbrake descent and the throttle levers are not advanced above the 30° quadrant travel position after all gears are down and locked, it will remain on, due to a holding circuit in the relay. This relay may be released by advancing at least one throttle lever above the 30° quadrant position, and the NO HORN light will go out, but the horn will not sound.

When the aircraft is airborne, the left nutcracker switch mounted on the left main gear torque arm energizes the landing gear ground lock solenoid, located in the circuit breaker panel in the cockpit, through the 5 ampere L.G. HORN circuit breaker. The solenoid, when energized, removes a locking tab from the path of the copilots landing gear handle which enables the pilot or copilot to move either gear handle to the UP position. This tab may also be removed manually if required due to circuit or solenoid failure. The de-energized solenoid is a safety ground lock device. When the aircraft is on the ground it prevents the movement of the landing gear handles to the UP position. At no time does the solenoid prevent movement of the handles to DOWN.

When the aircraft is on the ground, the left and right nutcracker switches which are wired in series energize the nutcracker relays located in the mid relay box. To place them in the ground configuration to control those circuits which require this information.

The intensity of the three green gear down lights, the landing gear handle red lights and the red NO HORN warning light can be regulated by the WHEEL LT. INTENSITY SW. located on the copilots left skirt panel. This switch, when placed in the DIM position energizes, the landing gear lights dimming relay(s) inserting resistors in the various light circuits. The left, right and nose gear green down lock lights are each provided with a 75 ohm, 10 watt resistor adjusted to 71 ohms. The NO HORN light is provided with a 75 ohm, 10 watt resistor set at 50 ohms. Each red landing gear handle light is provided with a 50 ohm, 10 watt resistor adjusted to 43 ohms.

With the aircraft in emergency dc configuration, the three green down lock lights and the two landing gear handle unsafe red lights are operative. The warning horn, horn cutout circuit, nutcracker, NO HORN warning light and the ground safety lock solenoid are inoperative since they receive power form the main dc bus.

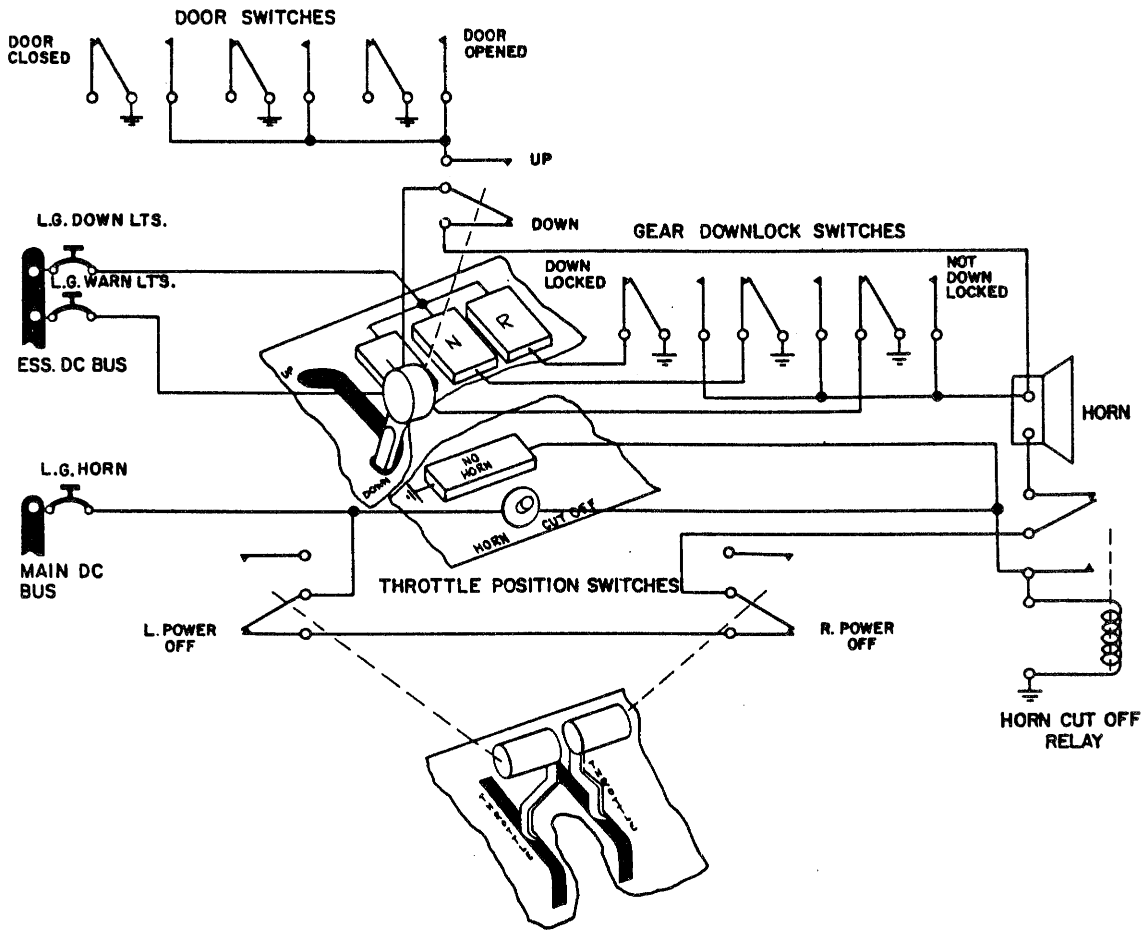
The following landing gear lights are tested through the WARN LIGHT TEST switches on the eyebrow panel:

- NO HORN light.
- Left gear down light.
- Right gear down light.
- Nose gear down light.

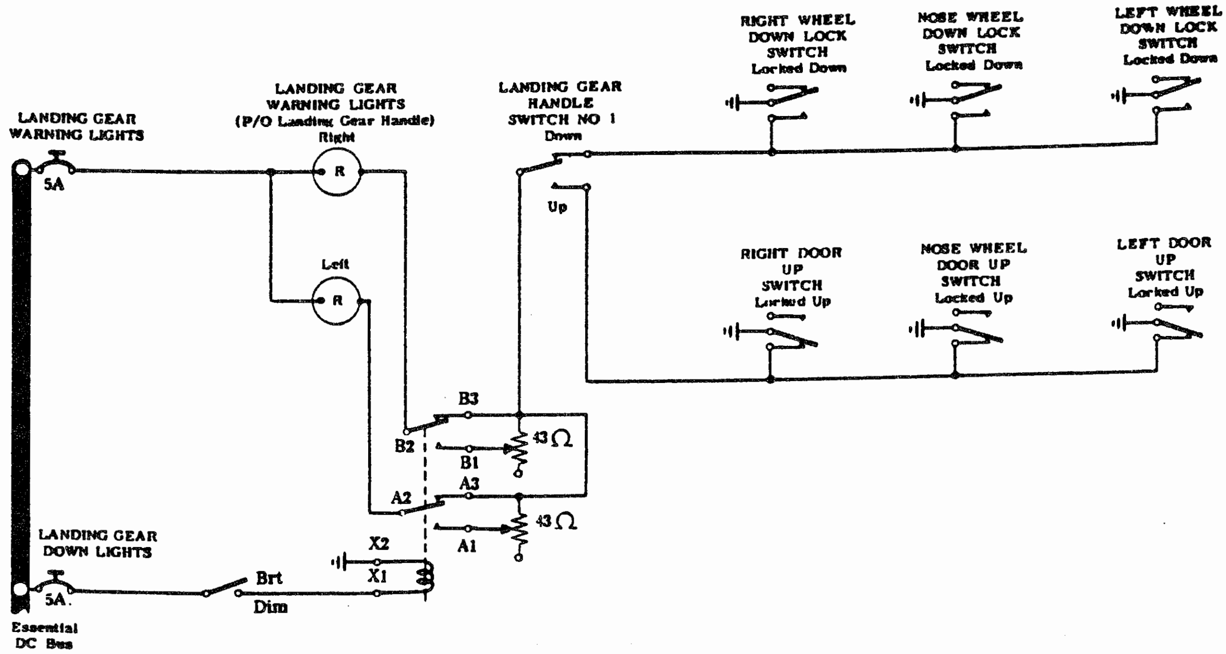
When depressed, the WARN LIGHT TEST switches (either one) will complete a circuit to all warning light test relays, energizing them for a bulb check on most warning lights in the cockpit. Warning light test relay D is involved with the landing gear lights listed above. This relay when energized isolates each light mentioned above from its normal system and places a ground on the particular light circuit to make the appropriate light come on for a bulb check, since they are not of the Press-To-Test type. Releasing the test switch returns the light circuit to its normal configuration. (Refer to Chapter 33 - LIGHTS - for details of the warning lights test system).

NOTE: The landing gear handle lights are not tested by the warning light test system. They are tested by deactivating the main dc bus (placing gang bar to Emergency) and retarding both throttles below 11,000 rpm (approximately 30° of quadrant travel).

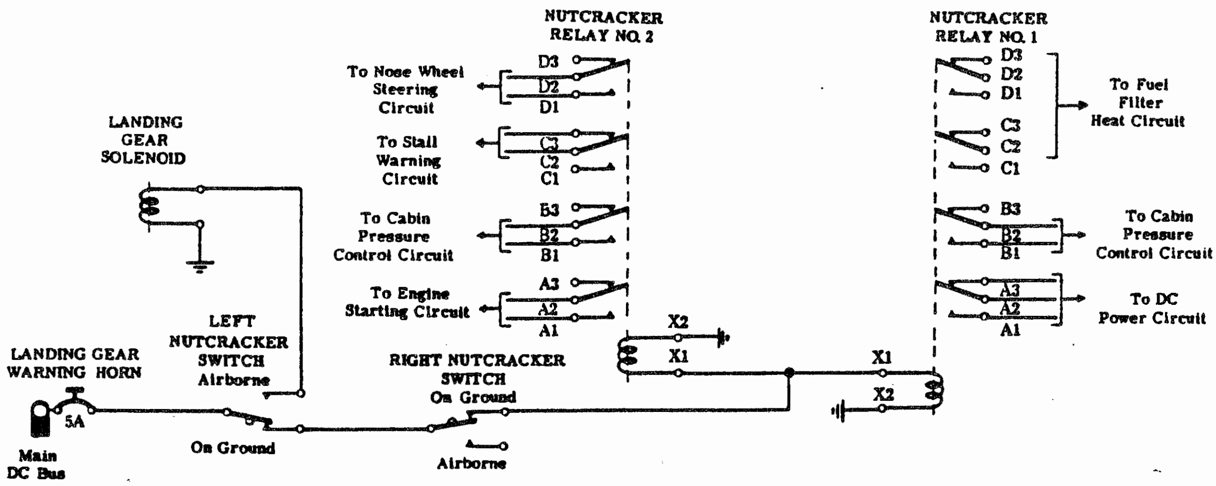
The warning light test function is not available in emergency dc operation.



Landing Gear Indication — Approach or Ground-Safe
Aircraft not having ASC 108A
Figure 1.

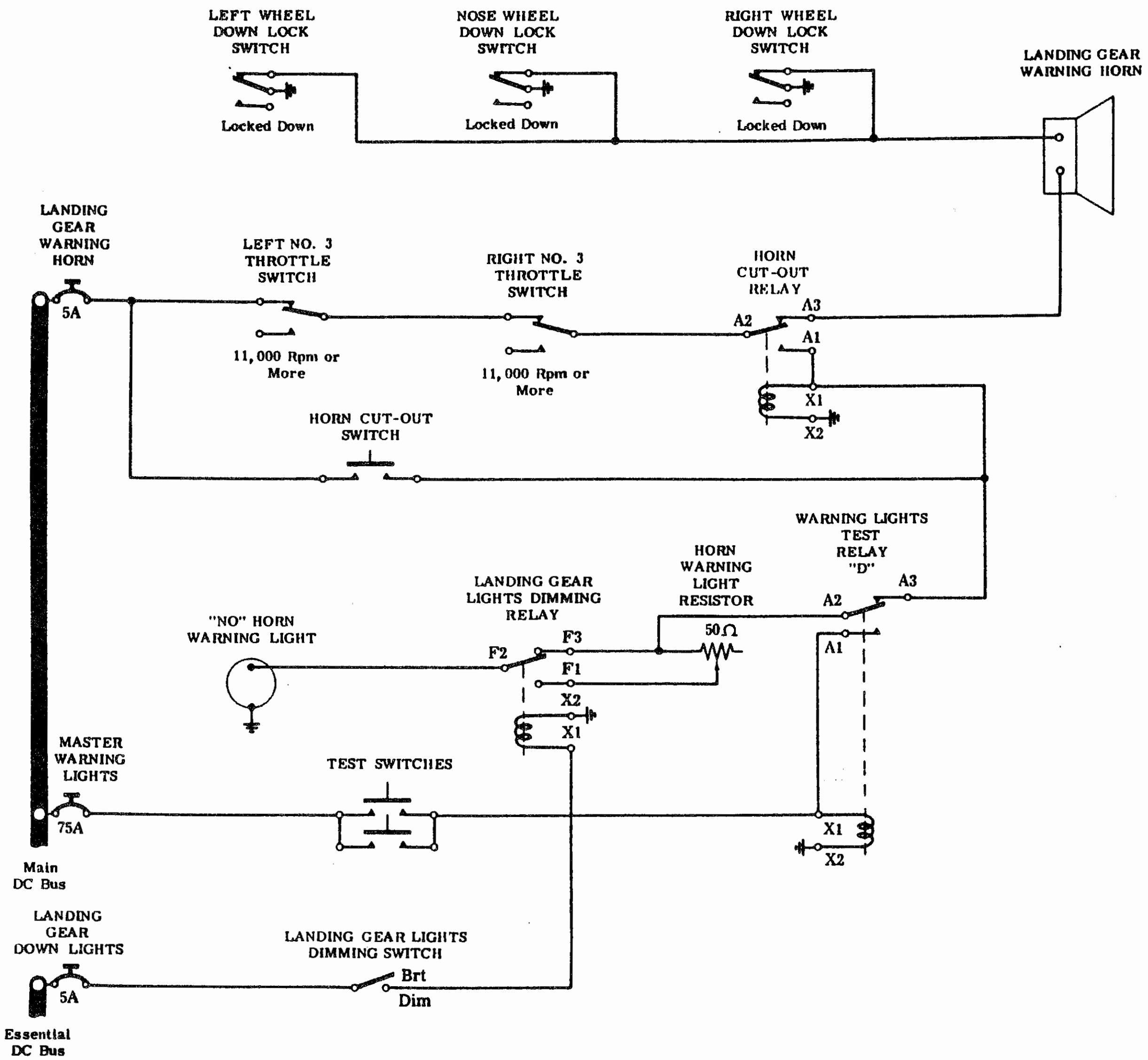


Landing Gear Handle Lights
(Without ASC 108A)

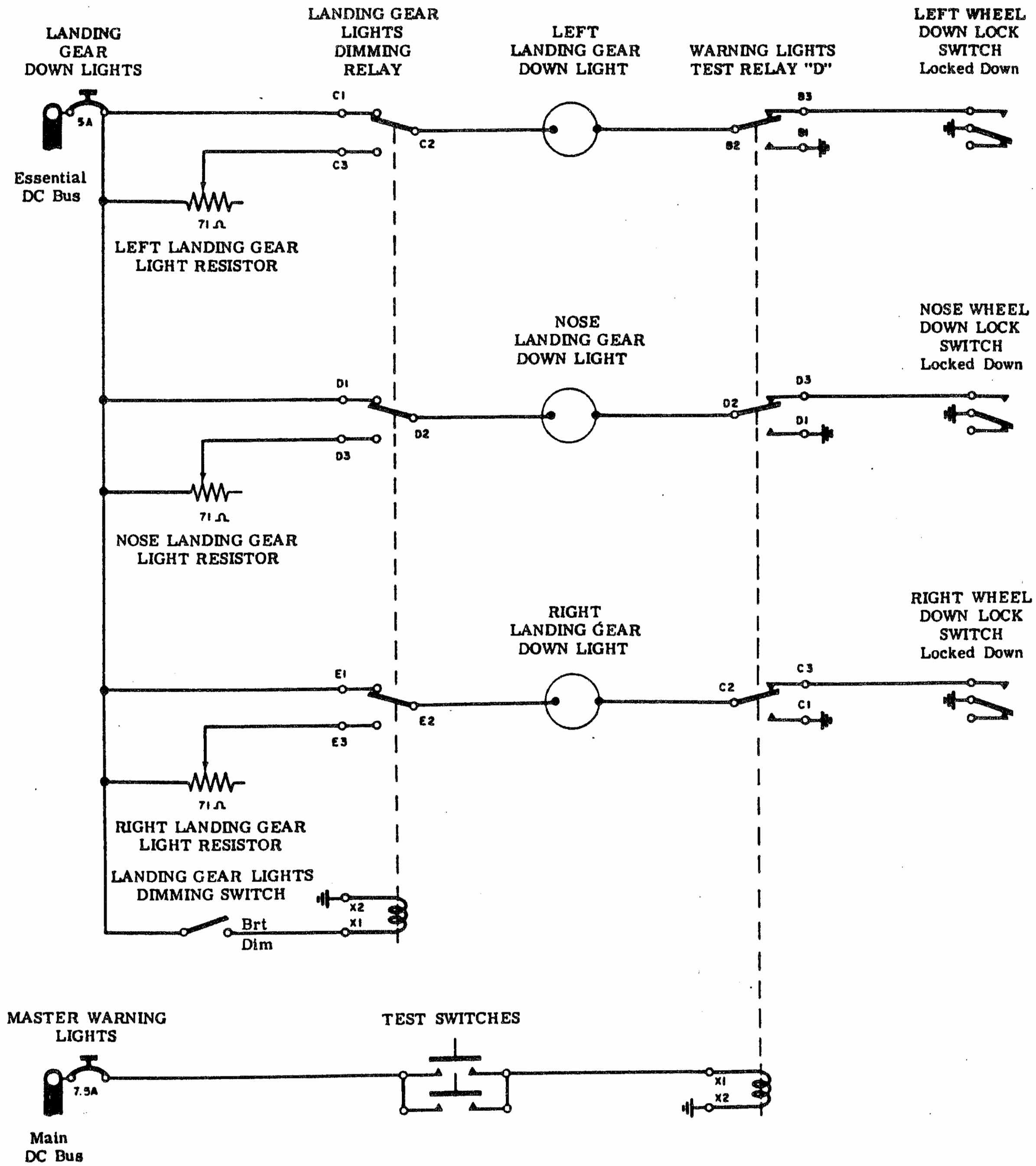


Landing Gear Safety Circuit
(Aircraft not having ASC 108A)

Landing Gear Position Indicating and Warning System — Schematic
Figure 2 (Sheet 1 of 3).



Landing Gear Position Indicating and Warning System — Schematic
Figure 2 (Sheet 2 of 3).



Landing Gear Down Lights
(Aircraft not having ASC 108A)

Landing Gear Position Indicating and Warning System — Schematic
Figure 2 (Sheet 3 of 3).

LANDING GEAR POSITION INDICATING AND WARNING SYSTEM NOT HAVING ASC 108A — MAINTENANCE PRACTICES

Refer Landing Gear System — Functional Test, see Section 32-0.

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LANDING GEAR POSITION INDICATING AND WARNING SYSTEM AIRCRAFT HAVING ASC 108A PART II — DESCRIPTION / OPERATION

1. **Description**

For landing and on ground, the safety requirement is for gear to be down and locked. In flight, safety requirements are for the gear to be up and locked. Doors cannot close unless the gear is up and locked. The indication circuit is arranged so that the gear downlock switches provide down indication and the door up switches provide up indication (See Figure 1) When landing gear handle is moved to DOWN, and gear extend, the corresponding green indicator light comes on as each gear locks down. As soon as handle is moved, the red handle lights will come on. As the last wheel locks down, these red lights will go out.

Circuits for the red handle lights are completed through the landing gear handle transfer switch, and green indicator lights are completed through each of the downlock switches. All landing gear lights are dimmable by a wheel light intensity switch located adjacent to the lights on the copilots left skirt panel. This switch operates a relay which cuts in individual resistances for dimming each light. Closing of the doors after gear is up is used in the up indication circuit. For each set of gear a switch attached to structure is actuated when door closes. If the handle is up and one set of doors do not close, the red handle lights do not go out. This circuit is completed through the landing gear handle switch and through each of the door uplock switches.

A landing gear warning horn sounds whenever all gear are not down and locked and throttle levers are retarded to 11,000 rpm (approximately 30° of quadrant travel) or the wing flaps are extended 20° or more (approach or full down). Flaps in the approach or full down position de-energizes the horn silencing relay and NO HORN light, causing the horn to sound warning to the crew that all three gear are not down and locked. The horn, when reactivated by flaps in approach or full down without three gear down and locked cannot be silenced by the horn cutout button. This makes it virtually impossible to land in speedbrake configuration without a warning with normal flap usage utilizing either normal flap extension or emergency flap extension.

Landing gear indication and warning system receive power from both the main and essential dc buses.

A. Component Location.

Listed below are the components of this system, and their quantity and locations:

Unit	No. Per A/C	Location
Landing Gear Handle Transfer Switch No. 1	1	Cockpit floor, forward of pedestal.
Landing Gear Handle Warning Lights	2	In each landing gear handle.
Wheel Lights Intensity Switch	1	Copilots left skirt panel.
Warning Lights Test Relay D	1	Landing gear lights resistor box, left side of Fuselage Station 97.
Right Wheel Downlock Switch	1	On right main gear drag brace.
Left WhDel Downlock Switch	1	On left main gear drag brace.
Nose Wheel Downlock Switch	1	Nose gear shock strut assembly.
Right Door Up Switch	1	Aft side of firewall of right main wheel well.
Left Door Up Switch	1	Aft side of firewall of left main wheel well.
Nose Wheel Door Up Switch	1	Right forward side of nose wheel well.
Landing Gear Warning Horn	1	Cockpit overhead.
No-Horn Warning Light	1	Copilots left skirt panel.
Horn Cutout Switch	1	Copilots left skirt panel.
Horn Cutout Relay	1	Left Fuselage Station 133 relay box.

Unit	No. Per A/C	Location
Right Throttle Lever Switch	1	Attached to right throttle lever on center pedestal console.
Left Throttle Lever Switch	1	Attached to left throttle lever on center pedestal console.
Landing Gear Handle Solenoid (Ground Lock)	1	Behind copilots left skirt panel.
Right Nutcracker Switch	1	On right main gear shock strut.
Left Nutcracker Switch	1	On left main gear shock strut.
Nutcracker Relay No. 1	1	Mid relay box floorboard No. 8.
Nutcracker Relay No. 2	1	Mid relay box floorboard No. 8.
Landing Gear Down Lights (L.G. DWN LTS.) Circuit Breaker (5-amp.)	1	Left circuit breaker panel in cockpit.
Landing Gear Warning Lights (L.G. WARN LTS.) Circuit Breaker (5-amp.)	1	Left circuit breaker panel in cockpit.
Landing Gear Warning Horn (L.G. HORN) Circuit Breaker (5-amp.)	1	Left circuit breaker panel in cockpit.
Left Landing Gear Down Light	1	Copilots left skirt panel.
Right Landing Gear Down Light	1	Copilots left skirt panel.
Nose Landing Gear Down Light	1	Copilots left skirt panel.
Landing Gear Light Dimming Resistors	6	Landing gear resistor box, left side of Fuselage Station 97.
Landing Gear Light Dimming Relay	1	Landing gear resistor box, left side of Fuselage Station 97.
Flap Gearbox Switch	1	Above flap central gearbox, rudder spring tab lockout sheave (FLR No. 15).
Flap Gearbox Switch Cam	1	On rudder spring tab lockout sheave, flap central gearbox (FLR No. 15)

2. Operation

When the landing gear is set to DOWN and the gear is extending, a circuit through the L.G. WARN LTS. 5 ampere circuit breaker causes the red handle landing gear warning lights to come on (See Figure 2). As the last gear locks down, these red lights go out. As each individual gear locks down, its green landing gear down light will come on. The circuit to the red handle lights is completed through the landing gear handle limit switch located on the cockpit floor, forward of the pedestal, and the nose, left and right wheel downlock limit switches located on the nose gear shock strut and the left and right main gear drag braces.

The green DOWN lights circuit is completed directly through each downlock switch.

When the handle is set to UP, if one of the three wheel doors fails to close after the gear is up, a circuit through the 5 ampere L.G. WARN LTS circuit breaker, the landing gear handle limit switch, and the three door up limit switches will keep the red handle lights on. The left, right and nose door up limit switches, are attached to the structure and are actuated when the doors close.

A landing gear warning horn, located on the cockpit overhead, sounds whenever both throttle levers are retarded to 11,000 rpm and all the gear are not down and locked. The horn circuit receives power from the main dc bus through the L.G. HORN 5 ampere circuit breaker through the left and right throttle lever switches and the de-energized contacts of the horn cutout relay, located on the left side of the Fuselage Station 133 relay box. The HORN CUTOFF switch, located on the copilots left skirt panel, deriving power through the same

circuit breaker and the energized contacts of the horn cutout relay, stops the horn from sounding and lights the NO HORN warning light on the copilots left skirt panel.

A horn rearming switch assembly is installed to alert the crew that all three gear are not down and locked when in APPROACH or FULL DOWN flap position. It consists of a switch attached to structure, and a cam fastened to the outboard side of the flap central gearbox rudder spring tab lockout sheave under FLR No. 15. This cam depresses the switch when the flaps are at 20° or more to accomplish the following:

A. Break the horn cutout relay holding circuit, making the relay de-energize.

B. Complete a hot feed from the L.G. HORN circuit breaker to the horn.

If the gearbox switch is depressed by the cam, and the horn silencing circuit having been previously activated, (such as in the case of a power retarded descent in speedbrake configuration), the switch will break the horn cutout relay holding circuit, turning off the NO HORN light, and another set of contacts in the same switch will complete a parallel hot feed to the horn, and if the three gears are not down and locked the horn will sound. This action occurs regardless of whether the throttles are retarded or not. The horn is incapable of being silenced with the flaps in this range by the use of the horn cutout button. The only condition which will silence the horn will be three gear down and locked, or flaps retracted to less than 20° This horn rearming switch system alerts the crew to the fact that, even though there are only two green gear downlocked lights on speedbrake descent, the nose gear must be lowered. The switch and cam action will take place during normal flap extension or flap extension utilizing the emergency flap handle, since the flap central gear box and rudder spring tab lockout sheave are motivated by either method of flap movement.

When speedbrakes are extended, the following is the sequence of indication: (starting with the gear up and locked and all doors closed, gear handles up)

A. Speedbrake handle is pulled.

B. Landing gear handles red lights come on as soon as first clamshell door begins to open. Both main gear clamshell doors open, and when open, both main gear uplocks release the gear.

C. Both gears are powered to down position and as each gear locks down the appropriate green gear down indication light signifies this fact through the downlock switches.

D. As each gear reaches down and locked, its clamshell doors close.

E. As last clamshell doors close, the red landing gear handle lights go out (as all doors are closed).

F. If throttle levers are retarded below 11,000 rpm (30° quadrant position) the landing gear warning horn will sound. The HORN CUTOFF switch may be utilized to silence horn in the event of a power retarded let-down with speedbrakes extended. In the event that the horn is CUTOFF by the switch, a red NO HORN light comes on, indicating that all three gear are not down and locked and that the horn is silenced, and that only two of the three green gear downlock lights are on (NOSE is out).

The following will take place, if in speedbrake configuration, the crew overlooks the fact that only two gear are down and locked and a landing is attempted.

As crew selects APPROACH flaps, flap drive motor through the central gearbox starts flap extension. As the cam on the flap central gearbox rudder spring tab lockout sheave reaches 20°, the switch mounted above it is depressed, breaking the holding circuit of the horn cutout relay. NO HORN light then goes out. Simultaneously, another set of contacts places a hot feed on the horn, and due to the fact that the nose gear is still up, its downlock switch contributes the ground to the horn, resulting in the horn sounding. Since the rearming switch feed goes directly from the L.G. HORN circuit breaker through the switch to the horn, movement of the throttle levers in advance or retard have no effect on the horn. The same is true of the HORN CUTOFF switch. Depressing the switch cannot silence the horn, since its relay circuit is broken by the disarming switch. Thus, without three gear down and locked, and in 20° or more (APPROACH or FULL DOWN), the horn cannot be silenced. This will prompt action on the part of the crew to place the gear handle down to lower the nose gear. As it locks down the horn will stop blowing.

When the crew selects gear down before landing, having made the first part of the descent in speedbrake configuration, the following is the indication sequence.

A. Gear handle down. Red gear handle lights come on immediately since transfer switch changes over the handle lights to the downlock switches, and nose gear is not down and locked. (Speedbrake handle automatically resets to the in position by mechanical linkage).

B. The nose gear clamshell doors open fully, nose gear up lock releases the nose gear. The nose gear is powered to down and locks. The green nose gear downlock light comes on when nose gear locks down, and the red landing gear handle lights go out, since all three gear are now down and locked. Nose gear clamshell doors then close to complete the cycle. Thus in the three down and locked configuration, three green gear downlock lights (L, R and N) are on, and the gear handle red lights are off, signifying that the aircraft is safe to land. If the horn rearming function of the flap gearbox switch assembly has been sounding the horn, as the third green light comes on, the horn will cease blowing, since the ground from the nose gear downlock switch has been removed from the horn circuit.

When the aircraft is airborne, the left nutcracker switch mounted on the left main gear torque arm energizes the landing gear solenoid, located in the circuit breaker panel in the cockpit, through the 5 ampere L.G. HORN circuit breaker. The solenoid, when energized, removes a locking tab from a path of the copilots landing gear handle which enables the pilot or copilot to move either gear handle to the up position. (This tab may also be removed manually if required due to circuit or solenoid failure). The de-energized solenoid is a safety ground lock device. When aircraft is on the ground it prevents the movement of the landing gear handles to the UP position. At no time does the solenoid prevent movement of handles to DOWN.

When the aircraft is on the ground, left and right nutcracker switches, which are wired in series, also energize the nutcracker relays located in the mid relay box to place them in the ground configuration to control these circuit which require this information.

The intensity of the three green gear down lights, the landing gear handle red lights and the NO HORN warning red light can be regulated by the WHEEL LIGHT INTENSITY SW. located on the copilots left skirt panel. This switch when placed in DIM energizes the landing gear lights relay(s) inserting resistors in the various light circuits. Left, right and nose gear green downlock lights are each provided with a 75 ohm, 10 watt resistor adjusted to 71 ohms. The NO HORN light is provided with a 75 ohm, 10 watt resistor set to 50 ohms. Each red landing gear handle light is provided with a 50 ohm, 10 watt resistor adjusted to 43 ohms.

With the BATT switch in EMERG, the three green downlock lights and two landing gear handle unsafe red lights are operative.

The warning horn, horn cutout circuit, horn rearming circuit (flap gearbox switch) NO HORN warning light, nutcracker and ground safety lock solenoid are inoperative since they receive power from the main dc bus.

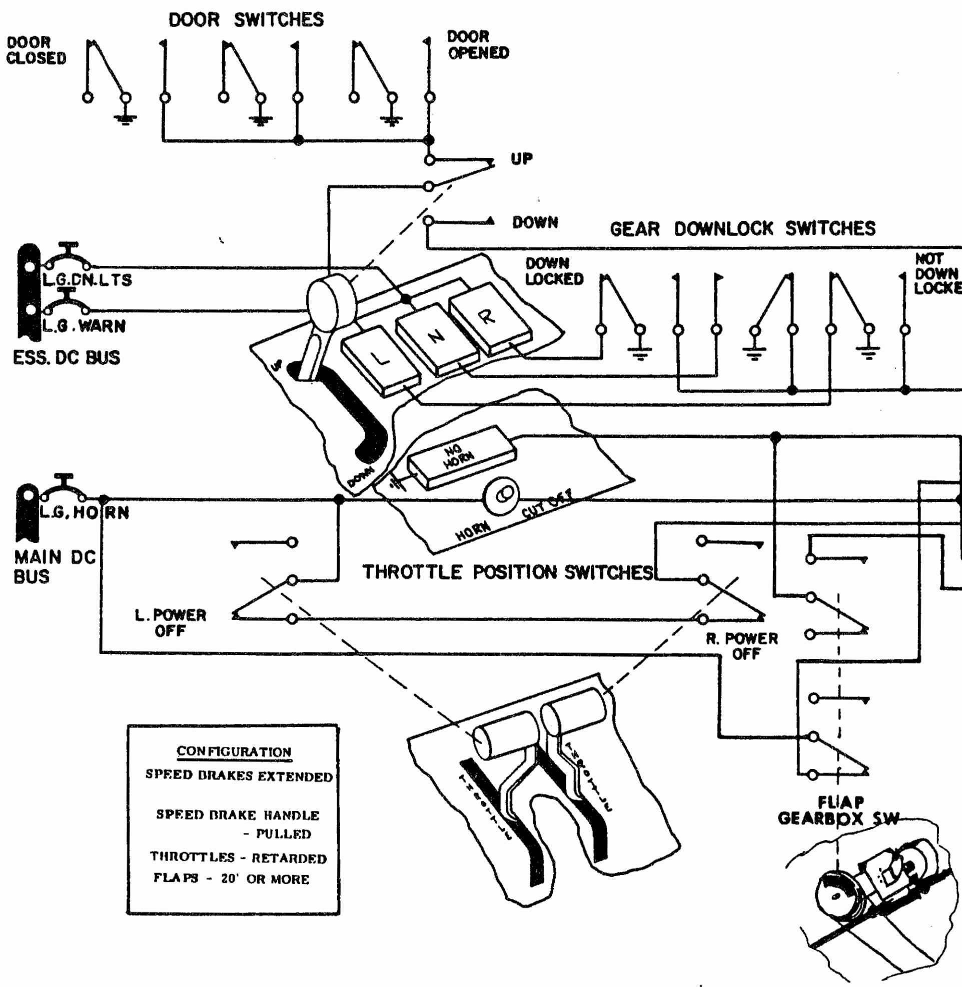
The following landing gear lights are tested through the WARN LIGHT TEST switches on the eyebrow panel:

- NO HORN light
- Left gear down light
- Right gear down light
- Nose gear down light

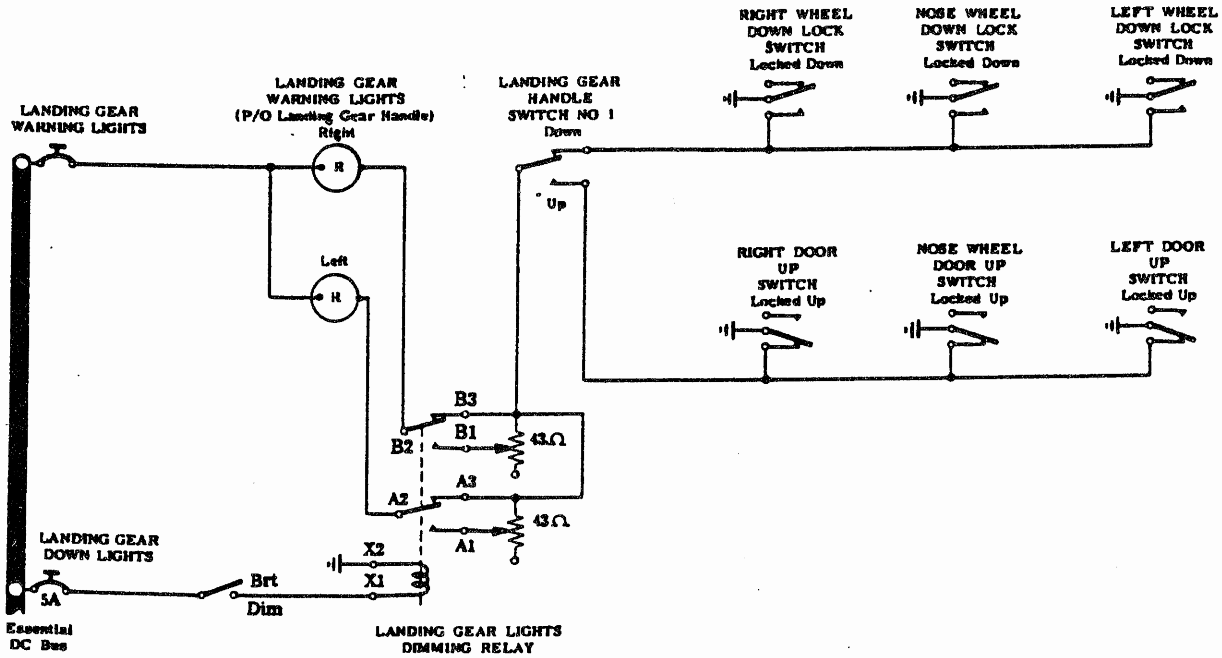
When depressed, the WARN LT TEST switches (either one) will complete a circuit to all warning light test relays, energizing them for a bulb check on most warning lights in the cockpit. Warning light test relay D is involved with the landing gear lights listed above. This relay when energized divorces each light mentioned above from its normal feed and places an appropriate feed on the particular light circuit to make the appropriate light come on for a bulb check, since they are not of the press-to-test type. Releasing the test switch returns the light circuit to its normal configuration.

NOTE: The LANDING GEAR HANDLE lights are not tested by warning light test system. They are tested by deactivating the main dc bus (placing gang bar to emergency) and retarding both throttles below 11,000 rpm (approximately 30° of quadrant travel) or extend the wing flaps to approach or below (20° or below).

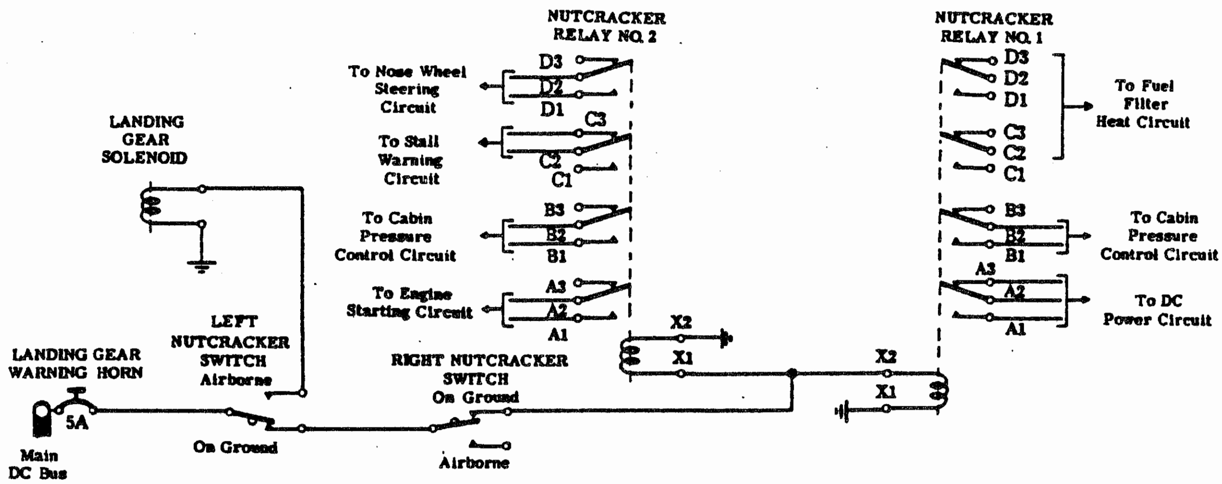
The warning light test function is not available in emergency dc operation.



Landing Gear Indication — Speedbrakes Extended Flaps 20° or More
(ASC 108A Part II)
Figure 1.

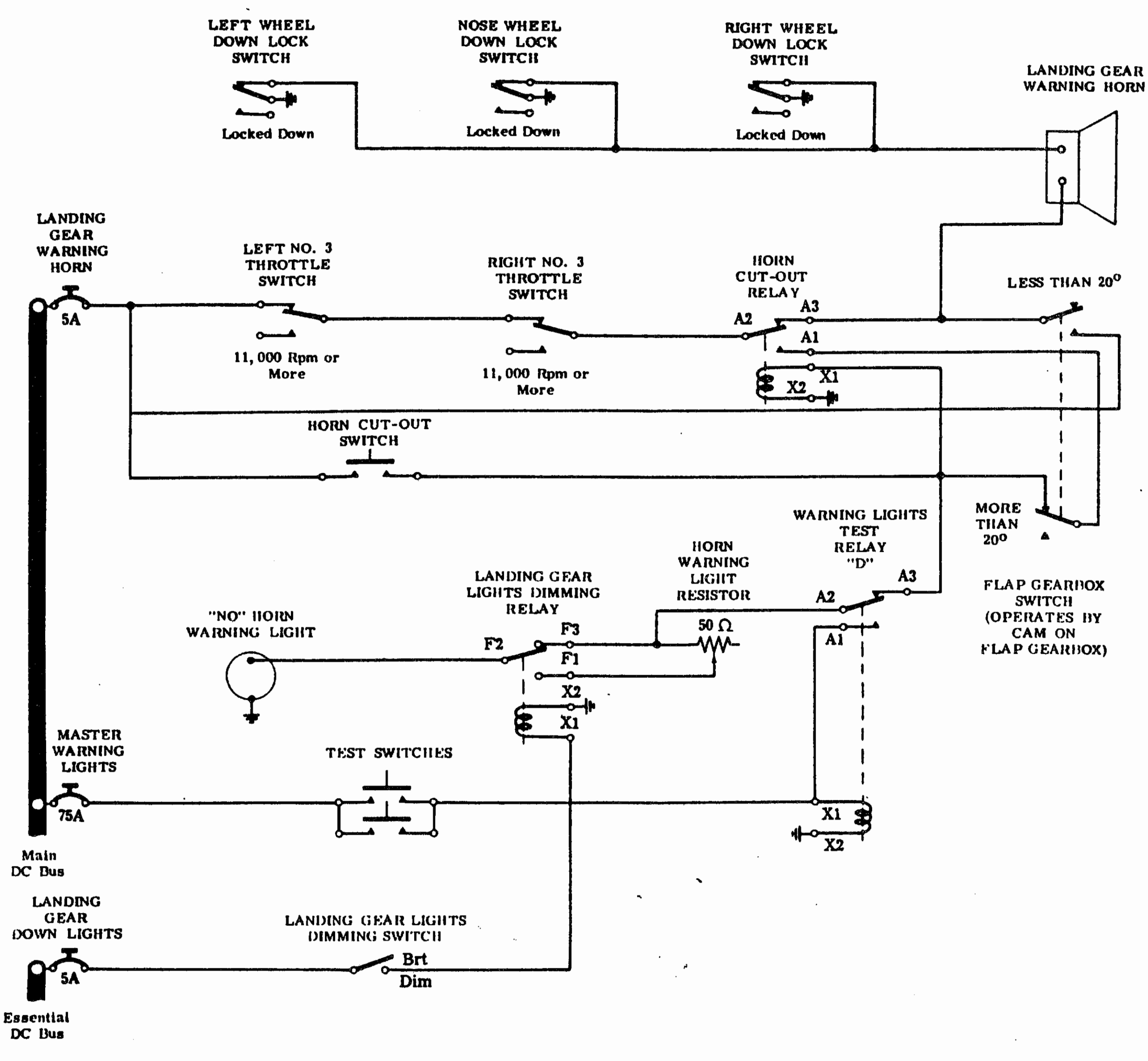


Landing Gear Handle Lights
(Aircraft having ASC 108A Part II)

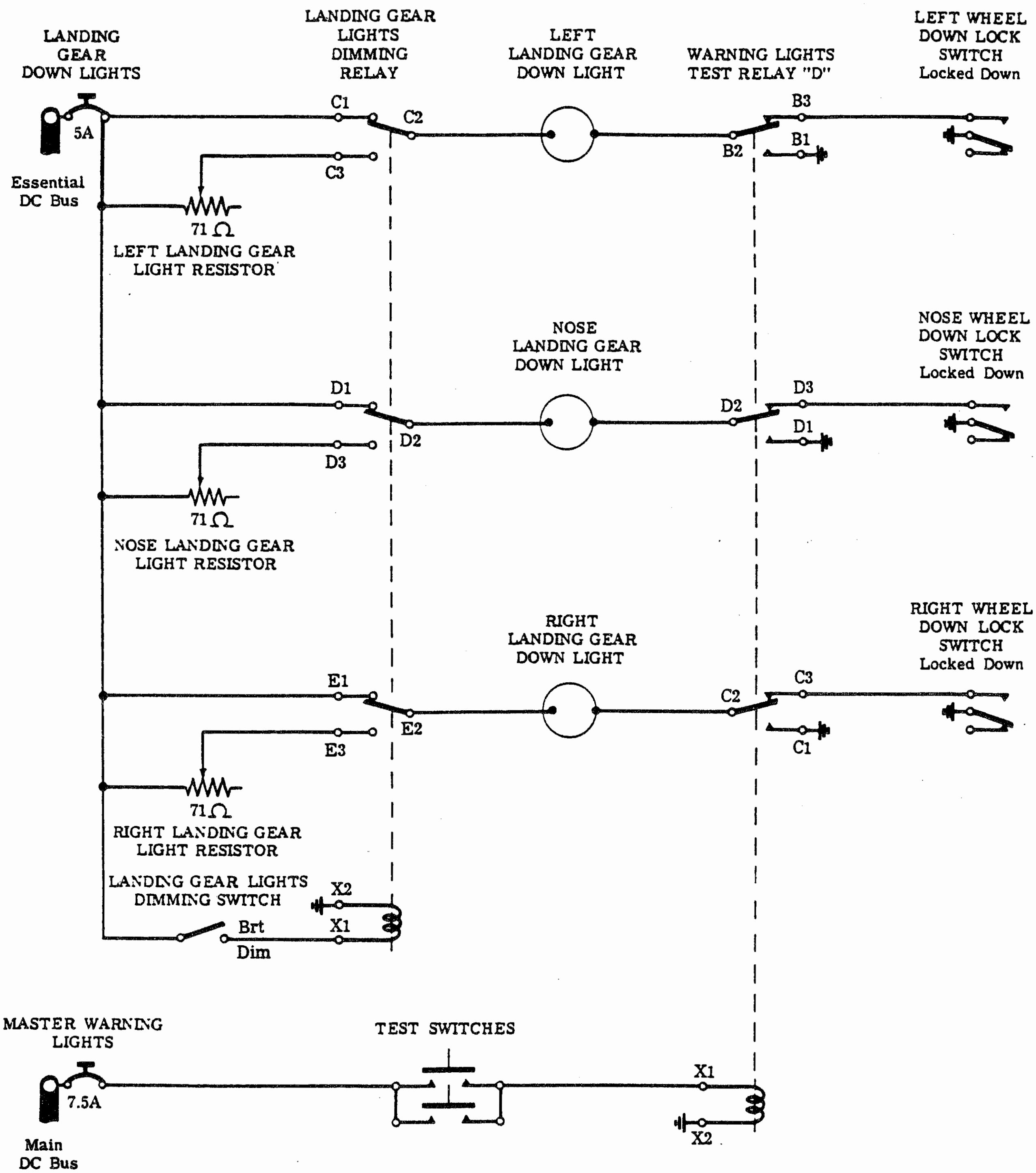


Landing Gear Safety Circuit
(Aircraft having ASC 108A Part II)

Landing Gear Position Indicating and Warning System — Schematic
Figure 2 (Sheet 1 of 3).



Landing Gear Position Indicating and Warning System — Schematic
Figure 2 (Sheet 2 of 3).



Landing Gear Position Indicating and Warning System — Schematic
Figure 2 (Sheet 3 of 3).

LANDING GEAR POSITION INDICATING AND WARNING SYSTEM AIRCRAFT HAVING ASC 108A PART II — MAINTENANCE PRACTICES

1. **Landing Gear Indication — Checkout**

   Refer to Landing Gear System — Operational Test. (32-0)

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LANDING GEAR POSITION INDICATING AND WARNING SYSTEM AIRCRAFT HAVING ASC 108A PART I — DESCRIPTION / OPERATION

1. Description

For landing and on ground the safety requirement is for gear to be down and locked. In flight, safety requirements are for gear to be up and locked. In addition, in flight the doors must be closed for a clean configuration. The doors cannot close unless gear are up and locked. The indication circuit is arranged so that the wheel downlock switches provide down indication and door up switches provide up indication. (See Figure 2) When the landing gear handle is moved to down position, and gear extend, the corresponding green indicator light comes on as each set of gear lock down. As soon as the handle is moved, the red handle lights will come on. As last gear locks down, these red lights will go out.

The circuits for the red handle lights are completed through the landing gear handle switch and through each of the downlock switches. The green gear down lights are turned on directly from the downlock switches. All landing gear lights are dimmable by a switch located adjacent to the lights. This switch operates relays which cuts in individual resistances for dimming each light. The closing of the doors after the gear is up is used in the up indication circuit. For each set of gear a switch attached to the structure is actuated when the door closes. If the handle is in the UP position, and one set of doors does not close, the red handle lights do not go out. This circuit is completed through the landing gear handle transfer switch and through each of the door uplock switches, therefore up indication is actually the red handle lights going out, signifying a safe configuration for flight. The landing gear warning horn sounds whenever the throttle levers are retarded to 11,000 rpm (approximately 30° of quadrant travel), or the normal flap handle is moved to approach or full down or the flaps are below 20° (approach or full down), and all the gear are not locked down. The flap handle or the flaps themselves in the APPROACH or FULL DOWN position also resets the horn silencing relay and turns off the NO HORN light. The speedbrake indicating lights are isolated from the landing gear indicators in speedbrake configuration only.

Power for this system is provided from the essential bus only.

A. Component Location.

Listed below are the components of this system, and their number and locations:

Unit	No. Per A/C	Location
Landing Gear Handle Transfer Switch No. 1	1	Cockpit floor, forward of pedestal.
Landing Gear Handle Warning Lights	2	In the landing gear handle.
Wheel Lights Intensity Switch	1	Copilots left skirt panel.
Warning Lights Test Relay D	1	Landing gear lights resistor box, left side of Fuselage Station 97.
Warning Lights Test Relay F	1	Right Fuselage Station 133 relay panel.
Right Gear Downlock Switch	1	On right main gear drag brace.
Left Gear Downlock Switch	1	On left main gear drag brace.
Nose Gear Downlock Switch	1	Nose gear shock strut assembly.
Right Door Up Switch	1	Firewall of right engine nacelle.
Left Door Up Switch	1	Firewall of left engine nacelle.
Nose Wheel Door Up Switch	1	Right side of nose wheel door.
Landing Gear Warning Horn	1	Cockpit overhead.
No-Horn Warning Light	1	Copilots left skirt panel.
Horn CUTOFF Switch	1	Copilots left skirt panel.

Unit	No. Per A/C	Location
Horn Cutout Relay	1	Left Fuselage Station 133 relay box.
Right Throttle Lever Switch	1	In center pedestal console, operated by right throttle lever.
Left Throttle Lever Switch	1	In center pedestal console, operated by left throttle lever.
Landing Gear Handle Ground Lock Solenoid	1	Behind copilots skirt panel.
Right Nutcracker Switch	1	On right main gear shock strut.
Left Nutcracker Switch	1	On left main gear shock strut.
Nutcracker Relay No. 1	1	Mid relay box.
Nutcracker Relay No. 2	1	Mid relay box.
Nutcracker Control Circuit Breaker (3 amp)	1	Left circuit breaker panel in cockpit.
Landing Gear Warning Lights (L.G. WARN LTS.) Circuit Breaker (5-amp.).	1	Left circuit breaker panel in cockpit.
Landing Gear Warning Horn (L.G. HORN) Circuit Breaker (5-amp.)	1	Left circuit breaker panel in cockpit.
Left Landing Gear Down Light	1	Copilots left skirt panel.
Right Landing Gear Down Light	1	Copilots left skirt panel.
Nose Landing Gear Down Light	1	Copilots left skirt panel.
Landing Gear Light Dimming Resistors	6	Below cockpit floor, Fuselage Station 97, left side.
Landing Gear Lights Dimming Relay	1	Below cockpit floor, Fuselage Station 97, left side.
Speedbrake Indicating Lights	2	Pilots inboard skirt panel.
Speedbrake Light Dimming Resistors	2	Right Fuselage Station 133 panel.
Speedbrake Lights Dimming Relay	1	Right Fuselage Station 133 panel.
Flap Down Wiring Relay	1	Right Fuselage Station 133 panel.
Flap Approach Warning Relay	1	Right Fuselage Station 133 panel.
Speedbrake Handle Transfer Switches	2	Pedestal.
Flap Gearbox Switch	1	Above flap central gearbox, rudder spring tab lockout sheave. FLR No. 15.
Flap Gearbox Switch Cam	1	Rudder spring tab lockout sheave, flap central gearbox. FLR No. 15

2. Operation

When landing gear is set to DOWN and gear is extending (not locked down), a circuit from the essential dc bus through the L.G. WARN LTS. 5 ampere circuit breaker causes parallel connected red handle landing gear warning lights to come on (See Figure 2). As last gear locks down, these red lights go out. As each individual gear locks down, its green landing gear down light will come on. The red handle lights circuit is completed through the landing gear handle transfer switch located on cockpit floor, forward of pedestal. The nose, left and right wheel downlock limit switches located on nose gear shock strut and left and right main gear drag braces. Green gear downlock lights circuit is completed directly through each downlock switch. When handle is set to UP, if one of the three clamshell doors fails to close after gears is up. A circuit through the 5 ampere L.G. WARN LTS. circuit breaker, landing gear handle transfer switch and three door up limit switches will the red

handle lights on. Three limit switches (left, right and nose door up) are attached to the structure and are actuated when the doors close.

Speedbrake indication is provided by two amber lights located on the pilots inboard skirt panel. Power is derived from the essential dc bus through the 5 ampere L.G. WARN LTS. circuit breaker. When speedbrake handle is pulled, two transfer switches are actuated in the pedestal which opens the circuit from the left and right green landing gear down indicator lights and closes circuit to left and right speedbrake indicator lights. When left and right main gear are in the down and locked position, gear downlock switches are actuated completing the circuit and the left and right speedbrake indicating lights will come on. Finally, as the clamshell doors reclose, the red landing gear handle lights will go out, since the landing gear handle is UP, sensing doors closed.

When landing gear handle is placed to DOWN, speedbrake control handle automatically retracts. Since the main gear is already in the down and locked position, as soon as speedbrake handle retracts, transfer switches are actuated, breaking the circuit to the two amber speedbrake indicator lights turning off the lights, and immediately a circuit closes turning on the left and right landing gear green indicator lights. As the landing gear handle is placed in the DOWN position the red handle lights come on, since the nose gear is not locked down. When nose gear reaches the down and locked position, its green indicator light will come on and red landing gear handle lights will go out.

A landing gear warning horn, located on the cockpit overhead, sounds whenever both throttle levers are retarded to 11,000 rpm and all the gears are not down and locked. The horn circuit receives power from the essential dc bus through the L.G. HORN 5 ampere circuit breaker through the left and right throttle lever switches and the de-energized contacts of the horn cutout relay, located on the left side of Fuselage Station 133 relay box. The HORN CUTOFF switch, located on the copilots left skirt panel, deriving power through the same circuit breaker and the energized contacts of the horn cutout relay, stops the horn from sounding providing the flaps are either in takeoff or up position, and lights the NO HORN warning light on the copilots left skirt panel. Depressing HORN CUTOFF switch momentarily completes a circuit to the horn cutout relay, energizing the relay via series connected contacts of de-energized flap down relay, flap approach relay and flap gearbox switch. The horn cutout relay remains energized by the circuit completed by the retarded throttle lever switches. The energized horn cutout relay also completes a circuit through the flap gearbox switch, de-energized contacts of flap approach and flap down relays to NO HORN warning light turns on the light. The NO HORN warning light is a reminder for the crew that horn has been deactivated. Should a landing be attempted with all landing gear not locked down and horn is deactivated, as soon as flap control handle is placed in either the APPROACH or DOWN position, appropriate flap relay is energized completing a circuit to the horn and the NO HORN warning light goes out. In this mode of operation, the throttle lever position switches and horn cutout relay are bypassed. The flap gearbox switch is used to reactivate warning horn when extending the flaps by emergency system. In emergency extension or retraction of flaps, the normal flap control handle is not used. If flaps are extended using the emergency method, when flaps reach the 20° approach position or lower, flap gearbox switch is activated by the gearbox cam sounding the horn and turning off the NO HORN warning light.

When aircraft is airborne, left nutcracker switch mounted on the left main gear torque arm energizes landing gear solenoid, located in the circuit breaker panel in the cockpit from the essential dc bus through the 3 ampere NUTCRACKER CONTR. circuit breaker. The landing gear handles are released by the solenoid. The de-energized solenoid is a safety ground lock device. When aircraft is on the ground it prevents movement of the landing gear handles to the UP position. At no time does solenoid prevent movement of handles to the DOWN position. When the aircraft is on ground, left and right nutcracker switches which are wired in series also energize nutcracker relays located in the mid relay box to place them in ground configuration to control those circuits which require this information. During a flight should the landing gear solenoid fail, preventing gear handle from being moved to UP position, the solenoid can be manually shifted to free the landing gear handles. Intensity of three green gear down lights, landing gear handle red lights, red NO HORN warning light and two amber speedbrake indicating lights can be regulated by the WHEEL LIGHT INTENSITY SW. located on the copilots left skirt panel. This switch when placed in DIM energizes the landing gear lights dimming relay and speedbrake indicating lights dimming relay inserting resistors in various light circuits. Left, right and nose gear green lights are each provided with a 10 watt 100 ohm resistor adjusted to 97 ohms $\pm$ 2 ohms. NO HORN light is provided with a 10 watt, 75 ohm resistor set at 50 ohms $\pm$ 2 ohms. Each red landing gear handle light is provided with a 10 watt, 450 ohm fixed resistor. Each speedbrake indicator light is provided with a 3 watt, 450 ohm fixed resistor.

With BATT switch in EMER position, the complete system is operative since it receives power from the essential dc bus.

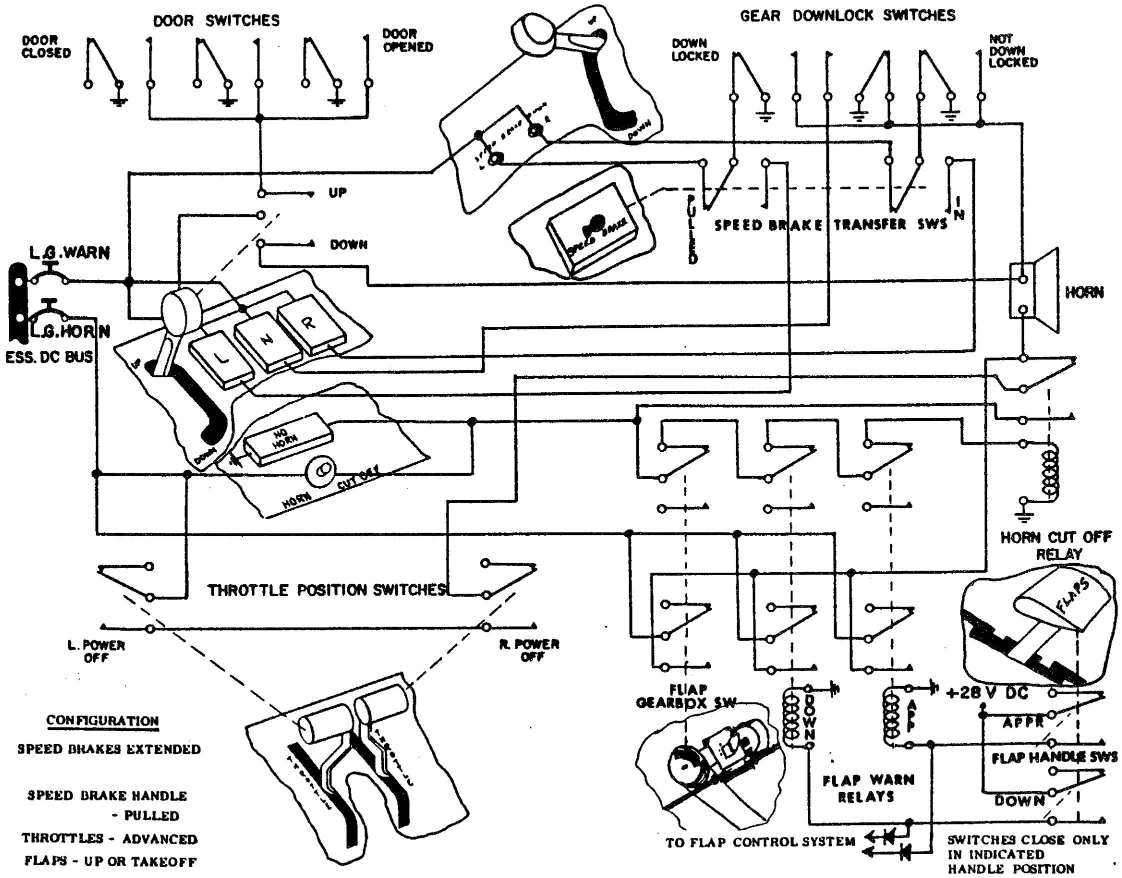
The following landing gear lights are tested through the WARN LT TEST switches on the eyebrow panel:

- NO HORN light
- Left gear down light
- Right gear down light
- Nose gear down light
- Speedbrake extended lights

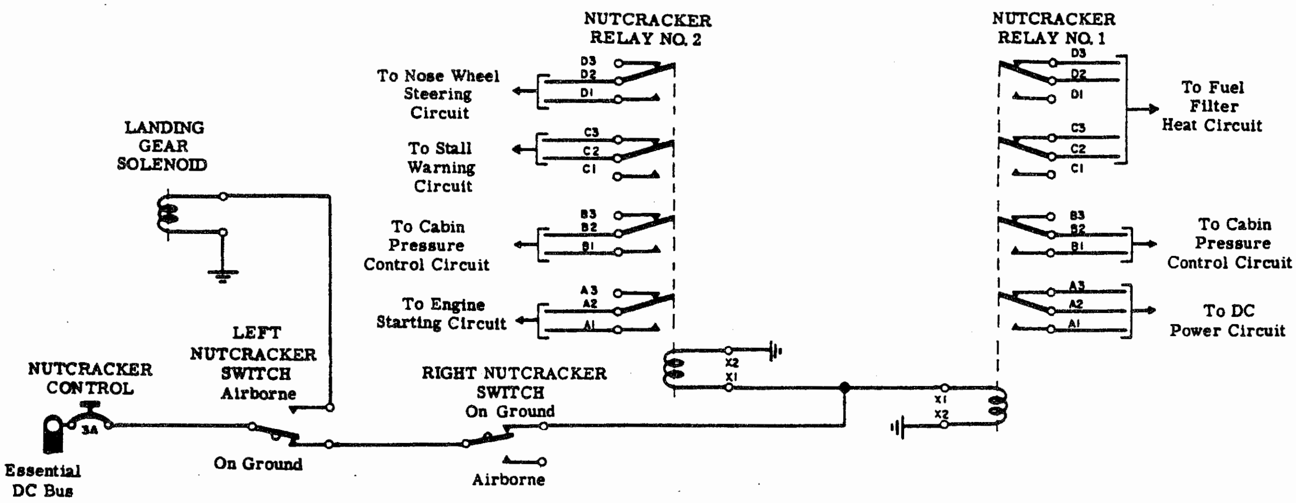
When depressed, the WARNING LIGHT TEST switches (either one) will complete a circuit to all warning light test relays, energizing them for a bulb check on most warning lights in the cockpit. Warning light test relays D and F are involved with the landing gear lights listed above. These relays when energized isolates normal system and places a ground on the particular light circuit to make the appropriate light come on for a bulb check, since they are not of the press-to-test type. Releasing the test switch returns the light circuit to its normal configuration.

NOTE: The landing gear handle lights are not tested by the warning lights test system. These lights are tested by pulling the L.G. WARN LTS circuit breaker and retarding both throttle levers below 11,000 rpm (approximately 30° of quadrant travel) or lower the wing flaps to 20° or more (approach or full down).

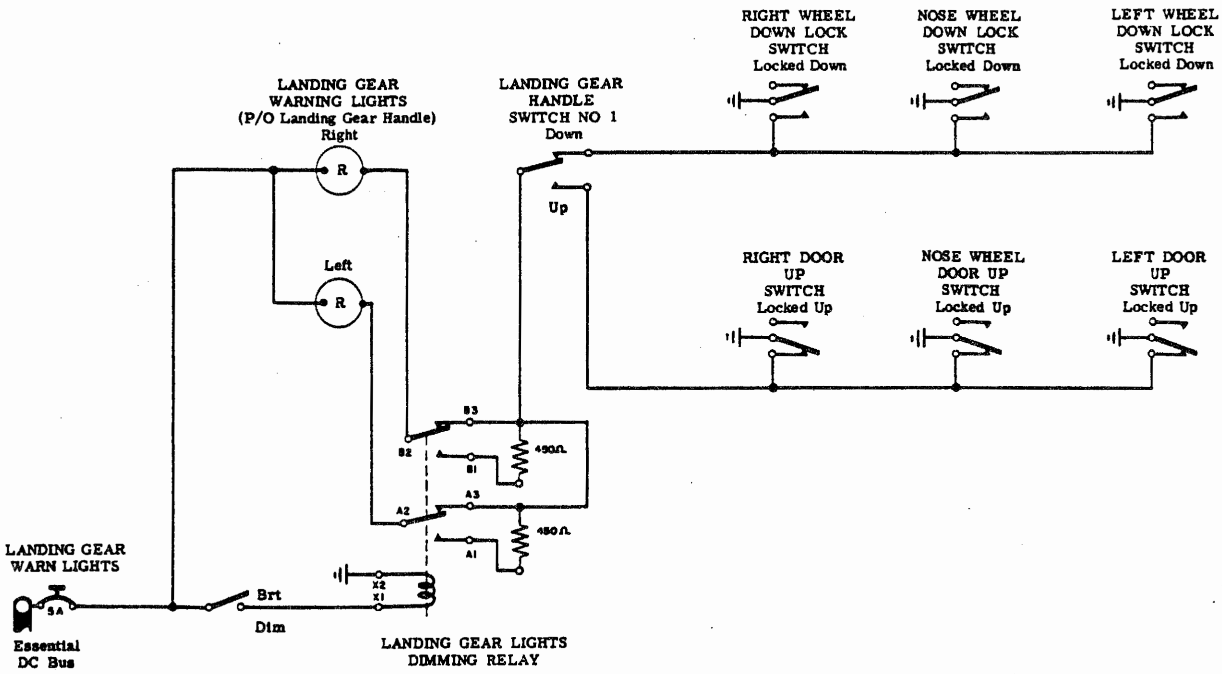
The warning light test function is not available in emergency dc operation.



Landing Gear Indication — Speedbrake Extended, Speedbrake Handle Pulled, Throttles Advanced, Flaps Up or Takeoff
Figure 1.

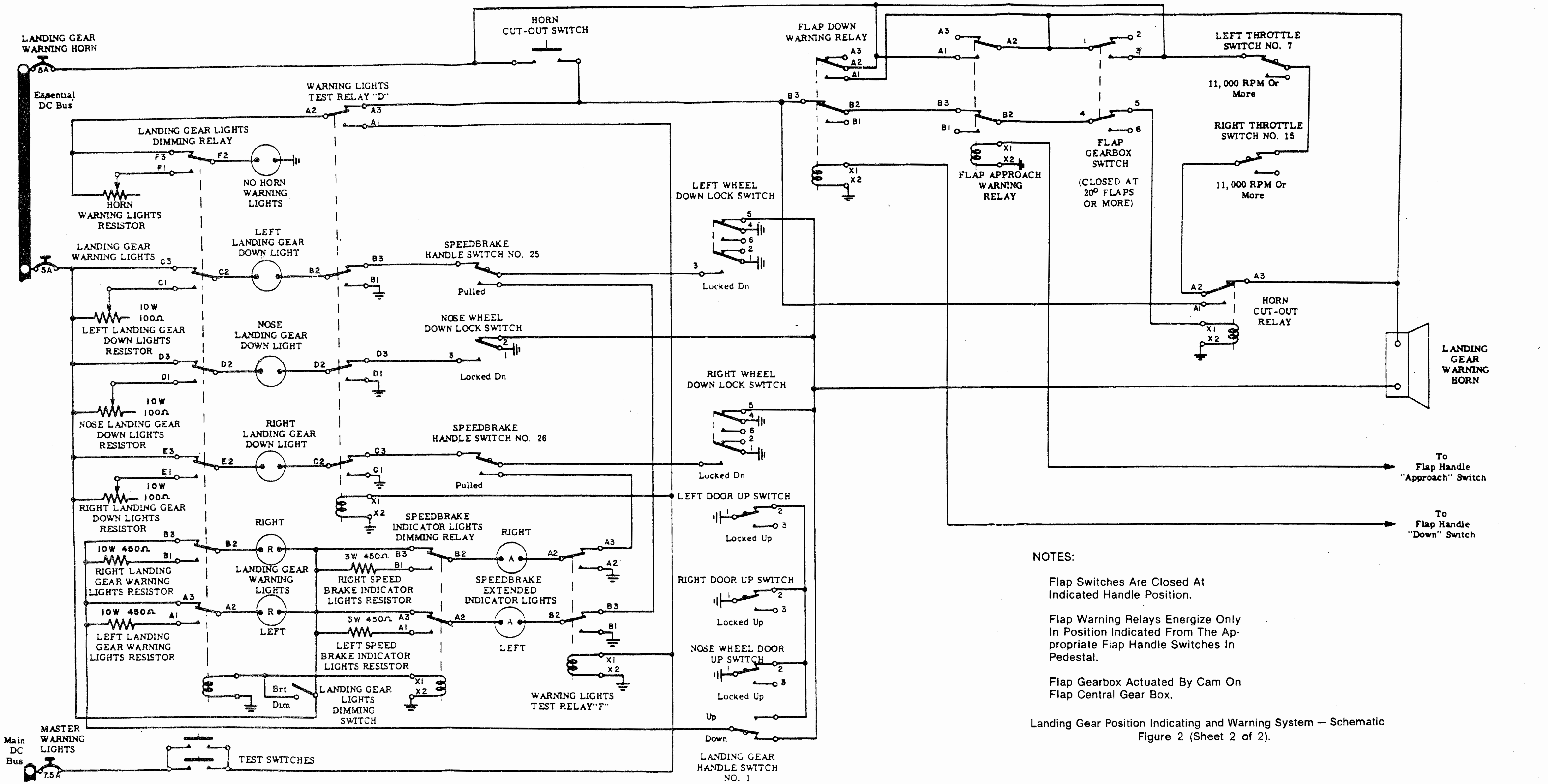


Landing Gear Handle Lights
(Aircraft Having ASC 108A Part I)



Landing Gear Safety Circuit
(Aircraft Having ASC 108A Part I)

Landing Gear Position Indicating and Warning System — Schematic
Figure 2 (Sheet 1 of 2).



NOTES:

Flap Switches Are Closed At Indicated Handle Position.

Flap Warning Relays Energize Only In Position Indicated From The Appropriate Flap Handle Switches In Pedestal.

Flap Gearbox Actuated By Cam On Flap Central Gear Box.

Landing Gear Position Indicating and Warning System — Schematic
Figure 2 (Sheet 2 of 2).

LANDING GEAR POSITION INDICATING AND WARNING SYSTEM AIRCRAFT HAVING ASC 108A PART I — MAINTENANCE PRACTICES

1. **Landing Gear Indication — Checkout**

   Refer to Landing Gear System — Functional Test, see Section 32-0.

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NUTCRACKER (SCISSOR) SYSTEM — DESCRIPTION / OPERATION

1. **General**

The nutcracker system provides on the ground or airborne indications to activate or deactivate various aircraft systems. Nutcracker switches are triggered when aircraft weight is removed from landing gear during takeoff and landing gear shock struts are extended. Nutcracker switches in turn actuate nutcracker relays.

2. **Description / Operation**

(Aircraft 1 - 93 and 114 not having ASC 108A Part I and not having ASC 206)

Power for nutcracker system (See Figure 1) is provided from the main dc bus through the LG HORN circuit breaker. When aircraft weight is removed from landing gear during takeoff, nutcracker switches (micro-switches), located on scissors, are triggered. As switches are triggered, nutcracker relays become de-energized. The two nutcracker switches are wired electrically in series, providing an airborne condition if weight is removed from either wheel. With weight on both main landing gear, three nutcracker relays are energized. Nutcracker relays are located in the mid-relay box under floorboard number 8. The LG HORN circuit breaker is located on the pilots circuit breaker panel.

NOTE: Depending on requirements of cabin pressurization system installed, only two nutcracker relays may be used.

In the airborne condition (nutcracker relays de-energized) the following systems are controlled as indicated:

A. Stall Warning System (operative)

B. Fuel Filter Heat System (operative)

C. Cabin Pressure Control System (operative)

D. Cabin Blower (operative)

E. Nose Wheel Steering System (inoperative)

F. Engine Start System (inoperative)

G. Landing Gear Safety Solenoid (inoperative when airborne)

H. DC Power Monitor System (inoperative)

3. **Emergency DC Operation**

(Aircraft 1 - 93 and 114 not having ASC 108A Part I and not having ASC 206)

Since the LG HORN circuit breaker is connected to the main dc bus, nutcracker relays cannot be energized at touchdown during emergency operation. In this situation, the systems controlled by nutcracker relays remain in airborne configuration, and the following conditions will exist:

NOTE: Conditions listed below also apply if the LG HORN circuit breaker pops or is pulled, and if either nutcracker switch fails in the airborne condition.

A. As stall conditions are reached, the stick shaker motor will operate. (Pull the STALL WARN circuit breaker to disable on ground.)

B. Cabin pressure is not dumped. (Land depressurized and place AIR COND-VENT switch in the VENT position to prevent cabin pressure buildup on ground.)

C. Blower does not stop on landing. (Place BLOWER switch OFF upon landing.)

D. Nose wheel steering system will not operate. (Use engine power and brakes for turns.)

E. On Aircraft 1 - 82 and 114 having ASC 117 and Aircraft 83 - 200, 322 and 323, fuel filter heaters will continue to operate. (Pull L FUEL FILTER and R FUEL FILTER circuit breakers to shutoff.)

F. DC Power Monitor System is inoperative.

4. **Description / Operation**

(Aircraft 1 - 93 and 114 having ASC 108A Part I and not having ASC 206 and Aircraft 94 - 200, 322 and 323 not having ASC 206)

Power for nutcracker system (See Figure 1) is provided from essential dc bus through NUTCRACKER circuit breaker. When aircraft weight is removed from landing gear during takeoff, nutcracker switches (micro-switches), located on scissors, are triggered. As switches are triggered, nutcracker relays become energized. The two nutcracker switches are wired, providing an airborne condition if weight is removed from either wheel. With weight on both main landing gear, three nutcracker relays are energized. The nutcracker relays are located in the mid-relay box under FLR No. 8. The NUTCRACKER circuit breaker is located on pilots circuit breaker panel.

NOTE: Depending on requirements of cabin pressurization system installed, only two nutcracker relays may be used.

In airborne condition (nutcracker relays de-energized) the following systems are in the conditions indicated:

A. Stall Warning System (operative)

B. Fuel Filter Heat System (operative or cyclic operation)

C. Cabin Pressure Control System (operative)

D. Cabin Blower (operative)

E. Nose Wheel Steering System (inoperative)

F. DC Power Monitor System (inoperative)

NOTE: If NUTCRACKER circuit breaker pops or is pulled, or if either nutcracker switch fails in the airborne condition, paragraph 3.A. thru 3.F. above will apply.

5. **Description / Operation**

(Aircraft 1 - 93 and 114 not having ASC 108A Part I and having ASC 206)

Power for nutcracker system (See Figure 2) is provided from main dc bus through the LG HORN circuit breaker. When aircraft weight is removed from landing gear during takeoff, nutcracker switches (micro-switches), located on the scissors, are triggered. As switches are triggered, nutcracker relays become energized. Each nutcracker switch is wired to its own nutcracker relay, located beneath the radio compartment floor at Fuselage Station 193. Normally-open contacts on each relay are connected in series from LG HORN circuit breaker. When energized by their respective nutcracker switches, relays provide a ground condition, and energize three more nutcracker relays in the mid-relay box under FLR No. 8. The LG HORN circuit breaker is located on pilots circuit breaker panel.

NOTE: Depending on requirements of cabin pressurization system installed, only two nutcracker relays may be used.

Two relays at Fuselage Station 193 are connected to the skid control system and prevent brake application in airborne (de-energized) condition. When relays beneath floorboard number 8 are in airborne (de-energized) condition, systems listed in paragraph 2.A. thru 2.H. above are controlled as indicated.

6. **Emergency DC Operation**

(Aircraft 1 - 93 and 114 not having ASC 108A Part I and having ASC 206)

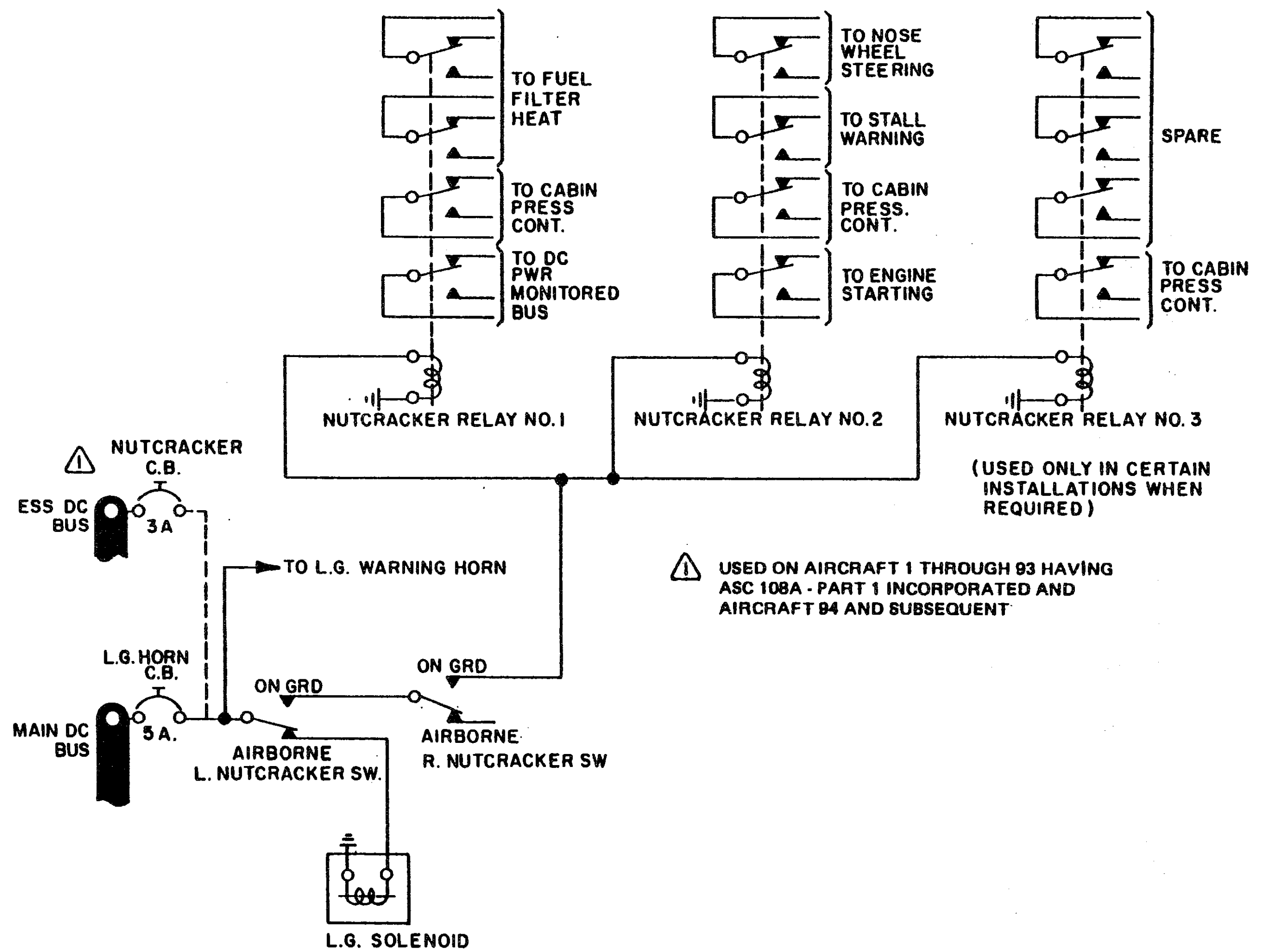
Since LG HORN circuit breaker is connected to the main dc bus, nutcracker relays cannot be energized at touchdown during emergency operation. Skid control system will become operative during wheel spin-up. Conditions listed in paragraph 3. above also apply.

7. **Description / Operation**

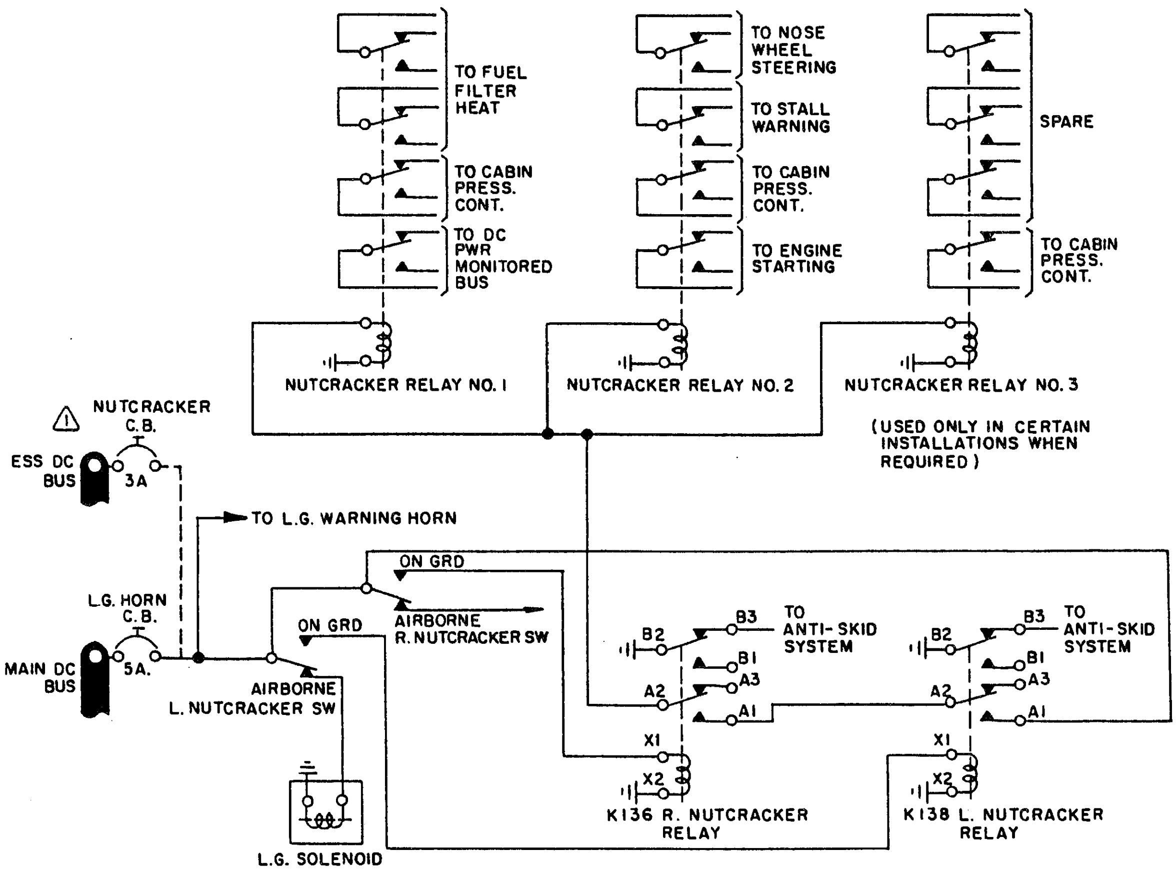
(Aircraft 1 - 93 and 114 having ASC 108A Part I and ASC 206. Aircraft 94 - 200, 322 and 323)

Power from the nutcracker system (See Figure 2) is provided from essential dc bus through the NUTCRACKER circuit breaker. Except for power source and circuit breaker nomenclature, operation is as given in paragraph 5. above.

NOTE: If the NUTCRACKER circuit breaker pops or is pulled, or if either nutcracker switch fails in airborne condition, paragraph 3.A. thru 3.F. above will apply.



Nutcracker System — Simplified Schematic
(Aircraft not having ASC 206)
Figure 1.



NOTE:

⚠1 USED ON AIRCRAFT 1 - 93 INCLUDING 114 HAVING ASC 108A. PART I AND AIRCRAFT 94 - 200 INCLUDING 322 and 323.

Nutcracker System — Simplified Schematic
(Aircraft having ASC 206)
Figure 2.

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NUTCRACKER (SCISSOR) SYSTEM — MAINTENANCE PRACTICES

1. **Nutcracker Switch — Adjustment**

NOTE: Aircraft having nutcracker relocation mod proceed to Step B.

A. Aircraft not having nutcracker relocation mod.

(1) Break lockwire at switch locking nuts on electrical connector. Disconnect connector.

(2) Jack aircraft until strut is fully extended.

(3) Loosen switch positioning nuts and permit switch to back away from strut until no continuity exists between pins A and B of electrical plug.

(4) Adjust switch so that a circuit between pins A and B is just made.

(5) Move switch closer to strut by two full turns of adjusting nuts. Ensure that switch plunger is not bottomed. If switch plunger bottoms, back off nuts to obtain over-travel. Final position of switch should be backed off at least one turn of nut from point at which continuity is established.

(6) Lower aircraft off jacks. As nutcracker compresses, switch should relax and its plunger move to fully extended position. Roller axis should be such that roller moves lengthwise along strut. Adjust roller alignment if necessary.

(7) Tighten locknuts and lockwire. Ensure that continuity does not exist between pins A and B.

(8) Jack aircraft. Ensure that before full strut extension, continuity between pins A and B occurs, and that it continues throughout strut extension.

B. Aircraft having nutcracker relocation mod.

(1) Break lockwire at switch locking nuts and at electrical connector. Disconnect connector.

(2) Jack aircraft until strut is fully extended.

(3) Loosen switch positioning nuts and permit switch to back away from strut until no continuity exists between pins A and B of electrical plug.

NOTE: Before final adjustment, insert a length of .020 or .025 stainless steel lock wire in top adjustment nut and twist a few turns. This will save time and effort should safety hole end up at rear of the bracket where it would be difficult to reach.

(4) Hold/secure torque arm so that microswitch roller is clear of cam crown.

(5) Adjust switch to provide .060 to .065 inch clearance between roller and cam. Switch should show continuity between pins A and B of electrical plug, when oleo piston extends approximately 12 inches. If necessary roller clearance should be adjusted slightly to read continuity between pins A and B.

(6) Safety nutcracker switch lock nuts.

(7) Check operation and alignment of switch by moving torque arm from approximate horizontal to the down position. Switch should operate smoothly and come to rest on crown of cam with torque arm in flight or extended position.

(8) If center line of the roller does not come approximately in center line of cam a shim shall be installed under bracket.

NOTE: When using shim, substitute 2 AN3-11A bolts for AN3-10A and use 1 AN960-10 washer under each bolt head instead of AN960-10L.

(9) Lower aircraft off jacks.

(10) Ensure that continuity does not exist between pins A and B.

(11) Jack aircraft. Ensure before full strut extension, continuity between pins A and B occurs, and continues throughout strut extension.

(12) Reconnect electrical connector and safety wire.

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NOSE WHEEL STEERING SYSTEM — DESCRIPTION / OPERATION

1. General

Nose wheel steering is accomplished through a hydraulically powered unit mounted integrally with the torque arm of the nose strut. Control is by a steering tiller bar which is mechanically connected to the nose wheel steering unit control valve. The tiller bar unit includes a self-centering artificial feel device. Powered steering is limited to 57 ± 20° either side of straight ahead. Fifty-seven degrees of nose wheel movement pivots the main wheel, which is permissible if brake is off.

Hydraulic pressure to nose wheel steering unit is supplied by the main hydraulic power system and is controlled by a solenoid operated valve. This valve is normally closed. To energize (open) this solenoid valve, the following conditions must be met: steering power switch located on the forward end of pilots console must be on (this is a guarded switch which is ON with the guard down), all three gear must have sufficient weight on them to fully extend all three nutcracker switches, and nose gear must be locked down.

The pilot can deactivate the power steering by lifting the guard and placing the STEERING POWER switch OFF. This is normally done if steering unit control malfunctions. With solenoid valve closed, or a loss of pressure in main hydraulic power system, nose wheel steering unit automatically assumes a shimmy damper configuration. During emergency dc operation, nose wheel steering is inoperative.

2. Description

Nose wheel steering system is comprised of a combination of electrical and hydraulic components: nose wheel steering valve, filter, steer swivel (located at the strut), steer-damper unit, also at the strut, nose wheel steering control, (located on the pilots side console) nose wheel power steering switch (located on the pilots side console) and other electrical components. (See Figure 1 and Figure 2) Nose wheel has a steering arc of approximately 60° left and 60° right with nose wheel steering actuated. With nose wheel steering OFF, nose wheel steer-damper unit acts as a shimmy damper. For ground handling, a pip pin connecting the steering mechanism to the lower section of the strut can be removed without the aid of tools, allowing nose gear to be rotated 360°.

3. Operation

A. Electrical

Nose wheel steering solenoid valve is energized through the on-ground position of nose wheel nutcracker switch, down position of nose wheel down lock switch, closed, on-ground position of both main gear nutcracker switches and the nose wheel steering switch. When the latter is set to ON by the pilot, power from the main dc bus through the S ampere nose wheel steering circuit breaker completes the circuit. (See Figure 1) A steering relay, wired in parallel with the nose gear nutcracker switch, energizes once the nose gear nutcracker switch closes on the ground. It provides a parallel circuit to keep nose wheel solenoid valve energized in event of a high strut opening and closing nose gear nutcracker switch while taxiing. A holding circuit in the steering relay bypasses nose wheel nutcracker switch.

NOTE: For electrical current to reach the nose wheel steering solenoid valve, three conditions must exist: both main gear on the ground, nose gear on the ground and nose gear down and locked. When these conditions of safety are met, electrical circuitry is completed and the solenoid valve may be energized by nose wheel POWER STEERING switch.

B. Hydraulic

With nose wheel steering solenoid valve energized, main (normal) hydraulic pressure enters the filter, solenoid valve, nose gear steer swivel and steer-damper unit. Aircraft is now steerable, by operating the nose wheel steering control tiller, through a steering arc of approximately 60° left and 60° right.

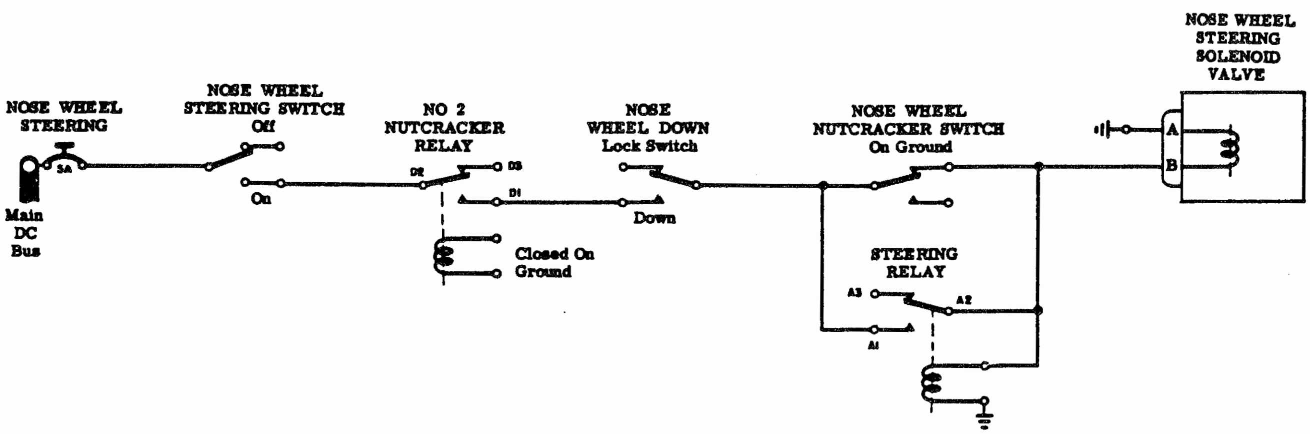
NOTE: Nose wheel steering system receives hydraulic pressure from main hydraulic system only. Auxiliary hydraulic system does not affect nose wheel steering system. With nose wheel POWER STEERING switch in OFF, no hydraulic pressure will enter nose wheel steering unit. In this mode, steer-damper unit will act as a shimmy damper.

Pressure is now up to the slide valve in the steer damper unit. The slide valve is operated by the motion of the nose wheel steering control tiller. Moving the valve to the left will permit hydraulic pressure to enter the rod end of the left cylinder, and head end of right cylinder. The left cylinder retracts, right cylinder extends, thereby steering aircraft to right. A follow-up mechanism, which is operated by the motion of the strut, will neutralize the valve. To steer aircraft to left, fluid flow is reversed.

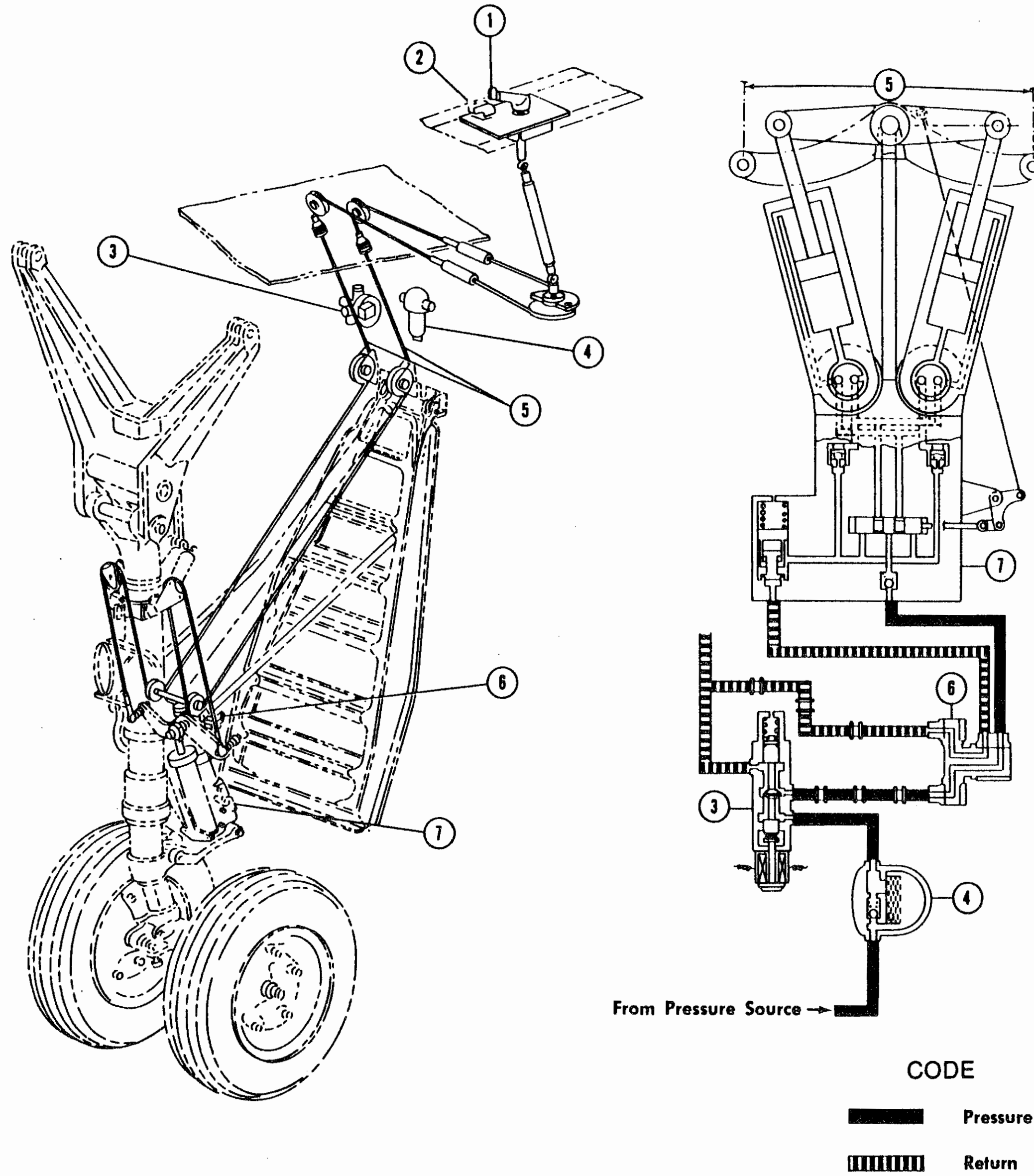
C. Towing

When the aircraft is to be towed, pip pin is removed from nose gear. This action frees the steering mechanism from lower section of nose gear strut and allows nose gear to swivel through a full 360°. The pip pin is removed by pressing the plunger on the pip pin to release ball locks, then pulling it out to disconnect the steering mechanism. With the pip pin removed, the tow bar, Gulfstream Aerospace part no. 159GT1049 is coupled to the nose gear wheel axis.

CAUTION: IT IS RECOMMENDED THAT THE STEERING UNIT BE DISCONNECTED FOR TOWING PURPOSES SINCE DAMAGE TO THE UNIT MAY RESULT.

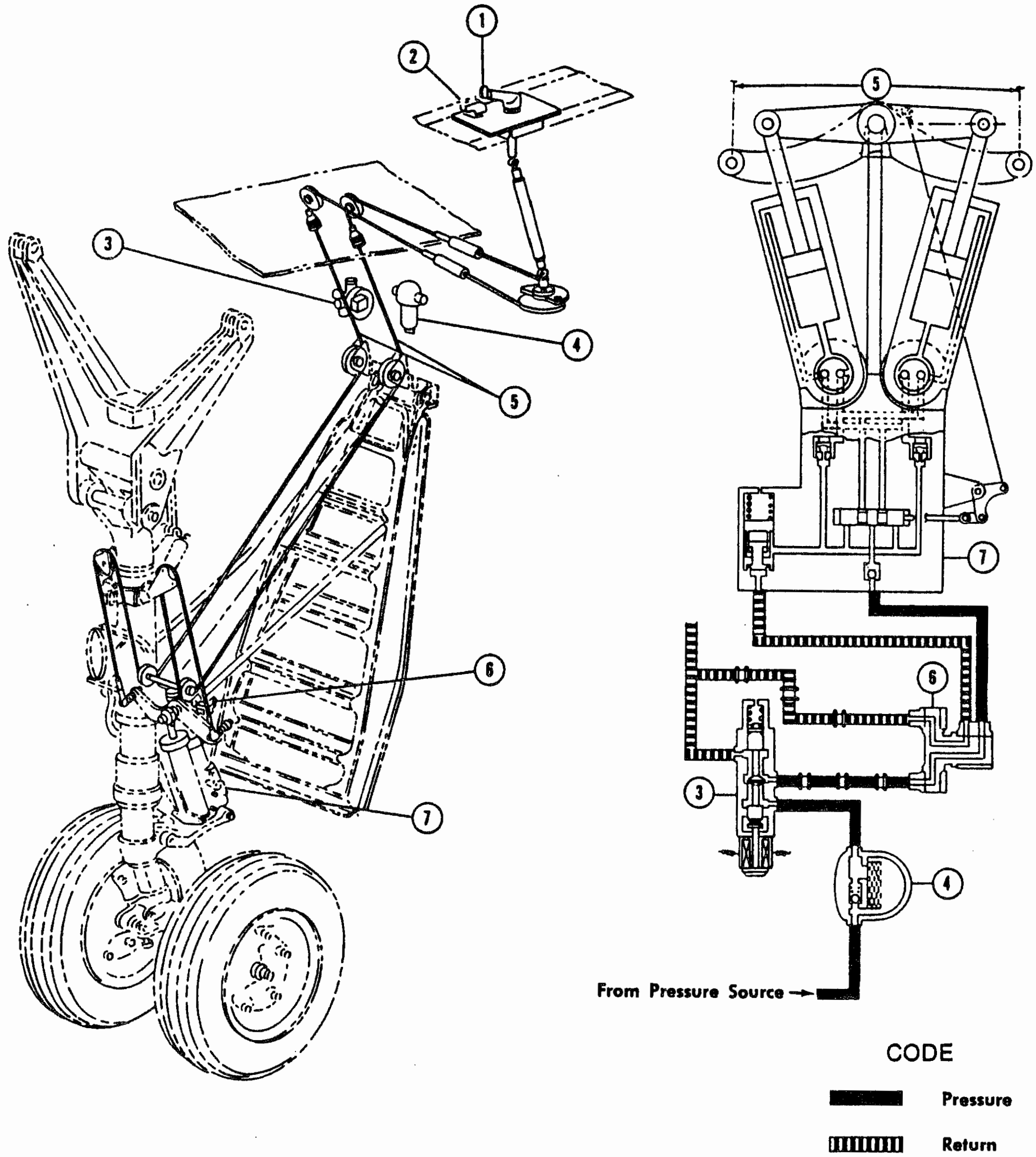


Nose Wheel Steering System — Schematic
Figure 1.



1. Nose Wheel Steering Control
2. Nose Wheel Power Steering Switch
3. Nose Wheel Steering Solenoid Valve
4. Filter
5. Control Cables
6. Steer Swivel
7. Steer Damper Unit

Nose Wheel Steering System
Figure 2



1. Nose Wheel Steering Control
2. Nose Wheel Power Steering Switch
3. Nose Wheel Steering Solenoid Valve
4. Filter
5. Control Cables
6. Steer Swivel
7. Steer Damper Unit

Nose Wheel Steering System
Figure 2.

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NOSE WHEEL STEERING SYSTEM — MAINTENANCE PRACTICES

1. **Nose Wheel Steering — Adjustment**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Jack aircraft and remove nose wheels and collars (Bendix 171529).
B. Attach nose wheel steering centering jig (159-10000-1 I.F.). Push tubes fully into axle. The bar should extend forward.
C. Drop a plumb bob from the leveling bracket located on the center line of the front bulkhead. The string must be attached so that it drops from the right of the leveling bracket.
D. Release all nitrogen from strut, and compress strut approximately 6 inches so that it is out of its centering cam. (Maintain in this position with pin attached to nose strut jack point supplied with 159-10000-1 I.F.)
E. Disconnect bottom bolt of adjustment rod connected to bellcrank (to which steering cables are attached).
F. Move strut until plumb bob lines up with center line on bar.
G. If bolt disconnected in Step E. above does not line up for installation, break safety-wire on adjusting turnbuckle and adjust so that rod lines up with bellcrank. Connect rod. Safety-wire adjusting turnbuckle. This step is to be accomplished with steering power on.
H. Adjust nose wheel steering cables to a tension of 20 $+0$ -5 pounds per cable.
I. Remove nose wheel steering centering jig.
J. Install collars (Bendix 171529) and nose wheels.
K. Inflate strut.
L. With nose strut in a static position (3 inches from fully compressed) and nose wheel steering control tiller against its LEFT or RIGHT stop, fixed stops in nose wheel steering control should limit nose wheel travel to 57° ± 5° left or right.

2. **Steering Torque Link Boot — Replacement / Lubrication**

CAUTION: ENSURE THAT NOSE WHEEL DOOR GROUND SAFETY LOCKS (159GT1007 AND 159GT1011) ARE IN PLACE BEFORE PROCEEDING.

A. Separate steering unit from lower torque arm by removing pip-pin.
B. Retain pip-pin.
C. Partially unscrew socket.
D. Remove and discard knee boot.
E. Apply Hi-Lo MS #1 grease to ball located on lower portion of steering unit and to sockets located in lower torque arm.

CAUTION: CARE MUST BE TAKEN TO PREVENT CLEANING SOLVENTS FROM CONTACTING RUBBER BOOT.

F. Slip center hole of new knee boot over greased ball, and slip end holes of boot over greased sockets.
G. Insert ball between bearings, and tighten socket loosened in Step C. above as follows:
   (1) Torque socket to 200 - 250 inch-pounds.
   (2) Back off socket until ball is loose.
   (3) Hand tighten socket until ball binds.
   (4) Back off socket to first adjusting lockpin hole.
   (5) Inspect and adjust at 10 hours and thereafter, at each regular inspection.
H. Install pip-pin removed in Step A. above.

3. **Nose Wheel Shimmy — Adjustments**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. If the nose landing gear has a tendency to shimmy, proceed as follows:
   (1) Jack nose gear until strut is fully extended and both wheels clear ground.
   (2) Check wheels for proper balance and tires for correct inflation.
   (3) Remove all shims between fuselage bushings and nose gear yoke. Check that total end play at each side does not exceed 0.024 inch. If this limit is exceeded, install new fuselage bushings. Use special tool 159GT1055 for removal and installation of bushings.
   (4) Check torque arm ball joint adjustment as directed in Bendix Overhaul Manual, Section 32-50-1.
   (5) Inspect nose wheel bearings for damage or wear and replace if necessary.

B. If the preceeding procedure fails to correct the shimmy tendencies, refer to Bendix Overhaul Manual for additional tests before removing nose wheel steering unit for close inspection and possible overhaul.

4. **Nose Wheel Steering Unit — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

   (1) Connect hydraulic test unit to ground test connections in right nacelle.

   CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND SLOWLY REMOVE RESERVOIR FILLER CAP TO BLEED RESIDUAL PRESSURE FROM RESERVOIR BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS.

   (2) Remove electrical wiring from steering unit.
   (3) Remove pip-pin to disconnect steering unit from lower torque arm.
   (4) Open bleeder plug on each of the steering unit actuating cylinders to bleed pressure trapped in unit.
   (5) Disconnect hydraulic lines and remove fittings from steering unit and cap openings.
   (6) Remove bolts securing hydraulic swivel fitting to actuator lever pin and set swivel fitting aside.
   (7) Disconnect steering cables from actuator lever assembly and retain hardware.
   (8) Remove bolts securing piston rod ends to trunnion pin assembly and retain.
   (9) Disconnect push-pull rod between unit body and actuator lever and retain.
   (10) Remove pin securing lower torque arm to piston assembly and remove lower torque arm.
   (11) Remove nut and washer from actuator lever pin.
   (12) Remove actuator lever pin supporting steering unit at same time.
   (13) Remove steering unit and actuator lever.

   NOTE: Remove lube fitting from trunnion pin assembly before removing trunnion pin assembly.

   (14) Remove trunnion pin assembly and spacer.

B. Installation
   (1) With replacement spacer in position between mounting lugs on outer cylinder, install trunnion pin assembly.
   (2) Align holes in spacer with holes in trunnion pin assembly.
   (3) Place steering unit and actuator crank in position and install actuator lever pin, washers and nut.
   (4) Install lube fittings in trunnion pin assembly.
   (5) With replacement, lower torque arm in position on piston, install pin and secure.
   (6) Install push-pull rod between unit body and actuator crank.
   (7) Connect piston rod ends to replacement trunnion pin assembly.
   (8) Connect steering cables to actuator lever assembly.
   (9) Place hydraulic swivel in position on lever pin and install mounting hardware.
   (10) Install fittings and connect hydraulic lines to steering unit.
   (11) Connect electrical wiring to steering unit.

NOTE: Steps (12) and (13) below are to be accomplished only if cable tension changes during steering unit change.

(12) Install rig pins (neutral pin in unit and one pin in sector at Fuselage Station 110 left side with no hydraulic power on aircraft).
(13) Gain access through left check panel and tension steering cables to 20 + 0 - 5 pounds per cable. Safety turnbuckles.
(14) Perform Nose Wheel Steering — Operational Test.
(15) Remove hydraulic test unit from right nacelle and secure nacelle.
(16) Remove ground safety devices.

5. **Nose Wheel Steering Filter Element — Removal / Inspection / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Locate filter housing in nose wheel well, aft bulkhead.
(2) Remove safety-wire and remove filter bowl from housing.
(3) Remove filter element, discard element and seals.

CAUTION: IF THERE IS EVIDENCE OF METAL PARTICLES IN FILTER BOWL, INVESTIGATE FURTHER FOR POSSIBLE COMPONENT FAILURE.

(4) Inspect filter element.

B. Installation

(1) Install new filter element with new O-rings.
(2) Install new O-ring on filter bowl.
(3) Install filter bowl and safety-wire to housing.
(4) Pressurize main hydraulic system and check for leakage.

6. **Nose Wheel Steering Selector Valve — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Pull and placard NOSE STEER circuit breaker.
(2) Place a suitable container with a minimum capacity of five gallons under selector valve.

NOTE: Nose wheel steering selector valve is located on aft bulkhead of nose wheel well.

(3) Connect a portable hydraulic pressure unit to ground test connections in right engine nacelle.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO ALLOW PRESSURE TO BLEED FROM RESERVOIR. BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS.

(4) Disconnect lines and remove fittings from valve. Remove electrical connector.
(5) Remove mounting bolts and remove valve from aircraft.

B. Installation

(1) Place valve on a clean work area and install fittings using new O-rings.
(2) Place valve in position and install mount bolts.
(3) Connect hydraulic lines and electrical connector to valve.
(4) Pressurize main hydraulic system.
(5) Depress NOSE STEER circuit breaker. Clear area around nose wheels.
(6) From pilot seat in cockpit, operate tiller bar noting that wheel movement coincides with tiller bar movement. Cycle four or five times. Check for leaks.

(7) Remove hydraulic test unit from right nacelle and secure nacelle.
(8) Remove ground safety devices.

7. **Nose Wheel Steering Swivel — Removal / Installation**
A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL. ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Remove hydraulic power from aircraft.
(2) Pull NOSE STEER circuit breaker.
(3) Disconnect flexible hydraulic lines from swivel and remove two rigid lines between swivel and steering unit.
(4) Remove hardware securing swivel to steering assembly.
(5) Remove swivel.

B. Installation
(1) Position swivel on steering assembly and secure with hardware.

CAUTION: USING HAND PUMP, ENSURE THAT SWIVEL IS NOT CONNECTED TO VALVE 180° OUT OF POSITION.

(2) Disconnect hand pump.
(3) Connect flexible hydraulic lines to swivel and install two rigid lines between swivel and steering assembly.
(4) Apply hydraulic and electrical power to aircraft.
(5) Depress NOSE STEER circuit breaker.
(6) Operate steering tiller in both directions and check nose wheels travel fully in both directions; return tiller to neutral position.
(7) Check for hydraulic fluid leakage at fittings.

8. **Steering Pin — Removal / Installation**
A. Removal

NOTE: Pliers, levers, etc. must not be used for removal of pin.

(1) Remove pip-pin safety clip.
(2) Depress pip-pin button while removing pip-pin by hand.
(3) Raise steering unit.
(4) Insert pip-pin through hole in trunnion pin assembly and in lug on strut assembly.

NOTE: This prevents damage to steering assembly during towing.

B. Installation

NOTE: Bolt NAS1104-22, nut MS20365-428, and two washers AN960-416L can be used in place of the pip-pin. No other combination can be used. Replace the nut at each installation.

(1) Inspect pip-pin for proper functioning of plunger and locking balls as follows:
  (a) Ensure balls work freely when plunger is depressed.
  (b) Ensure balls are immovable when plunger is released.
  (c) Discard pins which fail this inspection.
  (d) Discard any pins with a hole 0.050 dia. visible in the end of the plunger.
(2) Install pin.
(3) Inspect for position of locking balls by attempting to withdraw pin without depressing plunger.
(4) Install pip-pin safety clip.

9. **Nose Wheel Steering — Operational Test**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Jack nose gear so that centering cam is fully engaged.

NOTE: Nose wheel drag link must be connected and strut serviced with fluid and air.

B. Attach pointer of steering tool to uplock roller bolt lugs.
C. Clamp protractor to outer cylinder approximately two inches above gland nut.
D. Place grease plate under wheels.
E. Lower nose until pointer almost touches protractor.
F. Energize aircraft electrical system and pressurize hydraulic system.

NOTE: Main gear nutcracker switches must be unmade (ground configuration).

G. Place nose wheel steering switch on.
H. Rotate tiller in cockpit full left, wheels should rotate 57° ± 2° left.
I. Release tiller, wheels should return to neutral ± 2°.
J. Rotate tiller in cockpit full right, wheels should rotate 57° ± 2° right.
K. Release tiller, wheels should return to neutral ± 2°.
L. Turn nose wheel steering switch off, nose wheels should remain in neutral.

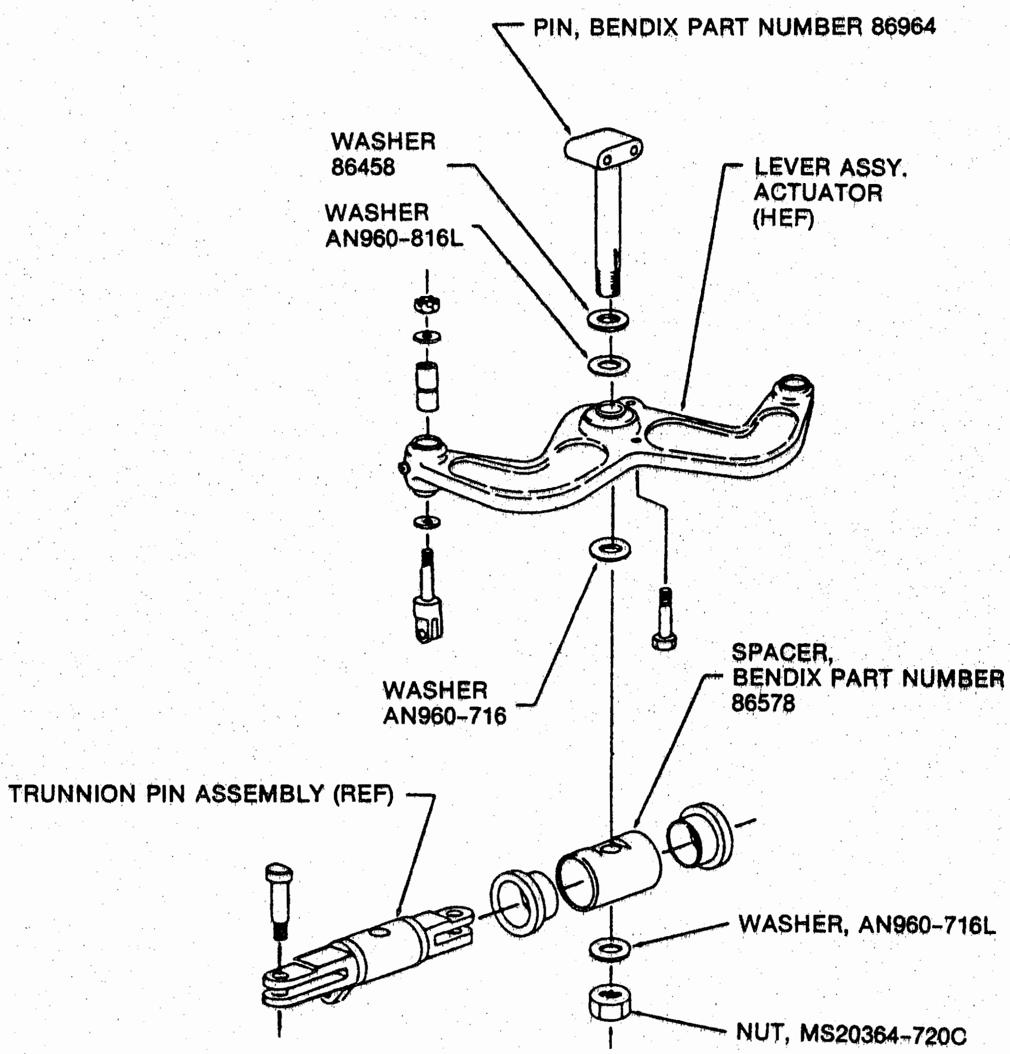
NOTE: If wheels do not rotate fully or return fully to neutral, operate steering with steering (mustache) bar on unit. Grease plate may not eliminate enough friction and cable travel will be lost through bungees in cable system.

M. Shut off electrical and hydraulic power.
N. Raise nose, remove grease plate, lower nose and remove jack.

10. **Nose Gear Steering Lever Pin and Trunnion Spacer — Inspection (NDT)**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Gain access to nose wheel well.
B. Remove pip pin to disconnect steering unit from torque arm.
C. Remove bolts attaching steering actuators to trunnion pin assembly.
D. Remove attaching hardware holding hydraulic swivel above actuator lever pin.
E. Remove nut and washer from actuator lever pin and remove pin and trunnion spacer assembly. See Figure 201.
F. Inspect actuator lever pin and trunnion spacer assembly for excessive fretting corrosion on pin and galling on spacer. Replace pin within 100 hours if pitting and gouging is readily discernible. Scratches less than 0.001 inches are acceptable.
G. Lubricate steering lever pin with MIL-G-21164 and install pin and spacer, torque nut to 270-300 inch-pounds. See Figure 201.
H. Attach hydraulic swivel.
I. Attach steering actuators to trunnion pin.
J. Install pip pin attaching steering unit to torque arm.
K. Perform operational test on nose steering system and check for leakage of components.



Nose Gear Steering Pin and Trunnion Spacer
Figure 201

NOSE WHEEL STEERING SYSTEM — MAINTENANCE PRACTICES

1. Nose Wheel Steering — Adjustment

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Jack aircraft and remove nose wheels and collars (Bendix 171529).

B. Attach nose wheel steering centering jig (159-10000-1 I.F.). Push tubes fully into axle. The bar should extend forward.

C. Drop a plumb bob from the leveling bracket located on the center line of the front bulkhead. The string must be attached so that it drops from the right of the leveling bracket.

D. Release all nitrogen from strut, and compress strut approximately 6 inches so that it is out of its centering cam. (Maintain in this position with pin attached to nose strut jack point supplied with 159-10000-1 I.F.)

E. Disconnect bottom bolt of adjustment rod connected to bellcrank (to which steering cables are attached).

F. Move strut until plumb bob lines up with center line on bar.

G. If bolt disconnected in Step E. above does not line up for installation, break safety-wire on adjusting turnbuckle and adjust so that rod lines up with bellcrank. Connect rod. Safety-wire adjusting turnbuckle. This step is to be accomplished with steering power on.

H. Adjust nose wheel steering cables to a tension of 20 $+0$ -5 pounds per cable.

I. Remove nose wheel steering centering jig.

J. Install collars (Bendix 171529) and nose wheels.

K. Inflate strut.

L. With nose strut in a static position (3 inches from fully compressed) and nose wheel steering control tiller against its LEFT or RIGHT stop, fixed stops in nose wheel steering control should limit nose wheel travel to $57° \pm 5°$ left or right.

2. Nose Wheel Steering Torque Link Boot — Replacement / Lubrication

CAUTION: ENSURE THAT NOSE WHEEL DOOR GROUND SAFETY LOCKS (159GT1007 AND 159GT1011) ARE IN PLACE BEFORE PROCEEDING.

A. Separate steering unit from lower torque arm by removing pip pin.

B. Retain pip pin.

C. Partially unscrew socket.

D. Remove and discard knee boot.

E. Apply Hi-Lo MS #1 grease to ball located on lower portion of steering unit and to sockets located in lower torque arm.

CAUTION: CARE MUST BE TAKEN TO PREVENT CLEANING SOLVENTS FROM CONTACTING RUBBER BOOT.

F. Slip center hole of new knee boot over greased ball, and slip end holes of boot over greased sockets.

G. Insert ball between bearings, and tighten socket loosened in Step C. above as follows:

(1) Torque socket to 200-250 inch-pounds.

(2) Back off socket until ball is loose.

(3) Hand tighten socket until ball binds.

(4) Back off socket to first adjusting lockpin hole.

(5) Inspect and adjust at 10 hours and thereafter, at each regular inspection.

H. Install pip pin removed in Step A. above.

3. Nose Wheel Shimmy — Adjustments

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION iS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COUi.D OCCUR.

A. If the nose landing gear has a tendency to shimmy, proceed as follows:

(1) Jack nose gear until strut is fully extended and both wheels clear ground.

(2) Check wheels for proper balance and tires for correct inflation.

(3) Remove all shims between fuselage bushings and nose gear yoke. Check that total end play at each side does not exceed 0.024 inch. If this limit is exceeded, install new fuselage bushings. Use special tool 159GT1055 for removal and installation of bushings.

(4) Check torque arm ball joint adjustment as directed in Bendix Overhaul Manual, Section 32-50-1.

(5) Inspect nose wheel bearings for damage or wear and replace if necessary.

B. If the preceding procedure fails to correct the shimmy tendencies, refer to Bendix Overhaul Manual for additional tests before removing nose wheel steering unit for close inspection and possible overhaul.

4. Nose Wheel Steering Unit — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Connect hydraulic test unit to ground test connections in right nacelle.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND SLOWLY REMOVE RESERVOIR FILLER CAP TO BLEED RESIDUAL PRESSURE FROM RESERVOIR BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS.

(2) Remove electrical wiring from steering unit.

(3) Remove pip pin to disconnect steering unit from lower torque arm.

(4) Open bleeder plug on each of the steering unit actuating cylinders to bleed pressure trapped in unit.

(5) Disconnect hydraulic lines and remove fittings from steering unit and cap openings.

(6) Remove bolts securing hydraulic swivel fitting to actuator lever pin and set swivel fitting aside.

(7) Disconnect steering cables from actuator lever assembly and retain hardware.

(8) Remove bolts securing piston rod ends to trunnion pin assembly and retain.

(9) Disconnect push-pull rod between unit body and actuator lever and retain.

(10) Remove pin securing lower torque arm to piston assembly and remove lower torque arm.

(11) Remove nut and washer from actuator lever pin.

(12) Remove actuator lever pin supporting steering unit at same time.

(13) Remove steering unit and actuator lever.

NOTE: Remove lube fitting from trunnion pin assembly before removing trunnion pin assembly.

(14) Remove trunnion pin assembly and spacer.

B. Installation

(1) With replacement spacer in position between mounting lugs on outer cylinder, install trunnion pin assembly.

(2) Align holes in spacer with holes in trunnion pin assembly.

(3) Place steering unit and actuator crank in position and install actuator lever pin, washers and nut.

(4) Install lube fittings in trunnion pin assembly.

(5) With replacement, lower torque arm in position on piston, install pin and secure.

(6) Install push-pull rod between unit body and actuator crank.

(7) Connect piston rod ends to replacement trunnion pin assembly.

(8) Connect steering cables to actuator lever assembly.

(9) Place hydraulic swivel in position on lever pin and install mounting hardware.

(10) Install fittings and connect hydraulic lines to steering unit.

(11) Connect electrical wiring to steering unit.

NOTE: Steps (12) and (13) below are to be accomplished only if cable tension changes during steering unit change.

(12) Install rig pins (neutral pin in unit and one pin in sector at Fuselage Station 110 left side with no hydraulic power on aircraft).

(13) Gain access through left check panel and tension steering cables to 20 +0 −5 pounds per cable. Safety turnbuckles.

(14) Perform Nose Wheel Steering — Operational Test, see this Section.

(15) Remove hydraulic test unit from right nacelle and secure nacelle.

(16) Remove ground safety devices.

5. Nose Wheel Steering Filter Element — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Locate filter housing in nose wheel well, aft bulkhead.

(2) Remove safety-wire and remove filter bowl from housing.

(3) Remove filter element, discard element and seals.

CAUTION: IF THERE IS EVIDENCE OF METAL PARTICLES IN FILTER BOWL, INVESTIGATE FURTHER FOR POSSIBLE COMPONENT FAILURE.

(4) Inspect filter element.

B. Installation

(1) Install new filter element with new O-rings.

(2) Install new O-ring on filter bowl.

(3) Install filter bowl and safety-wire to housing.

(4) Pressurize main hydraulic system and check for leakage.

6. Nose Wheel Steering Selector Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Pull and placard NOSE STEER circuit breaker.

(2) Place a suitable container with a minimum capacity of five gallons under selector valve.

NOTE: Nose wheel steering selector valve is located on aft bulkhead of nose wheel well.

(3) Connect a portable hydraulic pressure unit to ground test connections in right engine nacelle.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO ALLOW PRESSURE TO BLEED FROM RESERVOIR. BEFORE LOOSENING ANY HYDRAULIC LINES OR FITTINGS.

(4) Disconnect lines and remove fittings from valve. Remove electrical connector.

(5) Remove mounting bolts and remove valve from aircraft.

B. Installation

(1) Place valve on a clean work area and install fittings using new O-rings.

(2) Place valve in position and install mount bolts.

(3) Connect hydraulic lines and electrical connector to valve.

(4) Pressurize main hydraulic system.

(5) Depress NOSE STEER circuit breaker. Clear area around nose wheels.

(6) From pilot seat in cockpit, operate tiller bar noting that wheel movement coincides with tiller bar movement. Cycle four or five times. Check for leaks.

(7) Remove hydraulic test unit from right nacelle and secure nacelle.

(8) Remove ground safety devices.

7. Nose Wheel Steering Swivel — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL. ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Remove hydraulic power from aircraft.

(2) Pull NOSE STEER circuit breaker.

(3) Disconnect flexible hydraulic lines from swivel and rigid lines between swivel and steering unit.

(4) Remove hardware securing swivel to steering assembly.

(5) Remove swivel.

B. Installation

(1) Position swivel on steering assembly and secure with hardware.

CAUTION: USING HAND PUMP, ENSURE THAT SWIVEL IS NOT CONNECTED TO VALVE 1800 OUT OF POSITION.

(2) Disconnect hand pump.

(3) Connect flexible hydraulic lines to swivel and rigid lines between swivel and steering assembly.

(4) Apply hydraulic and electrical power to aircraft.

(5) Depress NOSE STEER circuit breaker.

(6) Operate steering tiller in both directions and check nose wheels travel fully in both directions; return tiller to neutral position.

(7) Check for hydraulic fluid leakage at fittings.

8. Nose Wheel Steering Pin — Removal / Installation

A. Removal

NOTE: Pliers, levers, etc. must not be used for removal of pin.

(1) Remove pip pin safety clip.

(2) Depress pip pin button while removing pip pin by hand.

(3) Raise steering unit.

(4) Insert pip pin through hole in trunnion pin assembly and in lug on strut assembly.

NOTE: This prevents damage to steering assembly during towing.

B. Installation

NOTE: Bolt NAS 1104-22, nut MS20365-428, and two washers AN960-416L can be used in place of the pip pin. No other combination can be used. Replace the nut at each installation.

(1) Inspect pip pin for proper functioning of plunger and locking balls as follows:

(a) Ensure balls work freely when plunger is depressed.

(b) Ensure balls are immovable when plunger is released.

(c) Discard pins which fail this inspection.

(d) Discard any pins with a hole 0.050 dia. visible in the end of the plunger.

(2) Install pin.

(3) Inspect for position of locking balls by attempting to withdraw pin without depressing plunger.

(4) Install pip pin safety clip.

9. Nose Wheel Steering — Operational Test

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Jack nose gear so that centering cam is fully engaged.

NOTE: Nose wheel drag link must be connected and strut serviced with fluid and air.

B. Attach pointer of steering tool to uplock roller bolt lugs.

C. Clamp protractor to outer cylinder approximately two inches above gland nut.

D. Place grease plate under wheels.

E. Lower nose until pointer almost touches protractor.

F. Energize aircraft electrical system and pressurize hydraulic system.

NOTE: Main gear nutcracker switches must be unmade (ground configuration).

G. Place nose wheel steering switch on.

H. Rotate tiller in cockpit full left, wheels should rotate 57° ±2° left.

I. Release tiller, wheels should return to neutral ±2°

J. Rotate tiller in cockpit full right, wheels should rotate 57° ±2° right.

K. Release tiller, wheels should return to neutral ± 2°

L. Turn nose wheel steering switch off, nose wheels should remain in neutral.

NOTE: If wheels do not rotate fully or return fully to neutral, operate steering with steering (moustache) bar on unit. Grease plate may not eliminate enough friction and cable travel will be lost through bungees in cable system.

M. Shut off electrical and hydraulic power.

N. Raise nose, remove grease plate, lower nose and remove jack.

10. Nose Wheel Steering Pin — NDT Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Gain access to nose wheel well.

B. Remove pip pin to disconnect steering unit from torque arm.

C. Remove bolts attaching steering actuators to trunnion pin assembly.

D. Remove attaching hardware holding hydraulic swivel above actuator lever pin.

E. Remove nut and washer from actuator lever pin and remove pin and trunnion spacer assembly. See Figure 201.

F. Inspect actuator lever pin and trunnion spacer assembly for excessive fretting corrosion on pin and galling on spacer. Replace pin within 100 hours if pitting and gouging is readily discernible. Scratches less than 0.001 inches are acceptable.

G. Lubricate steering lever pin with MIL-G-21164 and install pin and spacer, torque nut to 270-300 inch-pounds. See Figure 201.

H. Attach hydraulic swivel.

I. Attach steering actuators to trunnion pin.

J. Install pip pin attaching steering unit to torque arm.

K. Perform operational test on nose steering system and check for leakage of components.

11. Nose Gear Steering Lug Lower Pin — Inspection (NDT)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove pip pin to disconnect steering unit from torque arm.

B. Remove nut and washer and supporting torque link, remove pin assembly. (See Figure 201)

C. Remove grease fitting from pin.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

D. Clean pin and remove paint around 0.06 inch diameter grease supply hole and head of pin assembly.

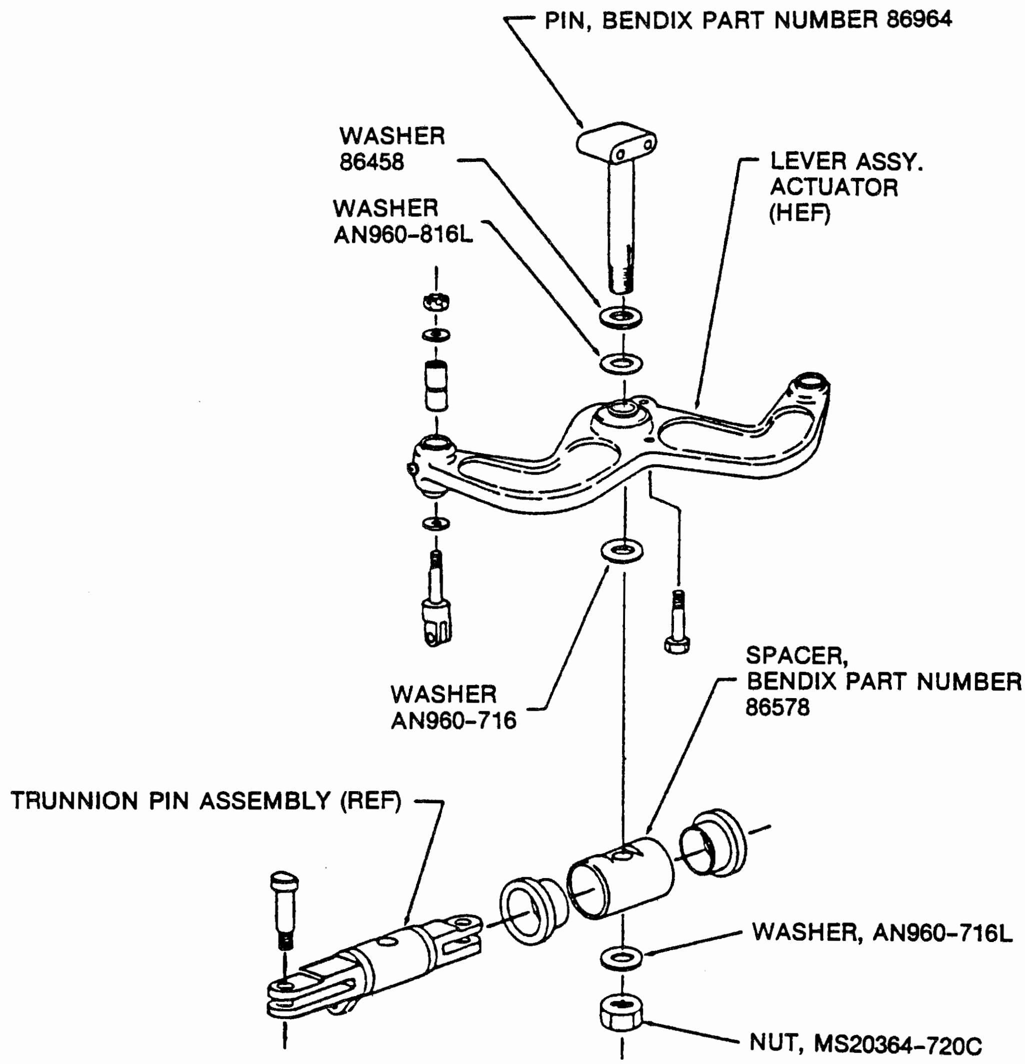
E. Inspect pin for radial cracks originating at the grease supply hole using visual and magnetic particle or equivalent (NDT) method.

F. Treat and refinish pin if required.

G. Install grease fitting in pin and treat pin with lubricant (MIL-G-21164) prior to installation.

H. Install pin using retained hardware. Grease pin with MIL-G-23827A.

I. Install pip pin connecting steering unit to torque arm.



Nose Gear Steering Pin and Trunnion Spacer
Figure 201.

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WHEEL BRAKE SYSTEM — DESCRIPTION / OPERATION

1. **Description**

Each main wheel contains a single disc brake consisting of a housing assembly, three piston assemblies, three grip and pilot pin assemblies, six brake lining segments, two bleeder valves, two bleeder screws, two bleeder adapters, O-rings and packings. The hydraulic components for brake system are: two power brake valves (located on cockpit forward bulkhead), a decelostat unit at the brake, six shuttle valves (four attached to wheel brake units and two attached in nose wheel well), a handle placarded, PARK / EMER BRAKE (located on pilots side of center pedestal), a parking and emergency brake selector valve (located in nose wheel well under the cockpit floor), an air-oil accumulator and an air filler valve (located on left side of nose wheel well), an air gage (located in the cockpit), an auxiliary pressure check valve (located on right side of nose wheel well), a check valve (located next to parking and emergency brake selector valve) and a restrictor (located next to parking and emergency brake selector valve). (See Figure 1.) Aircraft 88 - 200 including 322, 323 excluding 114 and aircraft having ASC 137 contain, a pressure relief valve and a pressure operated shutoff valve in the parking and emergency brake system.

The main hydraulic system provides pressure for the normal operation of the brakes through the normal side of the power brake valves. If main system pressure fails, auxiliary system must be turned on. This pressure is supplied through auxiliary side of the power brake valves. Pulling the PARK / EMER BRAKE handle aft causes stored accumulator pressure or auxiliary pump pressure to be released to wheel brakes. The pilots and copilots brake pedals are interconnected. Three self-adjusting pins in each brake compensate automatically to maintain a running clearance of 0.014 - 0.039 inch between the disc and the flat surface of the housing.

2. **Operation**

A. Normal Brake System

The wheel brakes are applied by either pilot or copilot by depressing rudder-brake pedals. The power brake valves are dual units, each containing a normal and an emergency unit in one assembly. The normal and emergency portions operate in unison. Since only main hydraulic system pressure is applied, auxiliary side of the power brake valve is inactive. If the auxiliary pump is running, normal system precedence is built into the valves. Depressing the rudder-brake pedals forces open the valve and the fluid return is shut off. Hydraulic fluid pressure builds up in proportion to the applied force on the pedals.

Fluid from the power brake valves, is directed through the decelostats (antiskid controllers when installed) to the individual brakes. The decelostats permit full application of the wheel brakes, under normal operating conditions, without the danger of skidding the main gear wheels. If during brake application, a wheel suddenly decelerates at a rate that would cause it to skid, the decelostat releases the brake by opening its exhaust valve to exhaust hydraulic fluid from the brake unit and, at the same time, shuts off inlet pressure. This removes the braking effort causing the tendency to skid. When the wheel regains its normal speed, the decelostat reapplies the brake by allowing hydraulic fluid from the brake valve to again flow to the brake unit.

CAUTION: IF ONE OR MORE DECELOSTATS ARE INOPERATIVE OR IF THE AUXILIARY PUMP IS OPERATING WHICH RENDERS THE DECELOSTATS INEFFECTIVE, BRAKES SHOULD BE APPLIED AS IF NO ANTISKID PROTECTION IS INSTALLED. REFER TO LATEST GULFSTREAM FLIGHT MANUAL FOR TAKEOFF AND LANDING RESTRICTIONS.

NOTE: An indicator pin is located on the top forward portion of the decelostat body. If the pin is visible above the body surface the unit is jammed. This must be checked on every preflight inspection.

B. Auxiliary Brake System

If the main hydraulic system fails, wheel brakes can be applied by means of pressure from the auxiliary hydraulic system. Auxiliary hydraulic system pressure passes through the check valve to the auxiliary sides of brake valves. Pressing the rudder pedals directs auxiliary hydraulic pressure through the two shuttle valves. In this operation, the normal side of the power brake valve is inactive. The brake system shuttle valves shift, directing hydraulic pressure to the shuttle valves at the wheel brake units. These shuttle valves also shift, directing pressure to the three pucks in each of the wheel brake units.

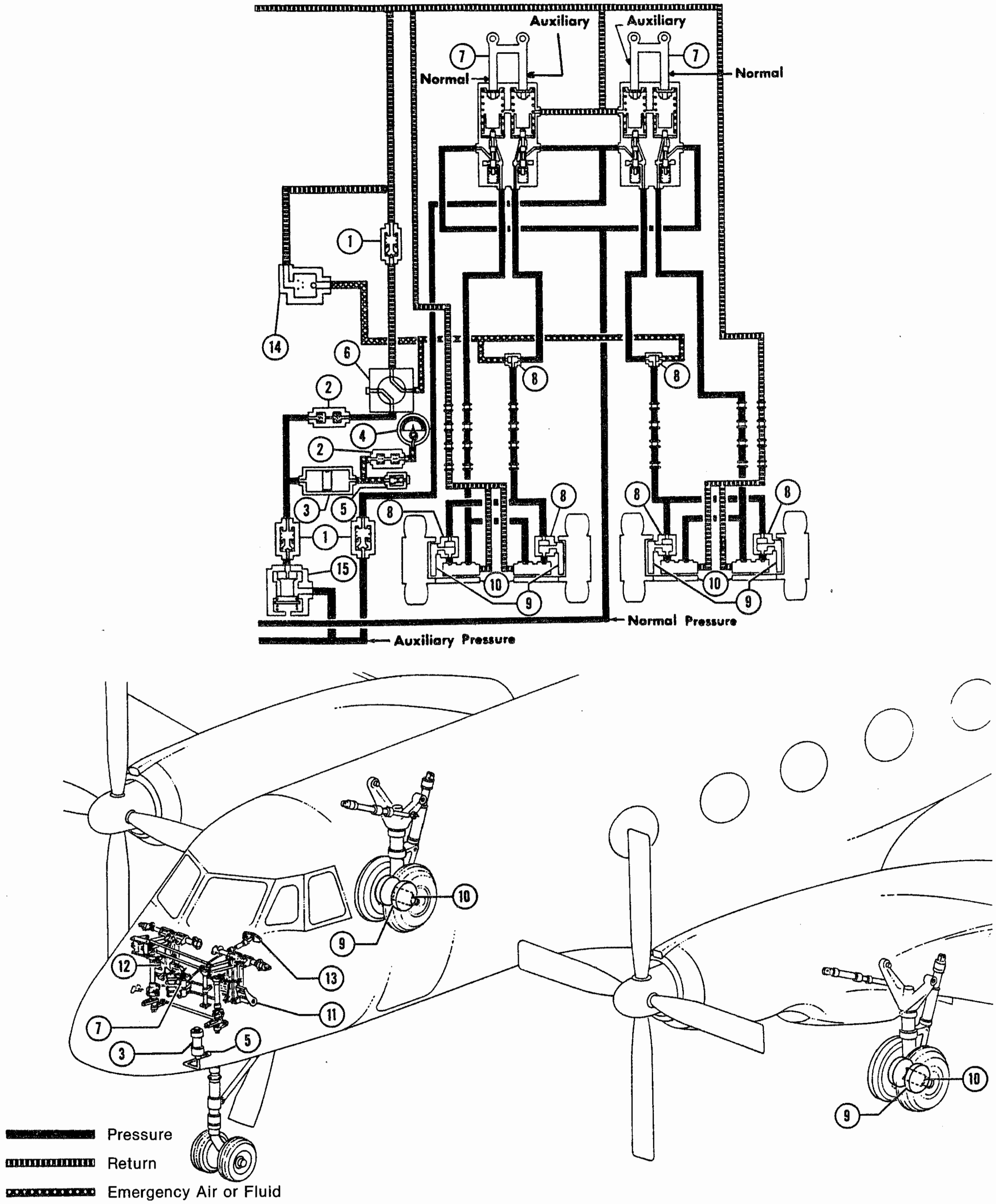
C. Parking and Emergency Brake System

The parking and emergency brake system is used for two purposes: parking the aircraft and for emergency stopping. Pulling the T-handle aft causes auxiliary hydraulic pressure to enter the wheel brakes. Pressure is directed through a check valve, a restrictor and the parking and emergency brake selector valve to the parking and emergency brake system. If a failure prevents auxiliary hydraulic pressure from entering wheel brakes, stored accumulator hydraulic fluid pressure is applied through the same restrictor and selector valve to wheel brakes. Returning the T-handle to forward position releases wheel brakes. After an emergency stop by use of emergency brakes, the system does not have to be bled, as hydraulic fluid was used in emergency operation.

On aircraft 88 - 200 including 322 and 323, excluding 114, and aircraft having ASC 137 a thermal relief valve is incorporated between shuttle valves and return path to reservoir. The thermal relief valve prevents the buildup of excessive pressure due to thermal expansion when PARK / EMER BRAKE is set to ON. Relief valve is set to release at a crack pressure of 1650 psi and a reset pressure of 1450 psi. In addition, a pressure operated shutoff vlave is incorporated to limit supply pressure from the auxiliary hydraulic pump entering the parking / emergency brake system. The valve closes between pressures of 1425 and 1500 psi.

WARNING: FIVE APPLICATIONS OF THE EMERGENCY AND PARKING BRAKE SYSTEM CAN BE MADE AFTER AUXILIARY PUMP SHUTDOWN.

NOTE: Do not set brakes while they are hot. Prevention of brake fluid leakage and longer seal life can be achieved if the brake seals are not held under pressure during the cooling period. When the brakes have cooled so that they can be touched by hand, the parking brake can be set regardless of local temperatures.



★Aircraft 88 – 200 including 322 and 323, excluding 114 and aircraft having ASC 137, only.

1. Check Valve
2. Restrictor
3. Accumulator
4. Accumulator Air Pressure Gage
5. Accumulator Air Filler Valve
6. Parking and Emergency Brake Selector Valve
7. Power Brake Valve
8. Shuttle Valve
9. Wheel Brake
10. Decelostat Assembly
11. Pilots Brake and Rudder Pedals
12. Copilots Brake and Rudder Pedals
13. Parking and Emergency Brake T-handle
★14. Pressure Relief Valve
★15. Hydraulic Shutoff Valve

Brake Control System
Figure 1

WHEEL BRAKE SYSTEM — DESCRIPTION / OPERATION

1. Description

Each main wheel contains a single disc brake consisting of a housing assembly, three piston assemblies, three grip and pilot pin assemblies, six brake lining segments, two bleeder valves, two bleeder screws, two bleeder adapters, O-rings and packings. The hydraulic components for brake system are: two power brake valves (located on cockpit forward bulkhead), a decelostat unit at the brake, six shuttle valves (four attached to wheel brake units and two attached in nose wheel well), a handle placarded, PARK/EMER BRAKE (located on pilots side of center pedestal), a parking and emergency brake selector valve (located in nose wheel well under the cockpit floor), an air-oil accumulator and an air filler valve (located on left side of nose wheel well), an air gage (located in the cockpit), an auxiliary pressure check valve (located on right side of nose wheel well), a check valve (located next to parking and emergency brake selector valve) and a restrictor (located next to parking and emergency brake selector valve). (See Figure 1) Aircraft 88 - 200, 322 and 323 excluding 114 and Aircraft having ASC 137 contain, a pressure relief valve and a pressure operated shutoff valve in the parking and emergency brake system.

The main hydraulic system provides pressure for the normal operation of the brakes through the normal side of the power brake valves. If main system pressure fails, auxiliary system must be turned on. This pressure is supplied through auxiliary side of the power brake valves. Pulling the PARK/EMER BRAKE handle aft causes stored accumulator pressure or auxiliary pump pressure to be released to wheel brakes. The pilots and copilots brake pedals are interconnected. Three self-adjusting pins in each brake compensate automatically to maintain a running clearance of 0.014 - 0.039 inch between the disc and the flat surface of the housing.

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fluid return is shut off. Hydraulic fluid pressure builds up in proportion to the applied force on the pedals. Fluid from the power brake valves, is directed through the decelostats (antiskid controllers when installed)

to the individual brakes. The decelostats permit full application of the wheel brakes, under normal operating conditions, without the danger of skidding the main gear wheels. If during brake application, a wheel suddenly decelerates at a rate that would cause it to skid, the decelostat releases the brake by opening its exhaust valve to exhaust hydraulic fluid from the brake unit and, at the same time, shuts off inlet pressure. This removes the braking effort causing the tendency to skid. When the wheel regains its normal speed, the decelostat reapplies the brake by allowing hydraulic fluid from the brake valve to again flow to the brake unit.

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shuttle valves shift, directing hydraulic pressure to the shuttle valves at the wheel brake units. These shuttle valves also shift, directing pressure to the three pucks in each of the wheel brake units.

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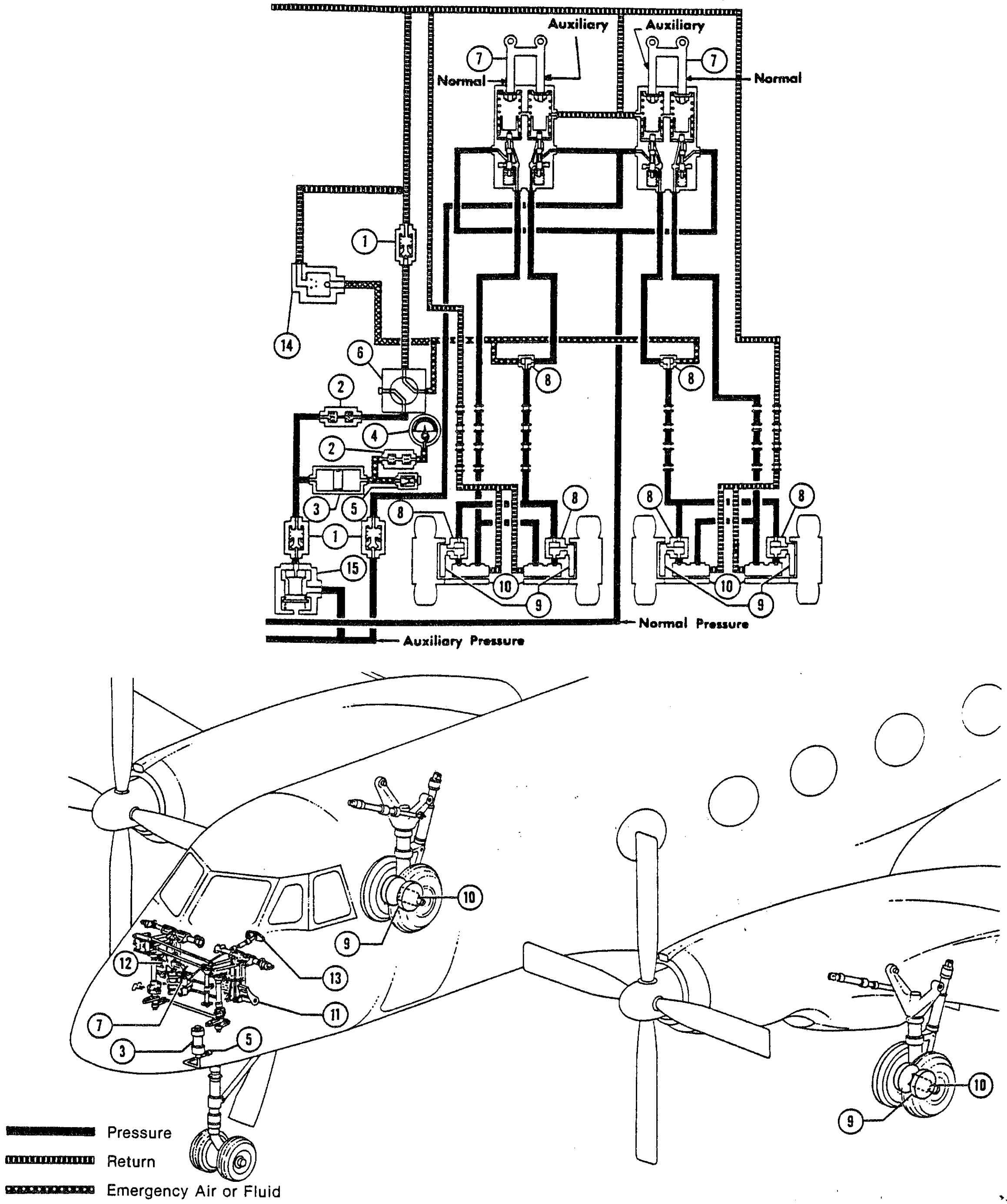

★Aircraft 88 - 200 including 322 and 323, excluding 114 and aircraft having ASC 137, only.

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2. Restrictor
3. Accumulator
4. Accumulator Air Pressure Gage
5. Accumulator Air Filler Valve
6. Parking and Emergency Brake Selector Valve
7. Power Brake Valve
8. Shuttle Valve
9. Wheel Brake
10. Decelostat Assembly
11. Pilots Brake and Rudder Pedals
12. Copilots Brake and Rudder Pedals
13. Parking and Emergency Brake T-handle
★14. Pressure Relief Valve
★15. Hydraulic Shutoff Valve

Brake Control System
Figure 1

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WHEEL BRAKE SYSTEM — MAINTENANCE PRACTICES

1. **Auxiliary System — Returning to Normal Operation**

To return auxiliary system to normal, shuttle valves in lines and at wheel brake units must be repositioned. This is accomplished by applying main hydraulic system pressure and depressing the rudder pedals. This supplies main hydraulic system pressure from the power brake valves to the shuttle valves.

2. **Brake Lining — Clearance Check**

With brakes applied, measure distance between face of disc and flat surface housing. If this measurement is 7/16-inch or more, the brake linings must be replaced. Refer to Goodyear Installation and Maintenance Instructions Handbook.

3. **Wheel Brake System Linkages — Rigging**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Following installation of right and left torque tube assemblies, right and left jack shaft assemblies, and right and left valve crank assemblies, insert four ¼-inch diameter neutral pins, as shown in Figure 201.

CAUTION: WITH ONE END OF THE ROD DISCONNECTED, IT IS POSSIBLE TO MOVE THE DISCONNECTED END FAR ENOUGH SIDEWAYS TO EXCEED THE EIGHT DEGREE ANGULAR TOLERANCE OF THE SELF-ALIGNING BEARING. THERE IS ENOUGH OF A MOMENT ARM THAT IT TAKES A VERY SMALL FORCE AT THE DISCONNECTED END TO DAMAGE THE BEARING AT THE CONNECTED END.

B. When installing four brake rod pedal assemblies, adjust rods so that aft ends in cranks are in contact with fixed pedal stops.

C. Remove four neutral pins, install two jackshafts to torque tube rod assemblies, and adjust four rods, referred to in Step B. above, so that they remain in contact with their stops.

D. Install power brake valve rod assemblies. Adjust rollers of left and right valve cranks, referred to in Step A. above, to clear foot of normal brake system valve stems of power brakes valves by 0.022 - 0.032 inch, while linkages are against their respective stops when held at rollers, as shown. (See Figure 201, Detail B).

NOTE: The foot of each auxiliary brake system power brake valve stem shall have equal or greater clearance than the normal brake system valve stem.

E. Having these clearances, lengthen power brake valve rods referred to in Step D. above one half turn of their rod ends and secure. After lengthening rods, clearance of 0.022 - 0.032 inch will be eliminated.

NOTE: This increase in length will remove valve stems from their internal stops by 0.010 - 0.020 inch, preloading the pedals as required.

F. Install four rod assemblies. Adjust four rods against their respective stops to obtain 23° pedal angle with center line of pedal hanger. (See Figure 201, Detail A).

4. **Normal / Emergency Brake Bleeding — Service**

A. Normal Brake System Bleeding:

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: Following procedure is given for bleeding both brakes on same side of aircraft. If all brakes must be bled, perform each sequence on all brakes before cont nuing to next brake. This will eliminate changing hydraulic power source repeatedly.

(1) Connect external hydraulic power to aircraft.
(2) Remove screw from bleeder valve at top of wheel brake and install bleeder hose.

NOTE: Either inboard or outboard brake assembly can be bled first, but both must be bled separately.

(3) Place free end of bleeder in suitable container to catch fluid.
(4) Depress individual brake pedal in cockpit.
(5) Open bleeder valve and allow fluid to flow until it is air free and clear. Tighten bleeder valve (bleed approximately one pint).
(6) Bleed opposite brake on same gear by repeating Steps (1) - (5) above.

B. Bleeding Brakes Using Auxiliary Hydraulic Pressure.
(1) Remove external hydraulic power from aircraft.

CAUTION: BECAUSE OF HIGH AMPERAGE DRAW REQUIRED FOR OPERATION OF AUXILIARY PUMP, USE OF AIRCRAFT BATTERY AS A POWER SOURCE TO OPERATE PUMP SHOULD BE LIMITED TO AS SHORT A PERIOD OF TIME AS POSSIBLE. IT IS STRONGLY RECOMMENDED THAT EXTERNAL ELECTRICAL POWER BE USED.

(2) Operate auxiliary hydraulic pump (normal system pressure depleted) and bleed both brakes by repeating Steps A.(4), A.(5) and A.(6) above.

C. Park / Emergency Brake Bleeding.
(1) Engage parking brake with PARK/EMER hand brake.
(2) Operate auxiliary hydraulic pump (normal system pressure depleted) and bleed both brakes by repeating Steps A.(4), A.(5) and A.(6) above.
(3) Bleed opposite brake as in Step C.(2) above and then tighten bleeder valve.
(4) Remove bleeder hoses and install all screws in bleeder valves.
(5) Perform Wheel Brake — Operational Test.
(6) Check fluid level of hydraulic reservoir and service as required.

5. **Main Gear Outboard And Inboard Wheel Brake — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Release PARK/EMER brake handle and remove hydraulic power from aircraft.
(2) Remove wheel.
(3) Disconnect hydraulic lines from brake shuttle valve.
(4) Remove, clean, visually inspect and retain mounting hardware for reuse. NDT any suspect hardware.
(5) Remove brake.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(6) Clean axle flange with solvent.
(7) Inspect axle flange and axle fillet for corrosion and rust. Carefully inspect axle fillet for evidence of cracks.

NOTE: While brakes are removed, it is recommended that main gear uplock roller bolts be inspected.

B. Installation
(1) Position brake on axle flange and install bolts, washers and nuts. Face bolt heads toward wheel (away from strut). See Figure 203.

NOTE: Ensure bolts are installed in their proper location so that normal required number of threads protrude beyond nuts. (1-1/2 to 3 threads). See Figure 203 for bolt locations.

(2) Torque nuts to 450 - 500 inch-pounds.
(3) Connect hydraulic lines to shuttle valve.
(4) Bleed brakes.
(5) Install wheel.

6. **Main Gear Wheel Brake — Retorque**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove wheel assembly.
B. Back off brake bolts to zero torque.
C. Retorque brake bolts to 450 - 500 inch-pounds.
D. Install wheel assembly.

7. **Wheel Brake Shuttle Valve — Removal / Installation**

A. Removal

(1) Release PARK / EMER brake handle and remove hydraulic power from aircraft.
(2) Disconnect hydraulic lines from brake shuttle valve.
(3) Break safety-wire and remove universal fitting that secures shuttle valve to brake. Discard O-rings.
(4) Remove shuttle valve and separate plate from valve by removing nuts, screws and washers.

B. Installation:

(1) Mount plate on shuttle valve and secure with bolts, washers and nuts.
(2) Install new O-ring on universal fitting and insert fitting in shuttle valve.
(3) Install new O-ring on shank of fitting, then install shuttle valve in brake. Torque fitting to 40-65 inch-pounds. Safety-wire fitting to cap on shuttle valve.

CAUTION: ENSURE HOLE IN PLATE IS ENGAGED WITH ALIGNMENT PIN ON BRAKE HOUSING.

(4) Connect hydraulic lines.
(5) Bleed brakes.
(6) Perform Wheel Brake System — Operational Test.

8. **Decelostat — Removal / Installation**

NOTE: Decelostats may be removed with or without removing main wheel assemblies. Following procedure is given without wheel removal.

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Operate PARK / EMER brake control handle until brake pressure gage needle stabilizes.
(2) With parking brake off, chock aircraft wheels.

CAUTION: ENSURE THAT HYDRAULIC FLUID DOES NOT CONTACT DECELOSTAT TIRES.

(3) Remove hydraulic lines from decelostat.
(4) Remove nut and washer from decelostat attaching bolt.

NOTE: To ensure decelostat tire is not damaged or gouged while removing bolt, place a thin piece of plastic or similar material between head of bolt and decelostat tire.

(5) Carefully withdraw attaching bolt from decelostat.
(6) Remove decelostat.

B. Installation

(1) Install springs and washer on decelostat mount pins.

CAUTION: DECELOSTATS P/N P25306-22 OR P/N P25340 MUST ALWAYS BE INSTALLED ON RIGHT OUTBOARD OR LEFT INBOARD WHEEL POSITIONS. P/N P25307-32 or P/N P25341 MUST ALWAYS BE INSTALLED IN LEFT OUTBOARD OR RIGHT INBOARD POSITION.

(2) Install decelostats and attaching bolt, bushings, washers, nuts and adjust. (See Figure 204).
(3) Install hydraulic lines on decelostat.
(4) Bleed brakes on applicable side of aircraft.

9. **Decelostat — Lubrication**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: At any time the flywheel of the decelostat is removed, the bearings and bearing races must be lubricated.

A. Remove decelostat from landing gear.
B. Remove safety-wire and backing plate retaining bolts.

CAUTION: USE CARE TO PREVENT LOSS OF 23 INDIVIDUAL BALL BEARINGS REMAINING IN BACKING PLATE AND 23 INDIVIDUAL BALL BEARINGS IN THE VALVE BODY.

C. Hold decelostat and flywheel assembly firmly with backing plate down. Remove valve body and flywheel assembly as one unit.
D. Hold flywheel and valve body as one unit, rotate so that valve body is down, then lift flywheel assembly from valve body.
E. Carefully remove bearings from backing plate and valve body (23 bearings in each assembly), clean and grease backing plate, valve body and flywheel. Replace defective ball bearings.

CAUTION: DO NOT DISTURB THE FELT SEALS IN THE BACKING PLATE OR VALVE BODY. SEALS CANNOT BE REPLACED WITHOUT SPECIAL TOOLS.

NOTE: It is recommended that when unit is disassembled that ball bearings be replaced.

F. Check condition of felt seals in backing plate and valve body.
G. Check flywheel assembly to ensure that weatherproof solder joint between tire shell assembly and flyweight is not broken.
H. Check tire assembly for excessive wear and damage and replace if necessary.
I. Coat bearing races of backing plate and valve body with grease.
J. Inbed ball bearings in grease applied to backing plate and valve body (23 ball bearings per side).
K. With valve body on bench, install flywheel assembly, using care not to dislocate ball bearings.
L. Install pivot blocks on backing plate into the valve body with large diameter hole away from assembly.

CAUTION: ENSURE THAT BALL BEARINGS DO NOT DROP FROM BACKING PLATE.

M. Install backing plate to flywheel and valve body, install and tighten bolts.
N. Rotate flywheel several times in both directions to ensure by feel and sound that the ball bearings are properly located.
O. Safety backing plate attaching bolts.
P. Install decelostat on landing gear.

10. **Brake Accumulator — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WEEL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock main gear wheels.
(2) Pull and placard DOOR CONTROL circuit breaker.
(3) Operate PARK/EMER brake handle (approximately eight times) on center pedestal until needle on accumulator pressure gage stabilizes.

WARNING: DO NOT LOOSEN VALVE HOUSING TO DISCHARGE NITROGEN. PRESSURE COULD EJECT VALVE ASSEMBLY AND INJURE PERSONNEL OR DAMAGE AIRCRAFT.

(4) Gain access to accumulator on forward side of nose wheel well and slowly open valve on accumulator servicing port to relieve preload air charge on accumulator; wait until pressure drops to zero before continuing further.

(5) Place a container suitable for hydraulic fluid under accumulator.
(6) Disconnect lines and remove fittings from accumulator.
(7) Remove attaching hardware from upper and lower mounting clamps; spread clamps and remove accumulator from aircraft.

B. Installation
(1) Place accumulator in position between upper and lower mounting clamps.
(2) Install attaching hardware.
(3) Replace fitting using new O-rings in upper fitting and in lower fittings.
(4) Connect hydraulic and air lines to fittings.
(5) Service accumulator with nitrogen or dry compressed air.
(6) Check for leaks on air side of accumulator using soapy solution.
(7) Depress DOOR CONT circuit breaker and operate auxiliary pump until accumulator reaches its full charge; turn auxiliary pump off.
(8) Ensure that accumulator charge does not dissipate at all with pump off.
(9) Check for leakage at hydraulic fittings.
(10) Check fluid level in hydraulic reservoir and service as required.
(11) Remove safety devices as required.

11. **Brake Pressure Line Restrictor — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.
(2) Gain access to nose wheel well vicinity of parking brake selector valve and place a container suitable for hydraulic fluid under work area.
(3) Bleed pressure from parking brake system by operating PARK / EMER brake selector and observe accumulator pressure gage; when needle on gage stabilizes, pressure has been bled, needle should stabilize at 800 $\pm$ 25 psi.
(4) Pull and tag DOOR CONT circuit breaker to keep auxiliary pump from operating while restrictor is being removed.
(5) Disconnect lines and fittings necessary to remove restrictor from aircraft. Remove restrictor.

B. Installation
(1) Discard old O-rings and replace with new ones.
(2) Install restrictor to aft side of parking brake selector valve and connect lines.
(3) Reset DOOR CONT circuit breaker.
(4) Using external power operate auxiliary pump and pressurize system.
(5) Operate PARK / EMER brake handle five or six times and check for leaks in work area.
(6) After completion of leak check remove ground safety devices as required.

12. **Brake Pressure Line Restrictor / Filter — Inspection**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Chock wheels.
B. Remove brake pressure line restrictor.
C. Remove the filter elements from restrictor.
D. Inspect filter elements for contamination and condition.
E. Clean or replace elements.
F. Install elements in restrictor.
G. Install restrictor using new O-rings.

13. **Park / Emergency Brake Selector Valve — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL. ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.

(2) Pull DOOR CONT circuit breaker and placard.
(3) Operate PARK / EMER brake handle approximately eight times on center console until needle on brake accumulator gage stabilizes.
(4) Gain access to valve located on right outboard side of nose wheel well and place a container suitable for hydraulic fluid under valve.
(5) Disconnect valve arm from valve shaft from inside cockpit.
(6) Disconnect hydraulic lines and remove fittings from valve. Discard O-rings.
(7) Remove mounting hardware from valve. Remove valve from aircraft.

B. Installation
(1) Place valve on a clean work area and install fittings using new O-rings.
(2) Install valve in aircraft using retained hardware.
(3) Install lines on valve fittings.
(4) Reconnect valve arm to valve shaft.
(5) Remove placard and depress DOOR CONT circuit breaker.
(6) Connect a source of electrical power to aircraft.
(7) Check accumulator gage. Pressure should be 800 ± 25 psi. Service as required.
(8) Operate auxiliary pump.
(9) Operate PARK / EMER brake handle on center console and check to see that brakes are parked.
(10) Operate handle three or four cycles to bleed system.
(11) Stop auxiliary pump operation. Accumulator pressure should remain stable.
(12) Check fluid level in hydraulic reservoir and service if required.
(13) Remove ground safety devices as required.

14. **Park / Emergency Brake Relief Valve — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.
(2) Pull DOOR CONT circuit breaker.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO ALLOW RESIDUAL PRESSURE TO BLEED FROM RESERVOIR.

(3) Cycle PARK / EMER brake handle approximately eight times until needle on accumulator pressure gage stabilizes.
(4) Disconnect lines and remove fittings from valve in nose wheel well. Discard O-rings.
(5) Remove valve from aircraft.

B. Installation
(1) Install fittings in valve using new O-rings.
(2) Place valve in position and connect hydraulic lines to valve.
(3) Connect external power to aircraft. Depress DOOR CONT circuit breaker and operate auxiliary pump.
(4) Cycle PARK / EMER brake handle three or four times to bleed air from system and check for leaks at all fittings.
(5) Check reservoir for proper fluid level and service if required.
(6) Remove ground safety devices as required.

15. **Parking Brake Shutoff Valve — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.
(2) Pull DOOR CONT circuit breaker and placard.

(3) Actuate PARK / EMER brake control until accumulator gage pressure stabilizes.
(4) Place a container suitable for hydraulic fluid under valve on right side of nose wheel well.
(5) Disconnect lines and remove fittings from valve. Remove valve from aircraft.

B. Installation
(1) Install fittings in valve using new O-rings.
(2) Place valve in position and install lines to fittings.
(3) Depress DOOR CONT circuit breaker and operate auxiliary pump.
(4) Pressure on accumulator pressure gage should not exceed 1600 psi.
(5) Cycle PARK / EMER brake handle approximately eight times to bleed systems.
(6) Check hydraulic reservoir for proper fluid level, service if required.
(7) Remove ground safety devices.

16. **Wheel Brake System — Operational Test**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

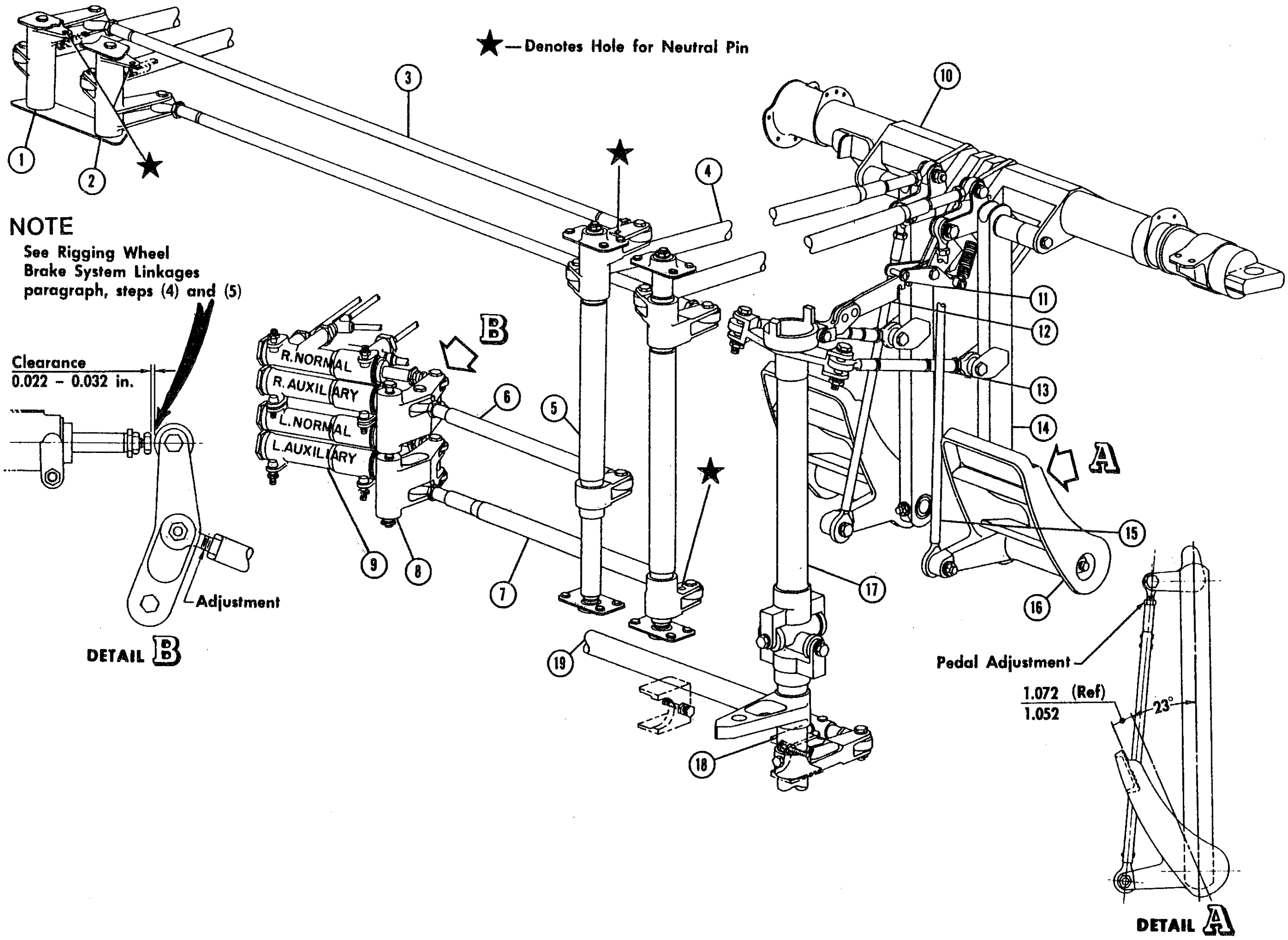
A. Jack aircraft.
B. With no hydraulic power on aircraft operate auxiliary hydraulic pump.
C. Spin wheels at each gear and depress brake pedals (with pump running); repeat three times.

NOTE: Braking action should be equal with each application. There should be no evidence of drag, soft or spongy pedals.

D. With brakes released, spin wheels. Wheels should rotate freely without drag when turned.
E. Connect external hydraulic power to aircraft.
F. Spin wheels at each gear and depress brake pedals; braking action should be same as noted in Step C. above.
G. With brakes released, spin wheels. Wheels should rotate freely without drag when turned.
H. Check that brake accumulator gage indicates 1450 to 1500 psi on aircraft having ASC 137A. Those Aircraft not having ASC 137A should read 1900 psi.
I. With brakes released, spin wheels; pull PARK / EMER handle; braking action should be sudden with no evidence of drag.

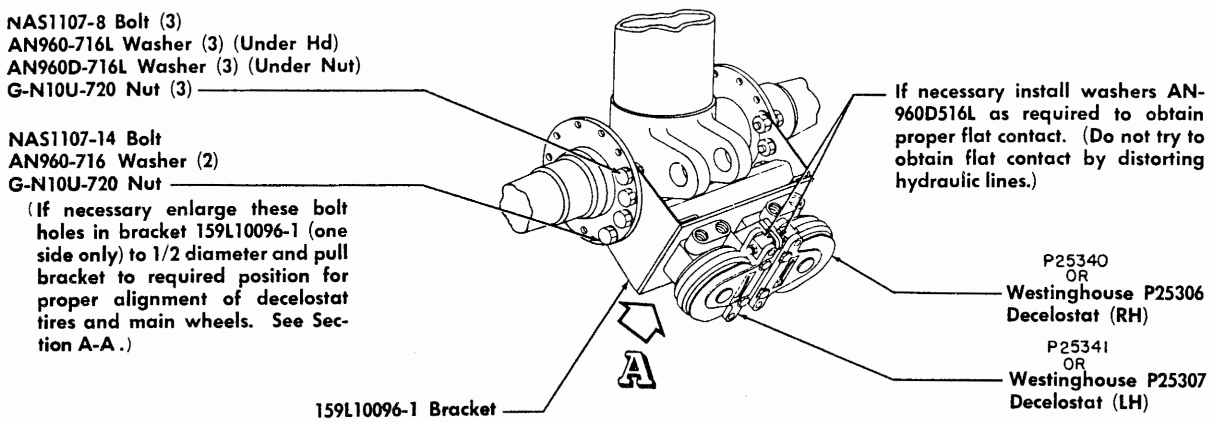
NOTE: Above brake application can be repeated several times before accumulator hydraulic pressure fully dissipates.

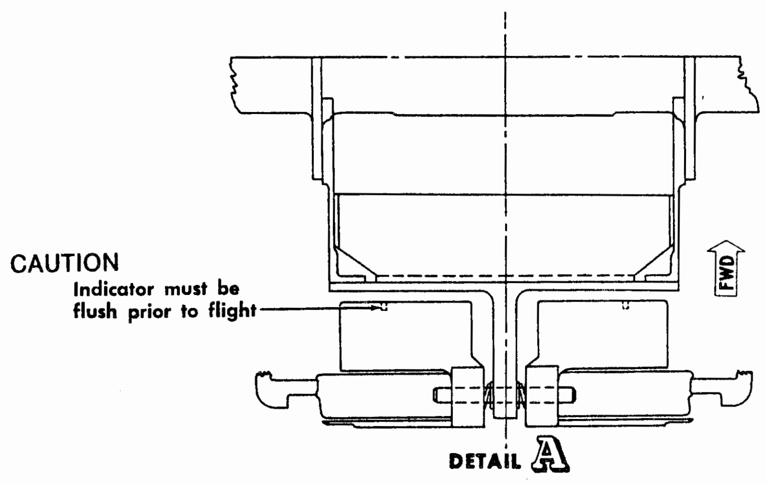
J. Release handle. Wheels should rotate freely without drag when they are spun.
K. Using auxiliary hydraulic pump, pressurize accumulator until gage reads 1450 to 1500 psi on aircraft having ASC 137A. Those aircraft not having ASC 137A should read 1900 psi.
L. Check brake system for leakage.
M. Lower jacks.

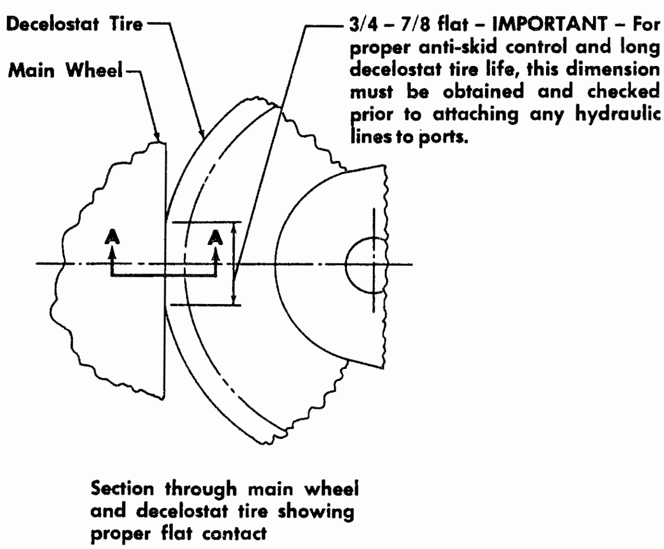


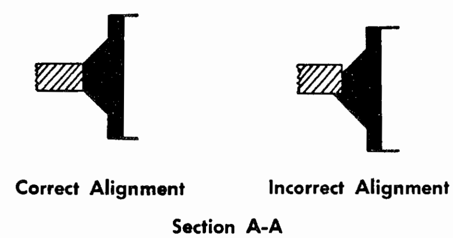
1. Jackshaft Assembly
2. Jackshaft Assembly
3. Jackshaft to Torque Tube Rod Assembly (2)
4. Rod Assembly (4)
5. Torque Tube Assembly (2)
6. Power Brake Valve Rod Assembly
7. Power Brake Valve Rod Assembly
8. Valve Crank Assembly (2)
9. Power Brake Valve (2)
10. Tube Assembly (2)
11. Arm Assembly (4)
12. Ratchet Assembly (2)
13. Hangar Rod Assembly (4)
14. Hangar Assembly (4)
15. Brake Pedal Rod Assembly (4)
16. Brake Pedal (4)
17. Tube Assembly (2)
18. Push Rod Tube Assembly (2)
19. Push Rod

Brake Linkage Rigging
Figure 201



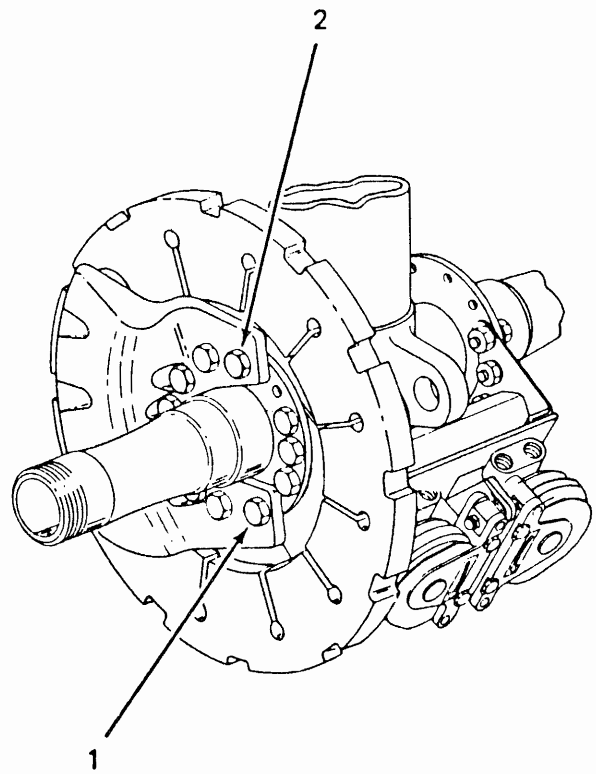




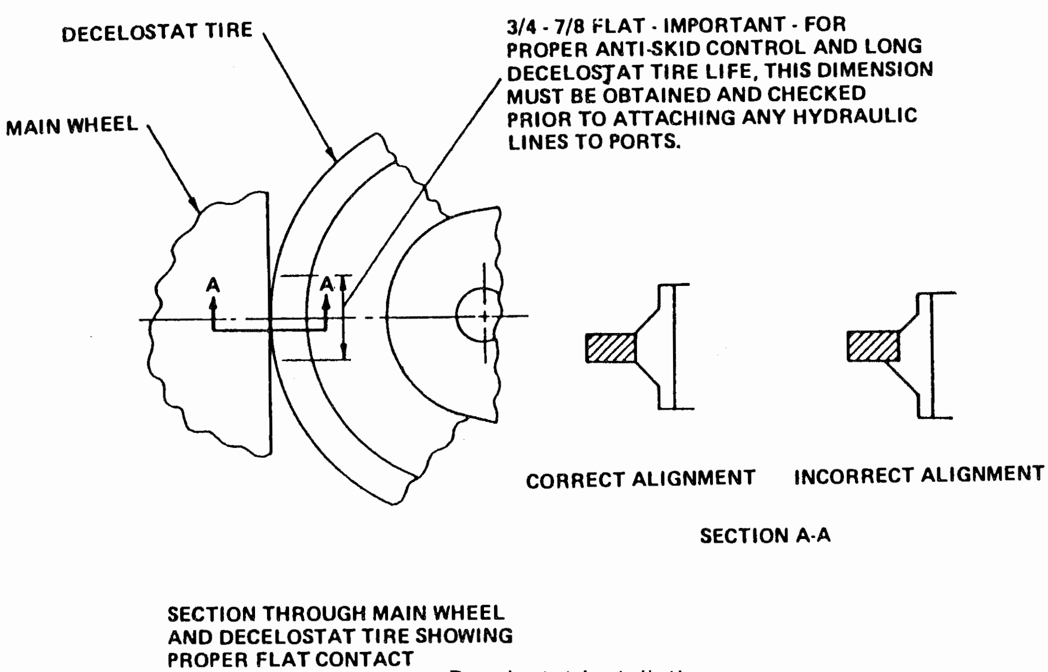


Installation and Adjustment of Decelostat Units
Figure 202

1. Bolt (1)
   Washer (1)
   Nut (1)
2. Bolt (7)
   Washer (7)
   Nut (7)



Main Gear Wheel Brake Assembly
Figure 203



Decelostat Installation
Figure 204

WHEEL BRAKE SYSTEM — MAINTENANCE PRACTICES

1. **Auxiliary System — Returning to Normal Operation**

To return auxiliary system to normal, shuttle valves in lines and at wheel brake units must be repositioned. This is accomplished by applying main hydraulic system pressure and depressing the rudder pedals. This supplies main hydraulic system pressure from the power brake valves to the shuttle valves.

2. **Brake Lining — Clearance Check**

With brakes applied, measure distance between face of disc and flat surface housing. If this measurement is 7/16 inch or more, the brake linings must be replaced. Refer to Aircraft Braking System Maintenance Manual.

3. **Wheel Brake System Linkages — Rigging**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Following installation of right and left torque tube assemblies, right and left jack shaft assemblies, and right and left valve crank assemblies, insert four 1/4 inch diameter neutral pins, as shown in Figure 201.

CAUTION: WITH ONE END OF THE ROD DISCONNECTED, IT IS POSSIBLE TO MOVE THE DISCONNECTED END FAR ENOUGH SIDEWAYS TO EXCEED THE EIGHT DEGREE ANGULAR TOLERANCE OF THE SELF-ALIGNING BEARING. THERE IS ENOUGH OF A MOMENT ARM THAT IT TAKES A VERY SMALL FORCE AT THE DISCONNECTED END TO DAMAGE THE BEARING AT THE CONNECTED END.

B. When installing four brake rod pedal assemblies, adjust rods so that aft ends in cranks are in contact with fixed pedal stops.

C. Remove four neutral pins, install two jackshafts to torque tube rod assemblies, and adjust four rods, referred to in Step B. above, so that they remain in contact with their stops.

D. Install power brake valve rod assemblies. Adjust rollers of left and right valve cranks, referred to in Step above, to clear foot of normal brake system valve stems of power brakes valves by 0.022-0.032 inch, while linkages are against their respective stops when held at rollers, as shown. (See Figure 201 Detail B).

NOTE: The foot of each auxiliary brake system power brake valve stem shall have equal or greater clearance than the normal brake system valve stem.

E. Having these clearances, lengthen power brake valve rods referred to in Step D. above one half turn of their rod ends and secure. After lengthening rods, clearance of 0.022-0.032 inch will be eliminated.

NOTE: This increase in length will remove valve stems from their internal stops by 0.010 - 0.020 inch, preloading the pedals as required.

F. Install four rod assemblies. Adjust four rods against their respective stops to obtain 23° pedal angle with center line of pedal hanger. (See Figure 201 Detail A).

4. **Normal / Emergency Brake Bleeding — Service**

A. Normal Brake System Bleeding

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: Following procedure is given for bleeding both brakes on same side of aircraft. If all brakes must be bled, perform each sequence on all brakes before continuing to next brake. This will eliminate changing hydraulic power source repeatedly.

(1) Connect external hydraulic power to aircraft.

(2) Remove screw from bleeder valve at top of wheel brake and install bleeder hose.

NOTE: Either inboard or outboard brake assembly can be bled first, but both must be bled separately.

(3) Place free end of bleeder in suitable container to catch fluid.

(4) Depress individual brake pedal in cockpit.

(5) Open bleeder valve and allow fluid to flow until it is air free and clear. Tighten bleeder valve (bleed approximately one pint).

(6) Bleed opposite brake on same gear by repeating Steps (1) thru (5) above.

B. Bleeding Brakes Using Auxiliary Hydraulic Pressure.

(1) Remove external hydraulic power from aircraft.

CAUTION: BECAUSE OF HIGH AMPERAGE DRAW REQUIRED FOR OPERATION OF AUXILIARY PUMP, USE OF AIRCRAFT BATTERY AS A POWER SOURCE TO OPERATE PUMP SHOULD BE LIMITED TO AS SHORT A PERIOD OF TIME AS POSSIBLE. IT IS STRONGLY RECOMMENDED THAT EXTERNAL ELECTRICAL POWER BE USED.

(2) Operate auxiliary hydraulic pump (normal system pressure depleted) and bleed both brakes by repeating Steps A.(4), A.(5) and A.(6) above.

C. Park / Emergency Brake Bleeding.

(1) Engage parking brake with PARK/EMER hand brake.

(2) Operate auxiliary hydraulic pump (normal system pressure depleted) and bleed both brakes by repeating Steps A.(4), A.(5) and A.(6) above.

(3) Bleed opposite brake as in Step C.(2) above and then tighten bleeder valve.

(4) Remove bleeder hoses and install all screws in bleeder valves.

(5) Perform Wheel Brake — Operational Test, see this Section.

(6) Check fluid level of hydraulic reservoir and service as required.

5. Main Gear Outboard / Inboard Wheel Brake — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Release PARK/EMER brake handle and remove hydraulic power from aircraft.

(2) Remove wheel.

(3) Disconnect hydraulic lines from brake shuttle valve.

(4) Remove, clean, visually inspect and retain mounting hardware for reuse. NDT any suspect hardware.

(5) Remove brake.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(6) Clean axle flange with solvent.

(7) Inspect axle flange and axle fillet for corrosion and rust. Carefully inspect axle fillet for evidence of cracks.

NOTE: While brakes are removed, it is recommended that main gear uplock roller bolts be inspected.

B. Installation

(1) Position brake on axle flange and install bolts, washers and nuts. Face bolt heads toward wheel (away from strut). See Figure 203.

NOTE: Ensure bolts are installed in their proper location so that normal required number of threads protrude beyond nuts. (1-1/2 to 3 threads). See Figure 203 for bolt locations.

(2) Torque nuts to 450-500 inch-pounds.

(3) Connect hydraulic lines to shuttle valve.

(4) Bleed brakes.

(5) Install wheel.

6. Main Gear Wheel Brake — Retorque

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove wheel assembly.

B. Back off brake bolts to zero torque.

C. Retorque brake bolts to 450-500 inch-pounds.

D. Install wheel assembly.

7. Wheel Brake Shuttle Valve — Removal / Installation

A. Removal

(1) Release PARK/EMER brake handle and remove hydraulic power from aircraft.

(2) Disconnect hydraulic lines from brake shuttle valve.

(3) Break safety-wire and remove universal fitting that secures shuttle valve to brake. Discard O-rings.

(4) Remove shuttle valve and separate plate from valve by removing nuts, screws and washers.

B. Installation:

(1) Mount plate on shuttle valve and secure with bolts, washers and nuts.

(2) Install new O-ring on universal fitting and insert fitting in shuttle valve.

(3) Install new O-ring on shank of fitting, then install shuttle valve in brake. Torque fitting to 40-65 inch-pounds. Safety-wire fitting to cap on shuttle valve.

CAUTION: ENSURE HOLE IN PLATE IS ENGAGED WITH ALIGNMENT PIN ON BRAKE HOUSING.

(4) Connect hydraulic lines.

(5) Bleed brakes.

(6) Perform Wheel Brake System — Operational Test, see this Section

8. Decelostat — Removal / Installation

NOTE: Decelostats may be removed with or without removing main wheel assemblies. Following procedure is given without wheel removal.

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Operate PARK/EMER brake control handle until brake pressure gage needle stabilizes.

(2) With parking brake off, chock aircraft wheels.

CAUTION: ENSURE THAT HYDRAULIC FLUID DOES NOT CONTACT DECELOSTAT TIRES.

(3) Remove hydraulic lines from decelostat.

(4) Remove nut and washer from decelostat attaching bolt.

NOTE: To ensure decelostat tire is not damaged or gouged while removing bolt, place a thin piece of plastic or similar material between head of bolt and decelostat tire.

(5) Carefully withdraw attaching bolt from decelostat.

(6) Remove decelostat.

B. Installation

(1) Install springs and washer on decelostat mount pins.

CAUTION: DECELOSTATS P/N P25306-22 OR P/N P25340 MUST ALWAYS BE INSTALLED ON RIGHT OUTBOARD OR LEFT INBOARD WHEEL POSITIONS. P/N P25307-32 OR P/N P25341 MUST ALWAYS BE INSTALLED IN LEFT OUTBOARD OR RIGHT IN BOARD POSITION.

(2) Install decelostats and attaching bolt, bushings, washers, nuts and adjust. (See Figure 204)

(3) Install hydraulic lines on decelostat.

(4) Bleed brakes on applicable side of aircraft.

9. Decelostat — Lubrication

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: At any time the flywheel of the decelostat is removed, the bearings and bearing races must be lubricated.

A. Remove decelostat from landing gear.

B. Remove safety-wire and backing plate retaining bolts.

CAUTION: USE CARE TO PREVENT LOSS OF 23 INDIVIDUAL BALL BEARINGS REMAINING IN BACKING PLATE AND 23 INDIVIDUAL BALL BEARINGS IN THE VALVE BODY.

C. Hold decelostat and flywheel assembly firmly with backing plate down. Remove valve body and flywheel assembly as one unit.

D. Hold flywheel and valve body as one unit, rotate so that valve body is down, then lift flywheel assembly from valve body.

E. Carefully remove bearings from backing plate and valve body (23 bearings in each assembly), clean and grease backing plate, valve body and flywheel. Replace defective ball bearings.

CAUTION: DO NOT DISTURB THE FELT SEALS IN THE BACKING PLATE OR VALVE BODY. SEALS CAN NOT BE REPLACED WITHOUT SPECIAL TOOLS.

NOTE: It is recommended that when unit is disassembled that ball bearings be replaced.

F. Check condition of felt seals in backing plate and valve body.

G. Check flywheel assembly to ensure that weatherproof solder joint between tire shell assembly and flyweight is not broken.

H. Check tire assembly for excessive wear and damage and replace if necessary.

I. Coat bearing races of backing plate and valve body with grease.

J. Inbed ball bearings in grease applied to backing plate and valve body (23 ball bearings per side).

K. With valve body on bench, install flywheel assembly, using care not to dislocate ball bearings.

L. Install pivot blocks on backing plate into the valve body with large diameter hole away from assembly.

CAUTION: ENSURE THAT BALL BEARINGS DO NOT DROP FROM BACKING PLATE.

M. Install backing plate to flywheel and valve body, install and tighten bolts.

N. Rotate flywheel several times in both directions to ensure by feel and sound that the ball bearings are properly located.

O. Safety backing plate attaching bolts.

P. Install decelostat on landing gear.

10. Brake Accumulator — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock main gear wheels.

(2) Pull and placard DOOR CONTROL circuit breaker.

(3) Operate PARK/EMER brake handle (approximately eight times) on center pedestal until needle on accumulator pressure gage stabilizes.

WARNING: DO NOT LOOSEN VALVE HOUSING TO DISCHARGE NITROGEN. PRESSURE COULD EJECT VALVE ASSEMBLY AND INJURE PERSONNEL OR DAMAGE AIRCRAFT.

(4) Gain access to accumulator on forward side of nose wheel well and slowly open valve on accumulator servicing port to relieve preload air charge on accumulator; wait until pressure drops to zero before continuing further.

(5) Place a container suitable for hydraulic fluid under accumulator.

(6) Disconnect lines and remove fittings from accumulator.

(7) Remove attaching hardware from upper and lower mounting clamps; spread clamps and remove accumulator from aircraft.

B. Installation

(1) Place accumulator in position between upper and lower mounting clamps.

(2) Install attaching hardware.

(3) Replace fitting using new O-rings in upper fitting and in lower fittings.

(4) Connect hydraulic and air lines to fittings.

(5) Service accumulator with nitrogen or dry compressed air.

(6) Check for leaks on air side of accumulator using soapy solution.

(7) Depress DOOR CONT circuit breaker and operate auxiliary pump until accumulator reaches its full charge, turn auxiliary pump off.

(8) Ensure that accumulator charge does not dissipate at all with pump off.

(9) Check for leakage at hydraulic fittings.

(10) Check fluid level in hydraulic reservoir and service as required.

(11) Remove safety devices as required.

11. Brake Pressure Line Restrictor — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.

(2) Gain access to nose wheel well vicinity of parking brake selector valve and place a container suitable for hydraulic fluid under work area.

(3) Bleed pressure from parking brake system by operating PARK/EMER brake selector and observe accumulator pressure gage, when needle on gage stabilizes, pressure has been bled, needle should stabilize at 800 $\pm$25 psi.

(4) Pull and tag DOOR CONT circuit breaker to keep auxiliary pump from operating while restrictor is being removed.

(5) Disconnect lines and fittings necessary to remove restrictor from aircraft. Remove restrictor.

B. Installation

(1) Discard old O-rings and replace with new ones.

(2) Install restrictor to aft side of parking brake selector valve and connect lines.

(3) Reset DOOR CONT circuit breaker.

(4) Using external power operate auxiliary pump and pressurize system.

(5) Operate PARK/EMER brake handle five or six times and check for leaks in work area.

(6) After completion of leak check remove ground safety devices as required.

12. Brake Pressure Line Restrictor / Filter — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Chock wheels.

B. Remove brake pressure line restrictor.

C. Remove the filter elements from restrictor.

D. Inspect filter elements for contamination and condition.

E. Clean or replace elements.

F. Install elements in restrictor.

G. Install restrictor using new O-rings.

13. Emergency Brake / Wheel Door Shuttle Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels and bleed hydraulic charge from accumulator by actuating park/emergency brake handle approximately eight times until needle on accumulator pressure gage stabilizes.

(2) Pull DOOR CONT circuit breaker and placard.

(3) Locate valve to be replaced and place a suitable container in area of valve to be replaced with a minimum capacity of five gallons.

CAUTION: SLOWLY REMOVE FILLER CAP FROM RESERVOIR TO BLEED OFF RESIDUAL PRESSURE THAT MAY BE TRAPPED IN RESERVOIR.

NOTE: Left emergency brake shuttle valve is located on left side of wheel well approximately at Fuselage Station 0.75. Right emergency brake and wheel door shuttle valves are mounted side by side approximately Fuselage Station 0.75 right side.

(4) Disconnect lines and fittings from valve to be replaced.

(5) Remove mount bolts and remove valve from aircraft.

B. Installation

(1) Place valve in position and install and lockwire mount bolts.

(2) Install lines and fitting to valve.

NOTE: If an emergency brake shuttle valve was replaced, both brakes on corresponding side must be bled. If a wheel door shuttle valve was replaced, clamshell door should be operated to bleed system.

(3) Remove jury struts from clamshell doors.

(4) Close DOOR CONT circuit breaker and remove placard.

NOTE: Perform Steps (5), (6) and (7) below only if wheel door shuttle valve has been replaced.

(5) Close doors. This may be done by rotating either propeller in direction of rotation until all doors are closed (or using AUX to MAIN selector valve if installed and auxiliary pump).

(6) Open doors by operating auxiliary pump and placing clamshell door ground servicing valve in doors open position.

(7) Close and open doors six times or until all entrapped air is forced out of system.

(8) Service brake accumulator.

(9) Check capacity of hydraulic reservoir, and service if required.

14. Park / Emergency Brake Selector Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL. ENSURE ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.

(2) Pull DOOR CONT circuit breaker and placard.

(3) Operate PARK/EMER brake handle approximately eight times on center console until needle on brake accumulator gage stabilizes.

(4) Gain access to valve located on right outboard side of nose wheel well and place a container suitable for hydraulic fluid under valve.

(5) Disconnect valve arm from valve shaft from inside cockpit.

(6) Disconnect hydraulic lines and remove fittings from valve. Discard O-rings.

(7) Remove mounting hardware from valve. Remove valve from aircraft.

B. Installation

(1) Place valve on a clean work area and install fittings using new O-rings.

(2) Install valve in aircraft using retained hardware.

(3) Install lines on valve fittings.

(4) Reconnect valve arm to valve shaft.

(5) Remove placard and depress DOOR CONT circuit breaker.

(6) Connect a source of electrical power to aircraft.

(7) Check accumulator gage. Pressure should be 800 $\pm$25 psi. Service as required.

(8) Operate auxiliary pump.

(9) Operate PARK/EMER brake handle on center console and check to see that brakes are parked.

(10) Operate handle three or four cycles to bleed system.

(11) Stop auxiliary pump operation. Accumulator pressure should remain stable.

(12) Check fluid level in hydraulic reservoir and service if required.

(13) Remove ground safety devices as required.

15. Park / Emergency Brake Relief Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.

(2) Pull DOOR CONT circuit breaker.

CAUTION: DEPLETE HYDRAULIC SYSTEM PRESSURE AND SLOWLY REMOVE FILLER CAP FROM HYDRAULIC RESERVOIR TO ALLOW RESIDUAL PRESSURE TO BLEED FROM RESERVOIR.

(3) Cycle PARK/EMER brake handle approximately eight times until needle on accumulator pressure gage stabilizes.

(4) Disconnect lines and remove fittings from valve in nose wheel well. Discard O-rings.

(5) Remove valve from aircraft.

B. Installation

(1) Install fittings in valve using new O-rings.

(2) Place valve in position and connect hydraulic lines to valve.

(3) Connect external power to aircraft. Depress DOOR CONT circuit breaker and operate auxiliary pump.

(4) Cycle PARK/EMER brake handle three or four times to bleed air from system and check for leaks at all fittings.

(5) Check reservoir for proper fluid level and service if required.

(6) Remove ground safety devices as required.

16. Parking Brake Shutoff Valve — Removal / Installation

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.

(2) Pull DOOR CONT circuit breaker and placard.

(3) Actuate PARK/EMER brake control until accumulator gage pressure stabilizes.

(4) Place a container suitable for hydraulic fluid under valve on right side of nose wheel well.

(5) Disconnect lines and remove fittings from valve. Remove valve from aircraft.

B. Installation

(1) Install fittings in valve using new O-rings.

(2) Place valve in position and install lines to fittings.

(3) Depress DOOR CONT circuit breaker and operate auxiliary pump.

(4) Pressure on accumulator pressure gage should not exceed 1600 psi.

(5) Cycle PARK/EMER brake handle approximately eight times to bleed systems.

(6) Check hydraulic reservoir for proper fluid level, service if required.

(7) Remove ground safety devices.

17. Wheel Brake System — Operational Test

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

A. Jack aircraft.

B. With no hydraulic power on aircraft operate auxiliary hydraulic pump.

C. Spin wheels at each gear and depress brake pedals (with pump running); repeat three times.

NOTE: Braking action should be equal with each application. There should be no evidence of drag, soft or spongy pedals.

D. With brakes released, spin wheels. Wheels should rotate freely without drag when turned.

E. Connect external hydraulic power to aircraft.

F. Spin wheels at each gear and depress brake pedals; braking action should be same as noted in Step above.

G. With brakes released, spin wheels. Wheels should rotate freely without drag when turned.

H. Check that brake accumulator gage indicates 1450 to 1500 psi on aircraft having ASC 137A. Those Aircraft not having ASC 137A should read 1900 psi.

I. With brakes released, spin wheels; pull PARK/EMER handle; braking action should be sudden with no evidence of drag.

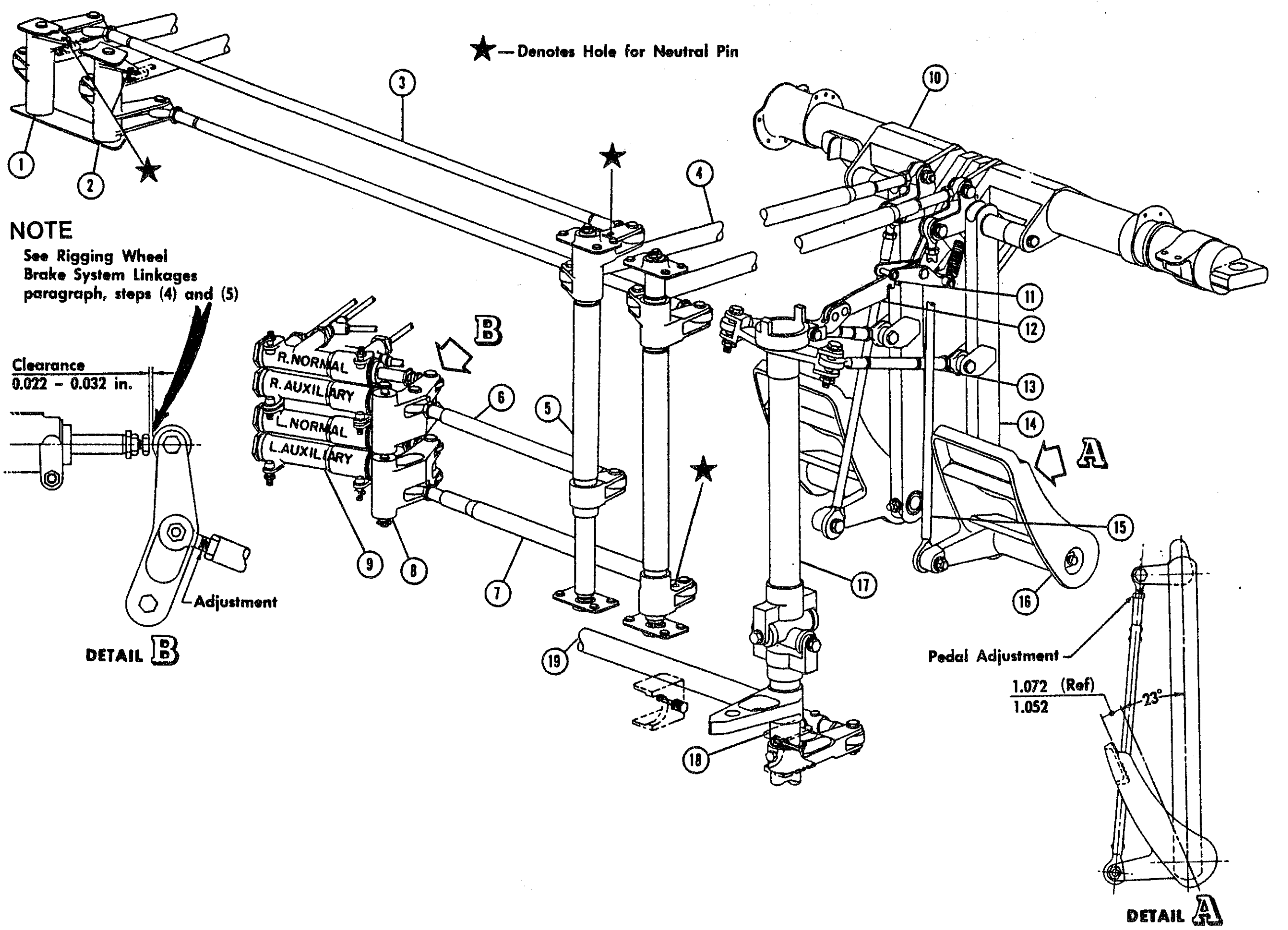
NOTE: Above brake application can be repeated several times before accumulator hydraulic pressure fully dissipates.

J. Release handle. Wheels should rotate freely without drag when they are spun.

K. Using auxiliary hydraulic pump, pressurize accumulator until gage reads 1450 to 1500 psi on aircraft having ASC 137A. Those aircraft not having ASC 137A should read 1900 psi.

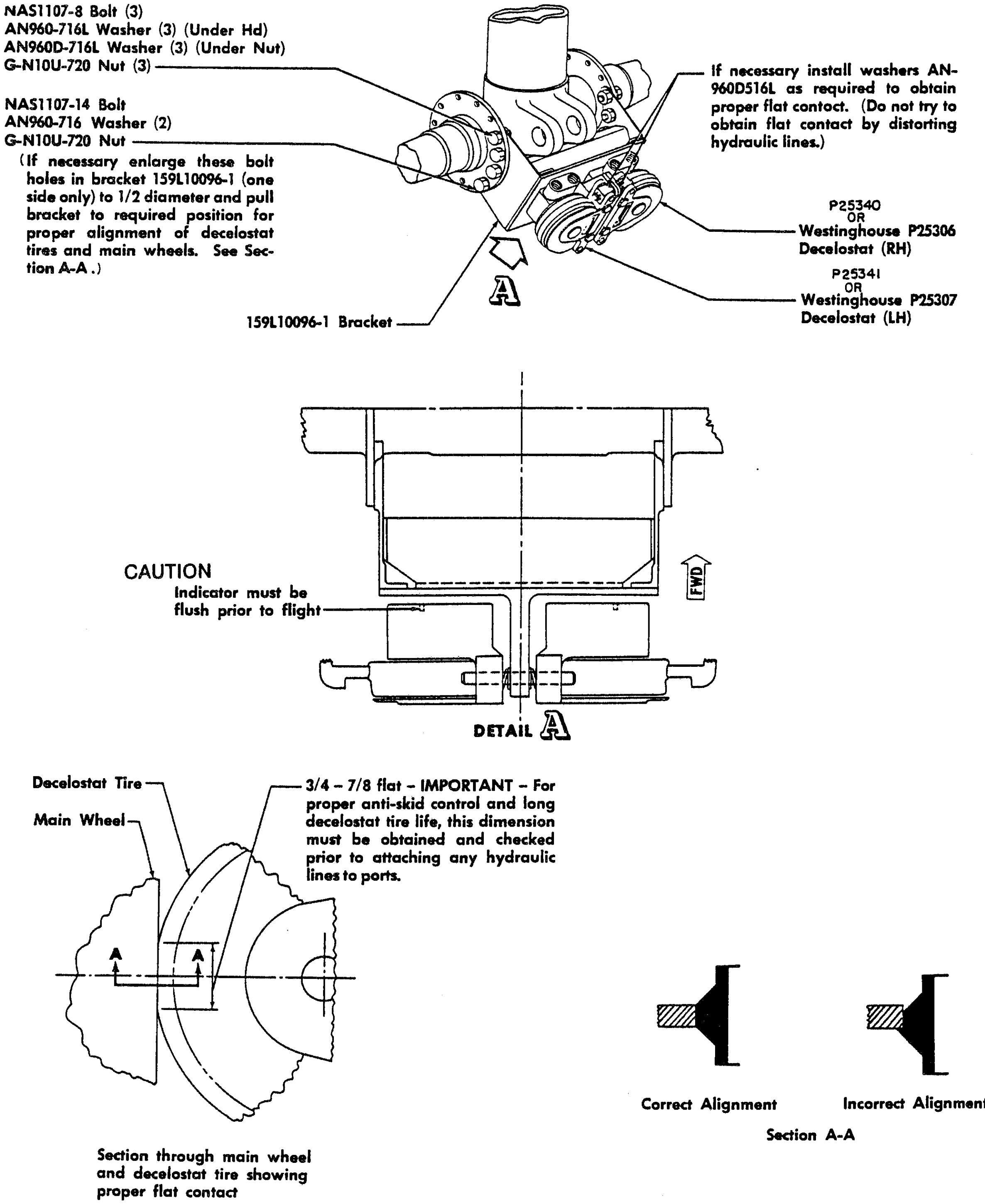
L. Check brake system for leakage.

M. Lower jacks.



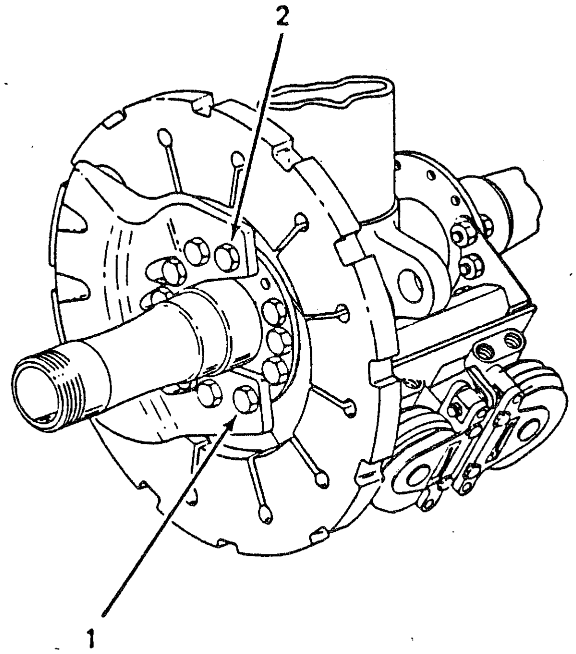
1. Jackshaft Assembly
2. Jackshaft Assembly
3. Jackshaft to Torque Tube Rod Assembly (2)
4. Rod Assembly (4)
5. Torque Tube Assembly (2)
6. Power Brake Valve Rod Assembly
7. Power Brake Valve Rod Assembly
8. Valve Crank Assembly (2)
9. Power Brake Valve (2)
10. Tube Assembly (2)
11. Arm Assembly (4)
12. Ratchet Assembly (2)
13. Hangar Rod Assembly (4)
14. Hangar Assembly (4)
15. Brake Pedal Rod Assembly (4)
16. Brake Pedal (4)
17. Tube Assembly (2)
18. Push Rod Tube Assembly (2)
19. Push Rod

Brake Linkage Rigging
Figure 201.

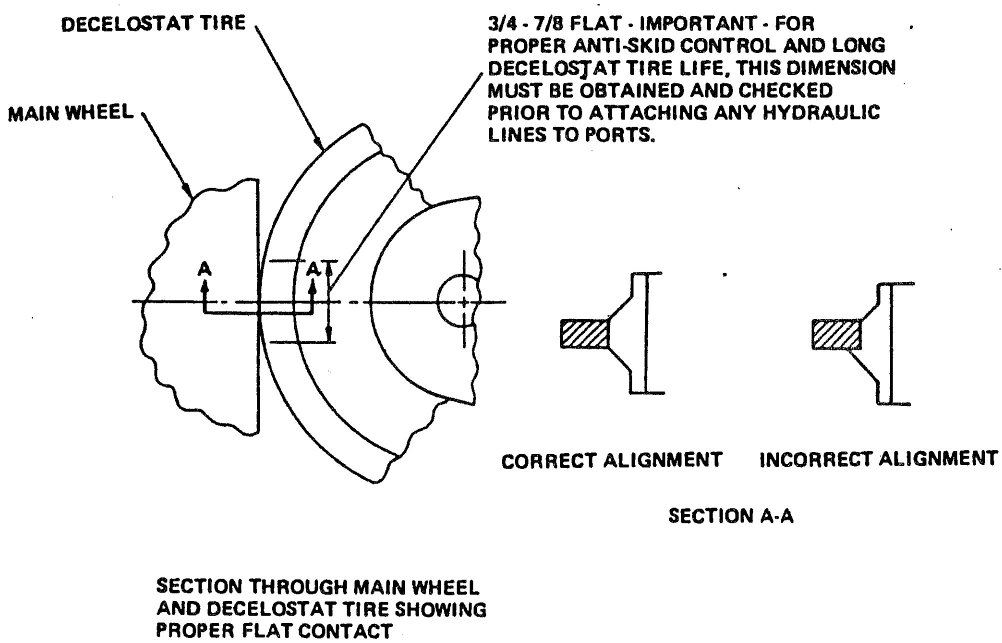


Installation and Adjustment of Decelostat Units
Figure 202.

1. Bolt (1)
   Washer (1)
   Nut (1)
2. Bolt (7)
   Washer (7)
   Nut (7)



Main Gear Wheel Brake Assembly
Figure 203.



Decelostat Installation
Figure 204.

AUXILIARY HYDRAULIC SYSTEM BRAKE ACCUMULATOR — DESCRIPTION

1. **Description**

The accumulator is an air-oil, 1500 psi operating pressure, 50 cubic-inch, piston type with an air connection and a hydraulic connection. It is located at the left side of nose wheel well. It is designed to operate with hydraulic fluid and at ambient and fluid temperatures from −54°C to +71°C (−65° F to +160°F). It is preloaded with dry compressed air or nitrogen to 800 psi. When fully charged the accumulator will hold 22.5 cubic-inches of hydraulic fluid.

CAUTION: WHEN CHECKING OR ADDING DRY COMPRESSED AIR OR NITROGEN TO THE ACCUMULATOR, ALL HYDRAULIC PRESSURE MUST BE RELIEVED.

The purpose of accumulator is to pressurize the propeller brake via the propeller brake selector valve and the parking / emergency brakes via the parking and emergency brake selector valve. Either PROP BRAKE handle or the PARK / EMER BRAKE handle discharges the stored accumulator hydraulic pressure.

WARNING: FIVE APPLICATIONS OF THE EMERGENCY AND PARKING BRAKE SYSTEM CAN BE MADE AFTER AUXILIARY PUMP SHUTDOWN.

NOTE: For propeller brake, see Chapter 83 — Accessory Gearbox.

An additional brake system accumulator gage and manual dump valve is installed in the nose wheel well by ASC 214. This change is to allow for one man operation of checking and servicing of the brake accumulator preload.

AUXILIARY HYDRAULIC SYSTEM BRAKE ACCUMULATOR — DESCRIPTION / OPERATION

1. Description

The accumulator is an air-oil, 1500 psi operating pressure, 50 cubic-inch, piston type with an air connection and a hydraulic connection. It is located at the left side of nose wheel well. It is designed to operate with hydraulic fluid and at ambient and fluid temperatures from −54°C to +71°C (−65°F to + 160°F). It is preloaded with dry compressed air or nitrogen to 800 psi. When fully charged the accumulator will hold 22.5 cubic-inches of hydraulic fluid.

CAUTION: WHEN CHECKING OR ADDING DRY COMPRESSED AIR OR NITROGEN TO THE ACCUMULATOR, ALL HYDRAULIC PRESSURE MUST BE RELIEVED.

The purpose of accumulator is to pressurize the propeller brake via the propeller brake selector valve and the parking/emergency brakes via the parking and emergency brake selector valve. Either PROP BRAKE handle or the PARK/EMER BRAKE handle discharges the stored accumulator hydraulic pressure.

WARNING: FIVE APPLICATIONS OF THE EMERGENCY AND PARKING BRAKE SYSTEM CAN BE MADE AFTER AUXILIARY PUMP SHUTDOWN.

NOTE: For propeller brake, see Accessory Gearbox Section Chapter 83.

An additional brake system accumulator gage and manual dump valve is installed in the nose wheel well by ASC 214. This change is to allow for one man operation of checking and servicing of the brake accumulator

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1. **Brake Accumulator — Service**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Chock wheels.
B. Operate PARK / EMER brake handle until brake pressure stabilizes.

NOTE: Pressure remaining is air pressure in accumulator. This must be 800 ± 25 psi.

C. If accumulator requires servicing, proceed as follows:
(1) Open landing gear doors to gain access to accumulator in nose wheel well.
(2) Ensure that 3/4-inch nut on accumulator filler valve is tight; then, remove valve cap.
(3) Connect nitrogen hose to filler valve. Loosen swivel nut on filler valve 3-1/2 turns.
(4) Slowly fill accumulator until pressure gage (in cockpit) reads 800 ± 25 psi.
(5) Tighten swivel nut on filler valve.
(6) Turn off nitrogen supply and remove hose.
(7) Install valve cap.
(8) Using auxiliary hydraulic pump, pressurize accumulator until pressure gage reads 1450 to 1500 psi on aircraft having ASC 137A. Those aircraft not having ASC 137A should read 1900 psi.

2. **Brake Accumulator Gage — Removal / Installation**
A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.
(2) Pull DOOR CONT circuit breaker and placard.
(3) Cycle PARK / EMER brake handle approximately eight times until needle on brake accumulator gage stabilizes.

WARNING: DO NOT LOOSEN VALVE HOUSING TO DISCHARGE NITROGEN. PRESSURE COULD EJECT VALVE ASSEMBLY AND INJURE PERSONNEL OR DAMAGE AIRCRAFT.

(4) Slowly open valve on accumulator air servicing port. Allow pressure to drop to zero before continuing.

NOTE: Accumulator air servicing port is located on forward side of nose wheel well.

(5) Disconnect air line from back of gage.
(6) Remove mounting screws. Remove Plexiglas shield and retain. Remove gage from underside of copilot instrument panel.

B. Installation
(1) Place gage in position on copilot instrument panel and install Plexiglas protector in front of gage face. Install mounting screws and tighten.
(2) Install fitting and air line on back of gage.
(3) Service accumulator with nitrogen or compressed air.
(4) Depress DOOR CONT circuit breaker and charge accumulator.
(5) Remove safety devices as required.

3. **Brake Accumulator Gage Restrictor — Removal / Installation**
A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.

(2) Gain access to nose wheel well in vicinity of parking brake selector valve, place a container suitable for hydraulic fluid under work area.
(3) Pull DOOR CONT circuit breaker and placard to prevent auxiliary pump from operating.
(4) Bleed pressure from system by operating PARK / EMER brake selector and observe accumulator pressure gage; when needle on gage stabilizes, pressure has been bled, needle should stabilize at 800 ± 25 psi.
(5) Disconnect lines and fittings and remove restrictor.

B. Installation
(1) Discard old O-rings and replace with new ones.
(2) Install restrictor, fittings and lines.
(3) Depress DOOR CONT circuit breaker and remove placard.
(4) Using external power source, operate auxiliary pump to pressurize system.
(5) Operate PARK / EMER brake handle five to six cycles and check for leaks at restrictor fittings.
(6) Check fluid level in hydraulic reservoir and service if required.

AUXILIARY HYDRAULIC SYSTEM BRAKE ACCUMULATOR — MAINTENANCE PRACTICES

1. **Brake Accumulator – Service**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Chock wheels.

B. Operate PARK/EMER brake handle until brake pressure stabilizes.

NOTE: Pressure remaining is air pressure in accumulator. This must be 800 $\pm$ 25 psi.

C. If accumulator requires servicing, proceed as follows:

(1) Open landing gear doors to gain access to accumulator in nose wheel well.

(2) Ensure that 3/4 inch nut on accumulator filler valve is tight; then, remove valve cap.

(3) Connect nitrogen hose to filler valve. Loosen swivel nut on filler valve 3 1/2 turns.

(4) Slowly fill accumulator until pressure gage (in cockpit) reads 800 $\pm$ 25 psi.

(5) Tighten swivel nut on filler valve.

(6) Turn off nitrogen supply and remove hose.

(7) Install valve cap.

(8) Using auxiliary hydraulic pump, pressurize accumulator until pressure gage reads 1450-1500 psi on aircraft having ASC 137A. Those aircraft not having ASC 137A should read 1900 psi.

2. **Brake Accumulator Gage — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.

(2) Pull DOOR CONT circuit breaker and placard.

(3) Cycle PARK/EMER brake handle approximately eight times until needle on brake accumulator gage stabilizes.

WARNING: DO NOT LOOSEN VALVE HOUSING TO DISCHARGE NITROGEN. PRESSURE COULD EJECT VALVE ASSEMBLY AND INJURE PERSONNEL OR DAMAGE AIRCRAFT.

(4) Slowly open valve on accumulator air servicing port. Allow pressure to drop to zero before continuing.

NOTE: Accumulator air servicing port is located on forward side of nose wheel well.

(5) Disconnect air line from back of gage.

(6) Remove mounting screws. Remove Plexiglas shield and retain. Remove gage from underside of copilot instrument panel.

B. Installation

(1) Place gage in position on copilot instrument panel and install Plexiglas protector in front of gage face. Install mounting screws and tighten.

(2) Install fitting and air line on back of gage.

(3) Service accumulator with nitrogen or compressed air.

(4) Depress DOOR CONT circuit breaker and charge accumulator.

(5) Remove safety devices as required.

3. **Brake Accumulator Gage Restrictor — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Chock wheels.

(2) Gain access to nose wheel well in vicinity of parking brake selector valve, place a container suitable for hydraulic fluid under work area.

(3) Pull DOOR CONT circuit breaker and placard to prevent auxiliary pump from operating.

(4) Bleed pressure from system by operating PARK/EMER brake selector and observe accumulator pressure gage, when needle on gage stabilizes, pressure has been bled, needle should stabilize at 800 $\pm$ 25 psi.

(5) Disconnect lines and fittings and remove restrictor.

B. Installation

(1) Discard old O-rings and replace with new ones.

(2) Install restrictor, fittings and lines.

(3) Depress DOOR CONT circuit breaker and remove placard.

(4) Using external power source, operate auxiliary pump to pressurize system.

(5) Operate PARK/EMER brake handle five to six cycles and check for leaks at restrictor fittings.

(6) Check fluid level in hydraulic reservoir and service if required.

LANDING GEAR WHEELS AND TIRES — DESCRIPTION / OPERATION

1. Main Gear

Each main gear consists of two cast magnesium alloy wheels. Each wheel is designed for use with a 7.50 X 14, type III, 12-ply rating, nylon, tubeless tire, or a tire and tube. The wheel is the divided type, and halves are held together by bolts, washers and self-locking nuts. When used with a tubeless tire, an O-ring seal is installed to seal the mating surfaces of the wheel halves, and a special valve assembly is installed in the wheel. When either type of tire is used, the wheels are balanced. When a tubeless tire is used, the red dot on the tire should be placed at the valve stem of the wheel. When a tire and tube are used, the tube should be placed in the tire with the yellow stripe on the base of the tube aligned with the red dot on the tire. The main wheel tires must be inflated to 110 psi at 33,600 pounds takeoff gross weight or to 120 psi at 35,100 pounds take-off gross weight or 123 psi at 36,000 pounds takeoff gross weight. An air-nitrogen trailer should be used to service the tires. Use a low-pressure air gage to check tire inflation.

2. Nose Gear

The nose gear consists of two cast magnesium wheels. Each wheel is designed for use with a 6.50 x 8 type III, 6-ply rating, or 6.50 x 8,8-ply rating, nylon, tubeless tire, or a tire and tube. The wheel is the divided type, and the halves are held together by bolts, washers and self-locking nuts. When used with tubeless tires, an O-ring seal is installed to seal the mating surfaces of the wheel halves, and a special valve assembly is installed in the wheel. When either type tire is used, the wheels are balanced. When a tubeless tire is used, the red dot on the tire should be placed at the valve stem of the wheel. When a tire and tube are used, the tube should be placed in the tire with the yellow stripe on the base of the tube aligned with the red dot on the tire. An air-nitrogen trailer (0-3000 psi) should be used to service the tires. A low pressure air gage (0-100 psi) should be used to check tire inflation. The nose wheel tire should be inflated to 50 psi for all takeoff gross weights.

3. Main Wheel Nut and Washer Improvement

On Aircraft 100 - 200, 322 and 323 excluding 114 and Aircraft having ASC 155, the existing nuts (32FE-624), washers (9522043) and washers (9522047) have been replaced with new improved nuts (42FLW-624 or 48FLW-624) and washers (9525616 and 952617) respectively. (See Figure 201) The bolt heads on aircraft modified during production and by operators have been identified by painting the bolt heads yellow. The bolts are no different, the heads have been painted yellow to indicate that new bolts are installed in conjunction with the new part number nuts and washers. The new nuts and washers are improved types which have better load distributing qualities and improved strength.

NOTE: Nuts (42FLW-624 and 48FLW-624) are completely interchangeable and either nut is acceptable.

4. Main Wheel Improvement

On Aircraft 115 - 200 the wheel assembly (9542306) is the latest design and supersedes the other assemblies (9531942 and 9532515). The 9542306 assembly used 1/2 inch diameter bolts (GY188-40) and the earlier 9531942 and 9532515 assemblies use 3/8-inch diameter bolts (GY626-40 or GY186-40). All bolts use the Lubtork method of tightening to obtain the maximum efficiency from the fasteners. (See Figure 201.

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LANDING GEAR WHEELS AND TIRES — MAINTENANCE PRACTICES

1. **Main Wheel and Nose Wheel Tire — Inspection**

WARNING: USE DRY NITROGEN ONLY WHEN SERVICING TIRES.

A. Ensure pressure of main tires is 110 psig for 33,600 pounds takeoff gross weight or 120 psig for 35,100 pounds takeoff gross weight or 123 psig for 36,000 pounds takeoff gross weight (as per OPTIONAL ASC 175).

B. Any of the following conditions necessitates tire replacement:

- Deep cuts
- Tread wear
- Flat spots
- Heat blisters

C. Ensure nose wheel tire pressure is 50 psig for all takeoff gross weights.

2. **Main Landing Gear Wheel — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

PARTIAL JACKING OF AIRCRAFT FOR A SINGLE GEAR RETRACTION IS PROHIBITED. SERIOUS INJURY TO PERSONNEL AND EXTENSIVE DAMAGE TO AIRCRAFT COULD OCCUR.

(1) Chock wheels that are not to be jacked.

(2) Jack main gear at axle jack point.

(3) Set PARK/EMERG brake handle to lock brakes. Deflate tire completely. Remove valve core.

(4) Remove wheel.

WARNING: DO NOT DISASSEMBLE A WHEEL OR RETURN ONE FOR REPAIR UNLESS TIRE HAS BEEN DEFLATED AND VALVE CORE REMOVED.

(5) It is recommended at this time to determine if brake must be removed. If brake is serviceable continue with step B. If brake requires replacement, continue with brake removal/installation.

NOTE: It should be noted that commencing at the first tire change after 1000 landings and each 300 landings thereafter, disassemble wheel halves and check for corrosion or cracks by (NDT).

B. Installation

(1) Before installing wheel, check and retorque brake bolts.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(2) Clean axle shaft with approved solvent and inspect for corrosion, rust, deep scratches or excessive wear at bearing surfaces.

(3) Lubricate axle nut threads and bearing surfaces with oil or bearing grease.

(4) Ensure bearings have been properly packed with grease and that felt seal is lubricated with motor oil.

(5) Ensure inner bearing, seal, retainers and lockring have been installed properly.

(6) Install bearing spacer.

(7) Install wheel assembly, bearing, key washer and axle nut.

(8) Rotate wheel and torque axle nut to 300 inch-pounds (25 foot-pounds).

(9) Back off axle nut to zero torque, keeping bearing surfaces in contact. Torque axle nut to 120 inch-pounds (10 foot-pounds). If axle nut slot does not align with an axle hole, tighten nut until alignment can be made.

NOTE: On aircraft having ASC 206 ensure wheel generator holes are aligned and install units.

(10) Install lockbolt, washer and nut.

(11) Install dust cap and lockring.

(12) Inflate tires as follows:

WARNING: USE DRY NITROGEN ONLY WHEN SERVICING TIRES.

CAUTION: OVERINFLATION WILL ADD UNDUE STRESS TO WHEEL BOLTS.

(a) 110 pounds at 33,600 pounds gross weight

(b) 120 pounds at 35,100 pounds gross weight

(c) 123 pounds at 36,000 pounds gross weight

(13) Check alignment of decelostat with wheel mating surface.

(14) Lower jack and remove.

3. Main Gear Wheel — Inspection / NDT

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: Because of detailed instructions required in disassembly, inspection, and reassembly of main gear wheel, it is recommended that Aircraft Braking System Manual AP-114 with latest revisions be referenced when performing following services. Following is general procedure only.

A. Remove wheel assembly.

WARNING: DO NOT ATTEMPT TO DISASSEMBLE A WHEEL OR RETURN ONE FOR REPAIR UNLESS VALVE CORE HAS BEEN REMOVED AND TIRE HAS BEEN COMPLETELY DEFLATED.

B. Break beads and disassemble wheel. Remove tire.

C. Remove bearings from both wheel halves.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

D. Clean all metal parts and wash felt seals with cleaning solvent.

E. Inspect wheel halves for cracks, nicks or scratches, particularly in bead seat, bolt boss and valve areas. Nicks and scratches may be polished out. Inspect bearing cups for security of mounting and condition. Send wheel to overhaul for replacement of loose or damaged cups. Inspect bearings for smoothness of rotation and condition. Replace worn or damaged bearings. Inspect disc drive keys for security and wear; if loose, tighten. If worn to 0.740 inch or less in width, replace. Refer to Aircraft Braking System Manual AP-114 for repair instructions.

F. Inspect wheel casting for corrosion and other surface damage. Check carefully for corrosion on surfaces that contact the tire beads, in drive key slot and drive key slot relief groove areas. Refer to Aircraft Braking System Manual AP-114 for repair and limitations.

CAUTION: INSPECTION OF BEAD SEAT AREAS BY EDDY CURRENT OR ULTRASONIC METHOD IS THE BEST MEANS OF PINPOINTING LATENT CRACKS. HOWEVER, WHEN USED TO INSPECT CAST MAGNESIUM WHEELS, FALSE READINGS MAY BE OBTAINED. THEREFORE, IT IS NECESSARY TO INSPECT PINPOINTED AREAS BY OTHER METHODS SUCH AS ZYGLO OR EQUIVALENT BEFORE MAKING A FINAL DETERMINATION TO DISCARD WHEEL HALF.

NOTE: A thorough inspection of the main wheel flanges in the bead seat area is required periodically using authorized and specific inspection methods. Wheel is removed, disassembled and paint is stripped from bead area for this inspection.

In cases where conditions indicate a requirement, inspection may be conducted using Ultrasonic or Eddy Current methods (NDT) which has the advantage that the wheel need not be removed from the aircraft or the tire be dismounted. **This inspection is supplemental to and does not replace or defer scheduled required inspections.**

G. Inspect bead seat area by Eddy Current or Ultrasonic methods (NDT). Inspect all other areas of wheel by Zyglo or other dye penetrant methods (NDT). Remove all cracked wheel forgings from service.

H. If mounting a tubeless tire reuse serviceable O-ring and wash with denatured alcohol or replace with new O-ring. Before assembly coat O-ring with a light coating of grease MIL-G-7711.

I. Assemble wheel as recommended in Aircraft Braking System Manual AP-114 for tubeless or tube type tires as applicable.

J. Lubricate threads of bolts (new or magnafluxed) and bearing surfaces of nut, bolt head and washers with antiseize compound. Install bolts, new nuts and reusable washers.

K. Torque nuts to values specified in Figure 201 for type of wheel.

CAUTION: OVERINFLATION OF TIRES WILL ADD UNDUE STRESS TO WHEEL BOLTS.

L. Inflate tires as follows:

(1) 110 pounds at 33,600 pounds gross weight.
(2) 120 pounds at 35,100 pounds gross weight.
(3) 123 pounds at 36,000 pounds gross weight.
(4) If wheel is to be stored, do not allow more than 20 psi air pressure to remain in tire.

M. Properly pack, clean, serviceable bearings with grease.

NOTE: If wheel is to be stored, wrap bearings in oilproof paper and coat bearing cones with grease. Cover with oilproof paper.

N. Install wheel assembly.

NOTE: List the part number and serial number of the wheel that the nondestructive test (NDT) Zyglo or equivalent was accomplished on.

4. Main Landing Gear Tire — Removal / Installation

NOTE: Because of detailed instructions required in disassembly, inspection and reassembly of main gear wheel, refer to Aircraft Braking System Manual AP-114 with latest revisions when performing following services. Following is general procedure only.

A. Removal (See Figure 201)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Remove wheel assembly.

(2) Completely deflate tire by removing valve core.

WARNING: DO NOT ATTEMPT TO DISASSEMBLE A WHEEL OR RETURN ONE FOR REPAIR UNLESS TIRE HAS BEEN DEFLATED AND VALVE CORE HAS BEEN REMOVED.

USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(3) Clean axle with cleaning solvent.

(4) Inspect axle shaft for corrosion, rust, deep scratches or excessive wear at bearing surfaces.

(5) Disassemble wheel assembly as follows or obtain a replacement tire and wheel assembly.

CAUTION: WHEEL SHOULD BE DISASSEMBLED ON A CLEAN FLAT SURFACE TO PREVENT SCRATCHING, GOUGING OR NICKING OF WHEEL HALVES. DO NOT PRY BETWEEN BEAD AND FLANGE WITH SHARP TOOLS, OR WHEEL AND TIRE MAY BE DAMAGED, DESTROYING THEIR SEALING AND STRUCTURAL QUALITIES.

(6) Break tire beads away from both wheel flanges by applying even pressure around entire sidewall as close to tire beads as possible.

(7) Remove bearings from both wheel halves.

(8) Remove nuts, washers, bolts and washers, separate wheel halves, remove tire.

NOTE: If necessary to reuse wheel O-ring, mark installation position in wheel and carefully place O-ring on a flat, clean surface to avoid damage or stretching.

(9) Carefully remove wheel O-ring seal from wheel half.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(10) Clean all metal parts and wash felt seals with cleaning solvent.

(11) Inspect wheel halves for cracks, nicks or scratches. Nicks and scratches may be polished out. Inspect bearing cups for security of mounting and condition. Send wheel to overhaul for replacement of loose or damaged cups; inspect bearings for smoothness of rotation and condition. Replace worn or damaged bearings. Inspect disc drive keys for security and wear, if loose or worn to 0.740 inch or less in width, refer to Aircraft Braking System Manual AP-114 for repair instructions.

(12) Visually check wheel for signs of corrosion or cracks at each tire change.

(13) Perform inspection (NDT) using Ultrasonic or Eddy Current and Zyglo or equivalent methods, depending on condition and requirement.

B. Installation

(1) If mounting a tubeless tire, reuse serviceable O-ring and wash with denatured alcohol or replace with new O-ring. Before assembly, coat O-ring with light coating of grease MIL-G-7711.

CAUTION: ASSEMBLE WHEEL ON A CLEAN FLAT SURFACE THAT WILL NOT NICK OR SCRATCH WHEEL HALVES.

(2) Assemble wheel as recommended in Aircraft Braking System Manual AP-114 for tubeless or tube type tire as applicable.

(3) Lubricate threads of bolts (new or magnafluxed) and bearing surfaces of nut, bolt head and washers with anti-seize compound. Install bolts, new nuts and reusable washers.

NOTE: Main gear wheel bolts, GY626-40 or GY186-40 and GY188-40 must be replaced or magnafluxed at each tire change.

(4) Torque nuts to values specified in Figure 203 for type wheel used.

(5) Lubricate axle, threads and felt retainers with oil.

(6) Ensure that wheel bearings have been properly inspected and packed with grease before installation.

(7) Install bearings and wheel assembly.

(8) Inflate tires as follows:

(a) 110 pounds at 33,600 pounds gross weight
(b) 120 pounds at 35,100 pounds gross weight
(c) 123 pounds at 36,000 pounds gross weight

(9) If tire and wheel assembly is to be stored, do not allow more that 20 psi air pressure to remain in tire.

5. **Main Landing Gear Wheel Bolt — Retorque**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR SAFETY DEVICES ARE INSTALLED.

A. Remove wheel assembly.

B. Retorque wheel bolts as follows:

(1) Wheel P/N 9531942 or 9532515, Lubtork to 300 inch-pounds (25 foot-pounds).

(2) Wheel P/N 9542306, Lubtork to 480 inch-pounds (40 foot-pounds).

CAUTION: IF ONE BOLT IS FOUND BROKEN, REPLACE ALL EIGHT WHEEL BOLTS AND NUTS.

IF INCORRECT TYPE WASHERS ARE FOUND INSTALLED ON A WHEEL ASSEMBLY, REPLACE ALL BOLTS, NUTS AND WASHERS WITH NEW PARTS.

IF INCORRECTLY INSTALLED WASHERS ARE FOUND ON A WHEEL ASSEMBLY, REPLACE BOLTS AND NUTS AFFECTED BY THESE WASHERS WITH NEW PARTS.

(3) Install wheel assembly.

6. **Main Landing Gear Wheel Bearings — Service**

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove wheel.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAME.

B. Clean, inspect and repack wheel bearings. Clean bearings with dry cleaning solvent and dry thoroughly using dry, filtered, compressed air.

C. Install wheel.

7. **Nose Landing Gear Wheel — Removal / Installation**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Jack nose gear at axle jack point.

(2) Completely deflate tire and remove valve core.

WARNING: DO NOT ATTEMPT TO DISASSEMBLE A WHEEL OR RETURN ONE FOR REPAIR UNLESS TIRE HAS BEEN COMPLETELY DEFLATED AND VALVE CORE HAS BEEN REMOVED.

(3) Remove lockring that secures axle nut to axle.

(4) Remove axle nut. Remove keyed washer.

(5) Remove outer bearing and spacer from wheel.

(6) Place a protective covering over axle threads and remove wheel from axle, taking care not to damage axle and inner bearing.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(7) Clean axle shaft with cleaning solvent.

(8) Inspect axle shaft and spacers for corrosion, rust, deep scratches or excessive wear at bearing surfaces.

NOTE: It should be noted that commencing at the first tire change after 1500 flights, disassemble wheel halves and inspect for corrosion or cracks by (NDT); repeat at every other tire change. After 2000 flights, repeat inspection at every tire change.

Nose gear uplock roller bolt may be inspected at this time if desired.

Nose strut friction damper torque check may be accomplished at this time on aircraft having ASC168.

B. Installation

(1) Lubricate axle nut threads and bearing surfaces with oil or bearing grease.

(2) Ensure that bearings have been properly packed with grease, and that felt seals are lubricated with oil.

(3) Ensure that wheel bearings, seal, and retainers have been installed properly and that retainer screws have been safetied.

(4) Install inboard bearing spacer.

(5) Place a protective cover over axle threads and install wheel on axle using care not to damage axle or inner bearing.

(6) Install outboard bearing spacer, key washer and nut.

(7) Rotate wheel and torque axle nut to 300 inch-pounds (25 foot-pounds).

(8) Back off axle nut to zero torque keeping bearing surfaces in contact. Torque axle nut to 120 inch-pounds (10 foot-pounds). If axle nut is not aligned with locking location, tighten nut to next locking location.

(9) Install lockring.

(10) Inflate tire to 50 psi (all takeoff gross weights).

(11) Check wheels for missing balance weights and weights for security. Replace if required, ensuring weights are equally divided on each wheel half if possible.

(12) Lower jack and remove.

8. Nose Landing Gear Wheel Bolt — Torque Check

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove wheel assembly.

B. Retorque (lubtork) wheel bolts to 80 inch-pounds.

CAUTION: IF ONE BOLT IS FOUND BROKEN REPLACE ALL SIX WHEEL BOLTS AND NUTS.

C. Install wheel assembly.

9. Nose Landing Gear Tire — Removal / Installation

NOTE: Because of detailed instructions required in disassembly, inspection and reassembly of nose gear wheel, it is recommended that personnel refer to Aircraft Braking System Manual AP-114 when performing following services. Following is a general procedure only.

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Jack nose gear.

(2) Remove nose wheel.

(3) Completely deflate tire by removing valve core.

WARNING: DO NOT ATTEMPT TO DISASSEMBLE A WHEEL OR RETURN ONE FOR REPAIR UNLESS VALVE CORE HAS BEEN REMOVED AND TIRE HAS BEEN COMPLETELY DEFLATED.

CAUTION: WHEEL SHOULD BE DISASSEMBLED ON A CLEAN FLAT SURFACE TO PREVENT SCRATCHING, GOUGING OR NICKING WHEEL HALVES. DO NOT PRY BETWEEN FLANGE AND BEAD WITH SHARP TOOLS OR WHEEL AND TIRE MAY BE DAMAGED, DESTROYING THEIR SEALING AND STRUCTURAL QUALITIES.

(4) Break beads away from flanges by applying even pressure around sidewall adjacent to flange.

(5) Disassemble wheel and remove tire.

(6) Remove bearings from both wheel halves.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(7) Clean all metal parts and wash felt seals with cleaning solvent.

(8) Inspect wheel halves for cracks, nicks or scratches. Nicks and scratches may be polished out. Inspect bearing cups for security of mounting and condition. Send wheel to overhaul for replacement of loose or damaged cups. Inspect bearings for smoothness of rotation and condition. Replace worn or damaged bearings.

B. Installation

(1) If mounting tubless tire, reuse servicable O-ring and wash with denatured alcohol or replace with new O-ring. Before assembly, coat O-ring with light coating of grease, MIL-G-7711.

(2) Assemble wheel as recommended in Aircraft Braking System Manual AP-114 for both tubeless or tube type tires as applicable.

(3) Install new or magnafluxed bolts and new nuts. Washers may be reused. First lube bolt threads and bearing surfaces of nuts, bolt heads and washers with antiseize compound.

(4) Lubtork all nuts to 80 inch-pounds in increments of 20 inch-pounds. Follow a crisscross pattern in torquing wheel bolts.

(5) Inflate all tires (tube and tubeless type) to 50 psi. If tire is to be stored, do not allow more than 25% air pressure to remain in tire (pressure not to exceed 20 psi).

(6) Properly pack, clean serviceable bearing with grease, MIL-G-81322A and install in wheel.

(7) Lubricate felt seals with oil.

(8) Install nose wheel.

10. Nose Gear Wheel — Inspection (NDT)

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: Because of detailed instructions required in disassembly, inspection, and reassembly of nose gear wheel, it is recommended that personnel refer to Aircraft Braking System Manual AP-114 when performing following services. Following is a general procedure only.

A. Jack nose gear.

B. Remove nose wheel.

C. Completely deflate tire by removing valve core.

WARNING: DO NOT ATTEMPT TO DISASSEMBLE A WHEEL OR RETURN ONE FOR REPAIR UNLESS VALVE CORE HAS BEEN REMOVED AND TIRE HAS BEEN COMPLETELY DEFLATED.

CAUTION: WHEEL SHOULD BE DISASSEMBLED ON A CLEAN, FLAT SURFACE TO PREVENT SCRATCHING, GOUGING OR NICKING WHEEL HALVES. DO NOT PRY BETWEEN FLANGE AND BEAD WITH SHARP TOOLS OR WHEEL AND TIRE MAY BE DAMAGED, DESTROYING THEIR SEALING AND STRUCTURAL QUALITIES.

D. Break beads and disassemble wheel. Remove tire.

E. Remove bearings from both wheel halves.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

F. Clean all metal parts and wash felt seals with cleaning solvent.

G. Inspect wheel halves for cracks, nicks or scratches. Nicks and scratches may be polished out. Inspect bearing cups for security or mounting and condition. Send wheel to overhaul for replacement of loose or damaged cups. Inspect bearings for smoothness of rotation and condition. Replace worn or damaged bearings.

H. Perform Zyglo or equivalent check of wheel halves. If any cracks are found, discard wheel half.

I. If mounting tubeless tire reuse serviceable O-ring and wash with denatured alcohol or replace with new O-ring. Before assembly coat O-ring with light coating of grease, MIL-G-7711.

J. Assemble wheel as recommended in Aircraft Braking System Manual AP-114 for tubeless or tube type tires as applicable.

K. Install new or magnafluxed bolts and new nuts. Washers may be reused. First lube bolt threads and bearing surfaces of nuts, bolt heads, and washers with antiseize compound.

L. Lubtork all nuts to 80 inch-pounds in increments of 20 inch-pounds. Follow crisscross pattern in torquing wheel bolts.

WARNING: PLACE WHEEL IN AN INFLATION CAGE FOR INITIAL INFLATION. DO NOT INFLATE TIRE IN EXCESS OF FULL OPERATING PRESSURE. TIRE AND/OR WHEEL FAILURE MAY OCCUR, CAUSING INJURY TO PERSONNEL OR DAMAGE TO EQUIPMENT IF TIRE IS INFLATED FROM ANY HIGH PRESSURE SOURCE. TIRE AND WHEEL ASSEMBLIES MUST BE SERVICED WITH INFLATION EQUIPMENT THAT HAS BEEN SPECIFICALLY DESIGNED FOR THIS OPERATION.

NOTE: It is recommended tires mounted on dual wheels have similar inflated outside diameters to ensure that each tire will carry its equal share of the load. The outside diameter of new tires (new or retread) should be measured at operating pressure and pairs matched within the tolerances prescribed by specification of manufacturer of tire being installed.

M. Inflate all tires (tube and tubeless type) to 50 psi. If tire is to be stored, do not allow more than 25% air pressure to remain in tire (pressure not to exceed 20 psi).

N. Properly pack, clean serviceable bearings with grease, MIL-G-81322A or MIL-G-3545A and install in wheel.

NOTE: If wheels are to be stored, cover wheel hubs with oilproof paper to prevent contamination of grease by foreign matter, dust or moisture.

O. Install nose wheel.

NOTE: List the serial number and part number of the wheel that the nondestructive test (NDT) Zyglo 1or equivalent was accomplished on, for further reference.

11. Nose Landing Gear Wheel Bearings — Service

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Remove wheel.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAME.

B. Clean, inspect and repack wheel bearings. Clean bearings with cleaning solvent and dry thoroughly using dry, filtered, compressed air.

C. Install wheel.

12. Wheel Balance Weight Retainers — Installation

CAUTION: TIRE BALANCE DOT MUST BE DIRECTLY ABOVE VALVE STEM BEFORE ANY BALANCING.

NOTE: The following procedure provides a more positive method of retaining the nose wheel balance weights to the wheel.

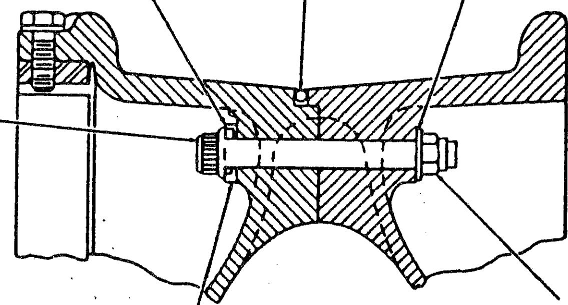
A. Drill 16 equally spaced holes of 5/64 inch diameter with centerline 0.188 inch from face of rim radially through rim. (See Figure 202) In order to accomplish best spacing, holes on wheel half 9524008 should straddle valve while other wheel half should have holes spaced halfway between those of mating half.

B. Treat the machined areas of the wheels with cold spot Dow treatment (Chrome Pickle Touch-Up Type #1 per G.S.S. Spec 9.012 II can be used as an alternate), and zinc chromate primer (use Finish #1012 as an alternate). Dow treatment #19 can also be used.

C. Rework A9525155 balance weights by drilling 3/32 inch diameter hole located 0.188 inch from the rim face contact of the clip. Break sharp edges of the hole. Reidentify the weight as a 9525538 part.

D. In balancing the wheel and tire combination, install balance weights as required at cotter pin holes to accomplish static and dynamic balance and secure weights with MS24665-134 cotter pin.

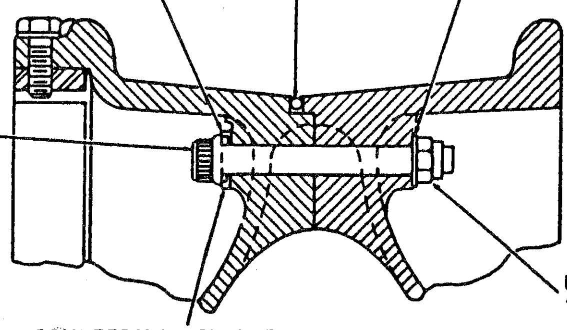
E. Install cotter pin from the outer flange toward the bearing. Divide weights equally on each wheel half if possible. To prevent corrosion, paint cotter pin, balance weight and rim area.



NOTE:
1. NUTS TO BE ON OUTSIDE OF WHEELS.
2. "LUBTORK" NUTS, BOLTS AND WASHERS TO 300 IN. LBS PER INSTRUCTIONS IN GOODYEAR WHEEL AND BRAKE MANUAL AP-114.

*GOODYEAR PART NUMBERS

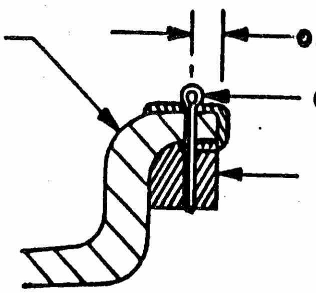
WHEEL BOLT INSTALLATION-CROSS-SECTIONAL VIEW
GOODYEAR MAIN WHEEL PART NOS. 9531942 OR 9532515



NOTE:
1. * 9542306 WHEEL INSTALLED AS ORIGINAL EQUIPMENT ON AIRCRAFT 115 AND SUBSEQUENT
2. NUT TO BE ON OUTSIDE OF WHEELS.
3. LUBTORK NUTS, BOLTS AND WASHERS TO 480 IN. LBS PER INSTRUCTIONS IN GOODYEAR WHEEL AND BRAKE MANUAL AP-114.

*GOODYEAR PART NUMBERS

Wheel Bolt Installation — Cross Section View
Figure 201.



Nose Wheel Balance Weight Retainer — Installation
Figure 202.

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ANTI-SKID CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

(See Figure 1 and Figure 2)

NOTE: The following information applies to aircraft having ASC 206.

A. The anti-skid control system consists of a control box, four wheel driven generators, four drive caps, four control valves (two dual valve assemblies), with associated aircraft interconnecting wiring and hydraulic lines.

B. The anti-skid control system is designed to function as a subsystem of the main landing gear and brake system. The purpose of the anti-skid control system is to monitor wheel deceleration and to control brake pressure so that safe, reliable and efficient braking is achieved under tire-to-ground conditions without allowing wheel skids.

2. Drive Caps and Wheel Generators

The wheels are mechanically coupled to the wheel generators by the drive cap assemblies. The wheel generators are DC tachometer generators that provide a voltage directly proportional to the speed of the wheel. The output voltage is used in the various circuits as a signal to control braking and to prevent skidding.

3. Control Box

The control box subassembly physically divides the skid control functions into inboard and outboard systems by using two identical plug-in circuit boards. Each circuit board provides individual skid control for a left and a right wheel. In addition, each board contains a locked-wheel prevention circuit, a touchdown protection circuit, a fail-safe circuit, and a functional check circuit. The anti-skid control box is located on a shelf in the radio rack.

A. Touchdown Protection Circuit

The touchdown protection circuit protects the wheels from inadvertent brake application before wheel spin-up at touchdown or during a bounce after touchdown. The right gear nutcracker relays control the outboard circuits and the left gear nutcracker relays control the inboard circuits. Before touchdown, the nutcracker relays are in the air position, driving the control valve amplifiers to full power, energizing the control valves and removing all brake pressure. After touchdown, when a load has been applied to the landing gear, the nutcracker relays return to the "ground" position, deactivating the touchdown protection circuit.

B. Anti-Skid Control Circuit

At touchdown, the wheel generators spin up and supply a voltage to the deceleration detector, the locked wheel detector, and the locked wheel arming circuit. During a normal stop (without skids), the wheel generator output voltage will decrease at a rate that will not exceed the predetermined deceleration threshold value. Pressure will be applied to the brake in accordance with the pedal force applied.

When excessive brake pressure is applied, an incipient skid occurs. The deceleration threshold value is then exceeded and a signal is sent to the control valve amplifier, which causes the brake pressure to be reduced in proportion to the amount by which the threshold is exceeded. As the speed of the aircraft is being reduced, the threshold value is automatically adjusted to a level that will continuously maintain the braking pressure near the skid threshold point.

C. Locked-Wheel Prevention Circuit

The locked-wheel prevention circuit dumps brake pressure when the wheel speed falls below 13 mph and the aircraft speed is above 22 mph. The aircraft speed reference is established by the output voltage of both the right and left inboard wheel generators. At aircraft speeds below 22 mph, the locked-wheel prevention circuit becomes inactive, allowing conventional operation during taxing or parking.

D. Fail.Safe Circuits

The inboard and outboard fail-safe circuits are functionally independent of each other, and a failure in either the inboard or outboard skid control circuits will light the Anti-skid (red) portion of the indicator switch on the center instrument panel and automatically remove power from the failed circuit. The failsafe circuits operate only when the aircraft is on the ground.

E. Functional-Check Circuits

The functional-check circuit provides a means to test the skid control circuits prior to takeoff or landing. The test switch (spring-loaded to the center OFF position) provides simulated wheel-speed signals when moved to either the right or left momentary position. This tests the operation of each side of the aircraft with simulated signals. If either the inboard or outboard control valves of the side under test fail to operate, the test light (green) will not light. If, during the check of one side of the aircraft, a signal is present on either of the valves of the side not being tested, the test light will not light.

4. Control Valves

The control valve assemblies are dual two-stage hydraulic valves, which are made up of two valves mounted on a common manifold. Each valve assembly controls the hydraulic pressure for the brakes on one side of the aircraft. Both halves of the valves are functionally the same, with one half controlling the inboard brake and the other half controlling the outboard brake. In each valve half, an electrical solenoid drives the first stage, which is the control stage. The first stage supplies a control pressure inversely proportional to the solenoid signal to the second stage, which then applies the appropriate controlled pressure to the brakes.

5. Controls and Indicators

The main wheel brake anti-skid system controls and indicators consist of a dual-indicator switch located on the center instrument panel, and a system test switch and light, located on the nose wheel steering panel.

A. Dual Indicator Switch

(1) The push on, push off, dual indicator switch provides a means of turning the anti-skid system on and off.

(2) The amber A/S OFF indicator, when on, indicates all of the following conditions have been met:

(a) The push on, push off switch is in the off condition and no power is being applied to the inboard and outboard anti-skid functions.

(b) The landing gear control handle is in the DOWN position.

(c) The essential dc bus is powered.

(3) The red ANTI-SKID indicator when on, indicates the switch is in the on condition (A/S indicator out) the landing gear control is in the DOWN position, and one or more of the following conditions has been met:

(a) OUTBOARD ANTI-SKID circuit breaker open or defective.

(b) IN BOARD ANTI-SKID circuit breaker open or defective.

(c) Failure of either power supply in the control box to maintain regulated voltage.

(d) Any anti-skid control valve in full dump condition for 2.5 seconds.

(e) Open electrical circuit in anti-skid control valve for 2.5 seconds with aircraft on ground.

(f) Failure of either nutcracker switch or nutcracker relay so that conflicting indications (airborne and touchdown) exist for 2.5 seconds after wheel speed is below 13 mph.

(g) A short to ground in any anti-skid control valve.

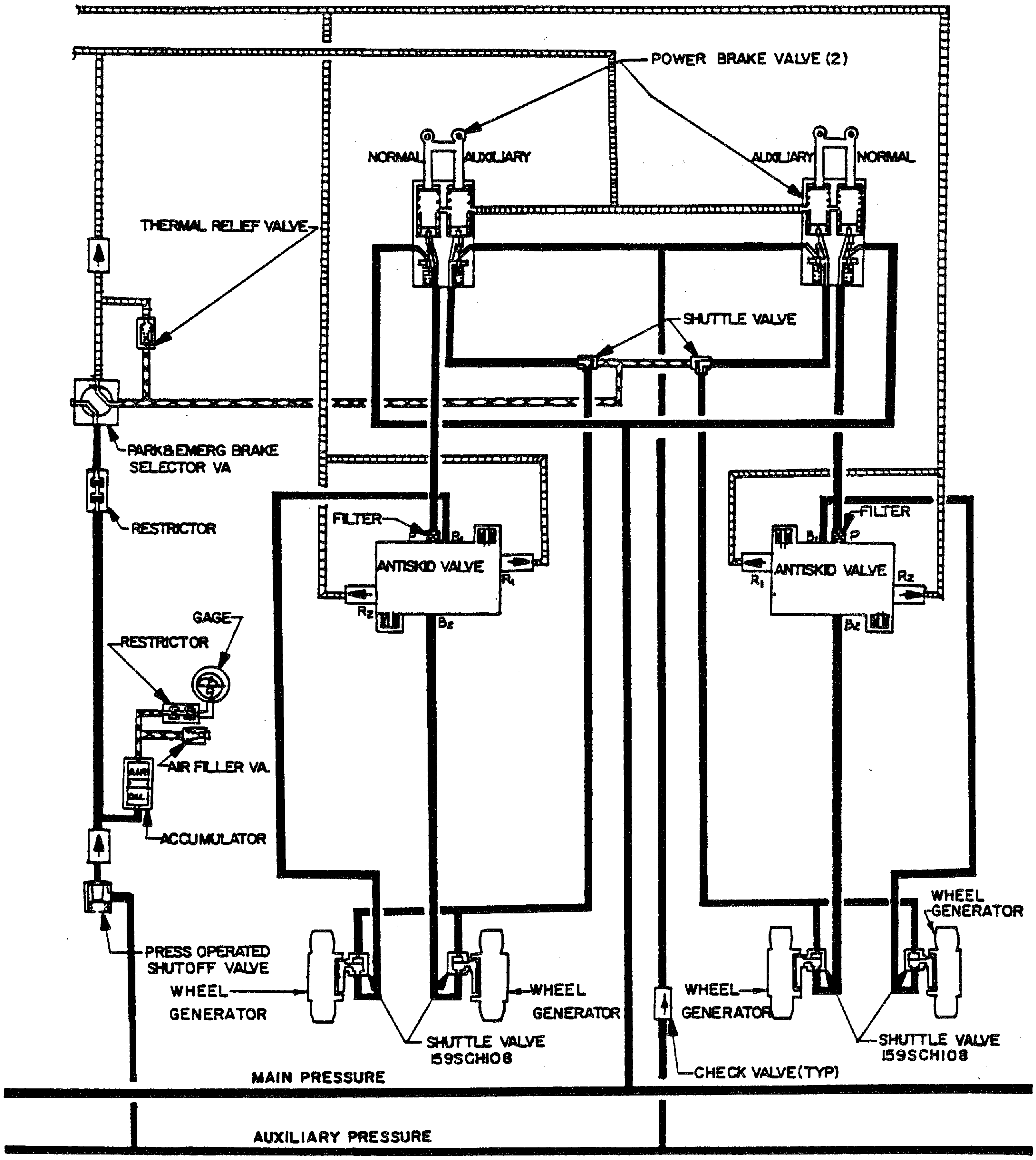
B. Anti-Skid Test Switch and Indicator Light

(1) The ANTI-SKID TEST switch is located immediately forward of the nose wheel tiller bar on the nose wheel steering panel (pilots side console). The green test light is located immediately to the right of the ANTI-SKID TEST switch. The test switch is a toggle type switch, spring-loaded to the center (off) position. The switch is used to initiate an electrical self-test of the main wheel brake anti-skid

system when the aircraft is stationary, or may be used to simultaneously check braking functions when taxiing at ground speeds not to exceed 10 mph.

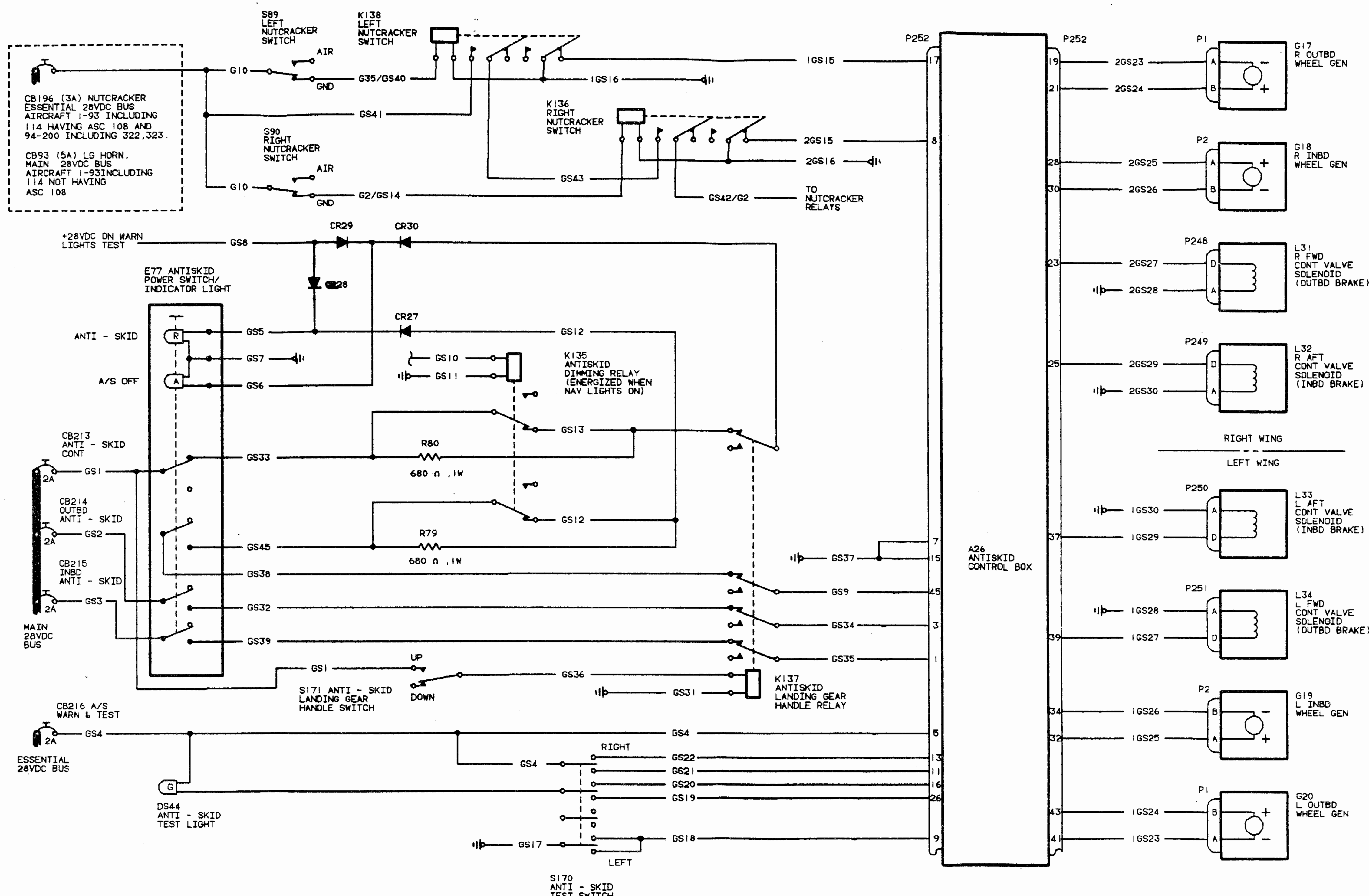
(2) Holding the ANTI-SKID TEST switch in the L (left) or R (right) position causes input from the opposite inboard and outboard wheel generators to be masked with simulated 40 mph (4 volts direct current) wheel speed signal. Since no corresponding wheel speed signal is being received from the opposite inboard and outboard wheel generators, a simulated locked-wheel condition is sensed by the control box. Proper operation of the system results in a full dump signal being sent to the opposite anti-skid control valves. The presence of proper full dump signal voltages cause voltage level detector circuits to turn the green light on, indicating proper operation.

NOTE: The test switch can be used during taxi operations to check out the anti-skid system. When taxiing moderate brake pressure should be applied in conjunction with test switch (i.e. test switch left, left brake pedal depressed, test switch right, right brake pedal depressed) to avoid aircraft swerving.



Legend	
(solid line)	Pressure
(ladder line)	Return
(cross-hatched line)	Pneumatic

Anti-Skid Hydraulic — Schematic
Aircraft Having ASC 206
Figure 1.



Anti-Skid System — Schematic
Figure 2.

NOTE: This schematic for Aircraft having ASC 206.

ANTI-SKID CONTROL SYSTEM — FAULT ISOLATION

1. **General**

Most malfunctions or component failures are indicated by the red ANTI-SKID indicator light coming on or by the green anti-skid test light failing to come on within five seconds after the ANTI-SKID TEST switch is moved to the L or R position. The following procedures indicate methods for isolating the trouble by using the ANTI-SKID TEST switch and other equipment on the aircraft. Fault isolation is limited to the anti-skid system, presuming the aircraft brake and hydraulic systems, exclusive of the control valves, are operating within normal limits.

2. **Operational Test Using Cockpit Test Switch**

The ANTI-SKID TEST switch provides the means of obtaining a GO-NO GO indication of system performance, with the green light giving a GO indication. The fault isolating procedures outlined in Figure 101 through 105 may be used to further isolate the fault to a replaceable component or assembly, or to indicate what further testing is required to isolate the fault.

A. Preparation

(1) Connect a 28V dc power source to aircraft. On center overhead panel, place the EXT PWR switch in the ON position and ANTI-SKID control switch to ON.

(2) On the left circuit breaker panel, ensure the following circuit breakers are depressed:

ANTI-SKID CONT	A/S WARN & TEST
OUTBD ANTI-SKID	NUTCRACKER
INBD ANTI-SKID	

(3) The landing gear control should be in the DOWN position, all landing gear extended and the aircraft weight resting on the wheels.

(4) On the eyebrow panel, depress and hold the WARN LTS TEST switch, ensuring the amber A/S OFF and red ANTI-SKID indicators on the center instrument panel come on. Release the switch, Red ANTI-SKID indicator should go off.

(5) If A/S OFF indicator remains on, press and release it. A/S OFF indicator should go out.

(6) Test and warning lights do not operate as specified above, a complete system check is required using Gulfstream Test Box 11595EAV20153 or Aircraft Braking System Test Box 9560835 to isolate defective components. Instructions are supplied with test units.

B. Control Box and Indicator Switch Test

(1) Perform fault isolating procedures in Figure 2.

(2) Failure See of A/S OFF indicator to go off indicates a defective indicator switch.

(3) Failure of green test light to come on in both tests indicates a defective indicator. The possibility that both inboard and outboard anti-skid functions have failed simultaneously is remote, but not impossible.

(4) If green test light fails to come on in only one test, proceed as follows

(a) Turn all aircraft electrical power off and disconnect external power from aircraft.

(b) Remove control box chassis in accordance with maintenance instructions. (See this section)

CAUTION: HANDLE CIRCUIT BOARDS CAREFULLY TO AVOID DAMAGE TO BOARDS OR CONNECTORS.

(c) Exchange positions of the circuit boards on control box chassis.

(d) Install control box chassis in proper position and secure with attaching hardware.

(e) See Figure 2 and repeat fault isolating procedures, beginning at RE-ENTER.

(f) If the fault (green test light fails to come on) remains on the same side, the control box is probably operating within normal limits. Perform a wheel generator test (see Paragraph C. below) and a control valve test (see Paragraph E. below).

(g) If the fault (green test light fails to come on) moves to the other side, a defective circuit board is indicated. Substitute serviceable circuit board as required to eliminate fault.

C. Wheel Generator Test (Tweak Test)

(1) Remove retaining rings and drive caps from wheels.

(2) Place anti-skid power switch ON.

(3) On center overhead panel, place EXT PWR switch ON.

(4) Pressurize main hydraulic system.

NOTE: The brake release can be verified by watching for corresponding movement of the brake puck as the rudder pedal toe brakes are applied, then return to former position as brake is released by the anti-skid control system.

(5) Perform fault isolating procedures as outlined in Figure 101.

(6) Replace defective components.

D. Locked-Wheel Test

(1) Perform fault isolating procedures as outline in Figure 102.

(2) If associated brake holds during spin-up, wheel generator or control valve may be defective.

(3) If red ANTI-SKID warning light does not come on after 2.5 seconds, control unit may be defective.

(4) Press A/S OFF - ANTI-SKID indicator switch twice to restore system to normal operation.

E. Control Valve Test

(1) Apply and hold full braking pressure to both toe brakes.

(2) On Aircraft 1 - 93 and 114 not having ASC 108A Part I, pull LG HORN circuit breaker. On Aircraft 1 - 93 and 114 having ASC 108A Part I, and Aircraft 94 - 200, 322 and 323, pull NUTCRACKER circuit breaker.

(3) All four wheel brakes should disengage.

(4) If both inboard or both outboard brakes fail to release, exchange circuit cards in control box in accordance with Steps B.(4)(a) thru B.(4)(d) above.

(5) Repeat Steps E.(1) through (3) above. If other pair of wheel brakes now fail to release, a defective circuit card is indicated. If same pair of brakes fail to release, nutcracker switches and relays may be defective or improperly adjusted.

(6) Depress circuit breaker pulled in Step (2) above.

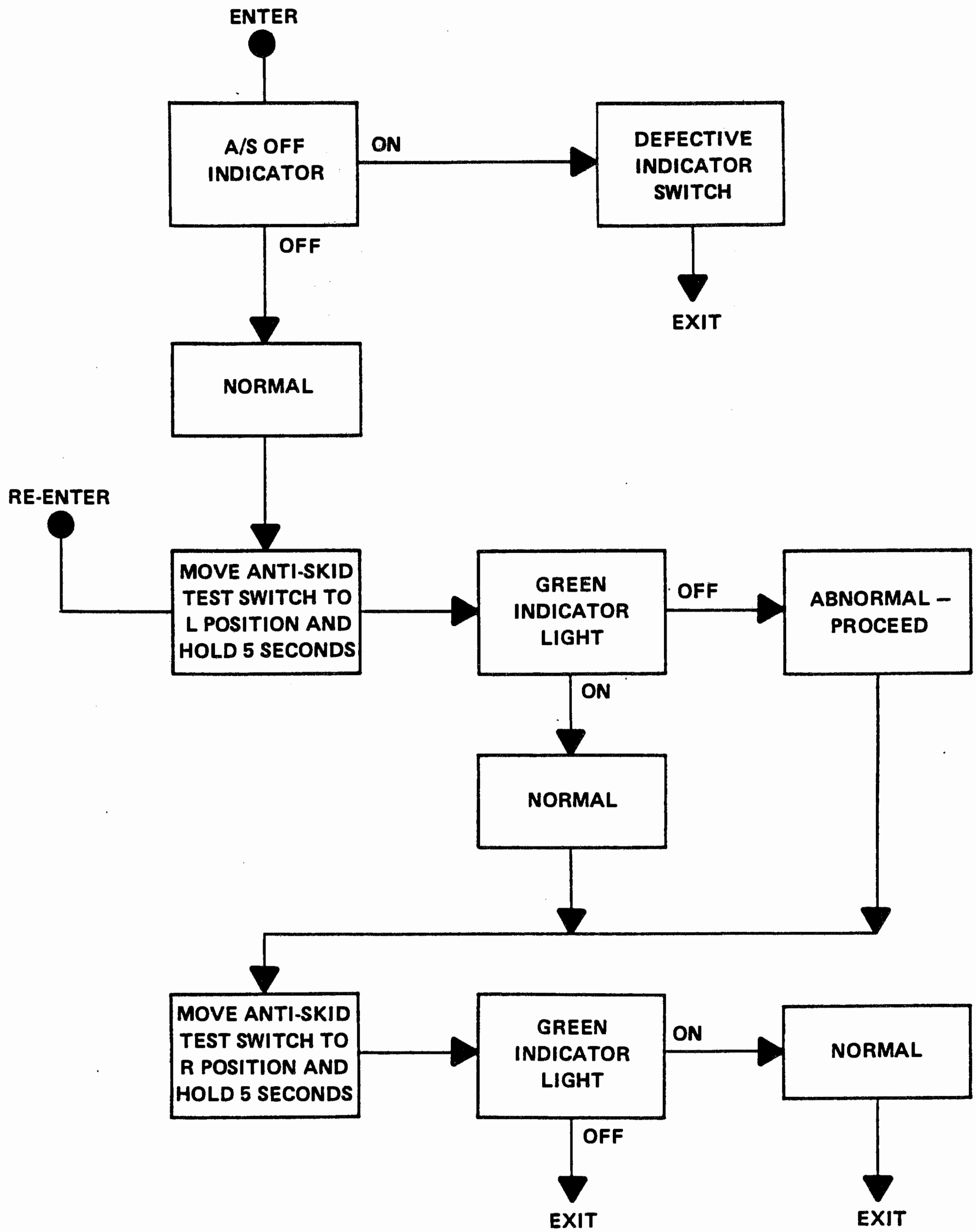
(7) Apply and hold full braking pressure to both toe brakes.

(8) Perform fault isolating procedures as outlined in Figure 103 using both nutcracker switches in turn.

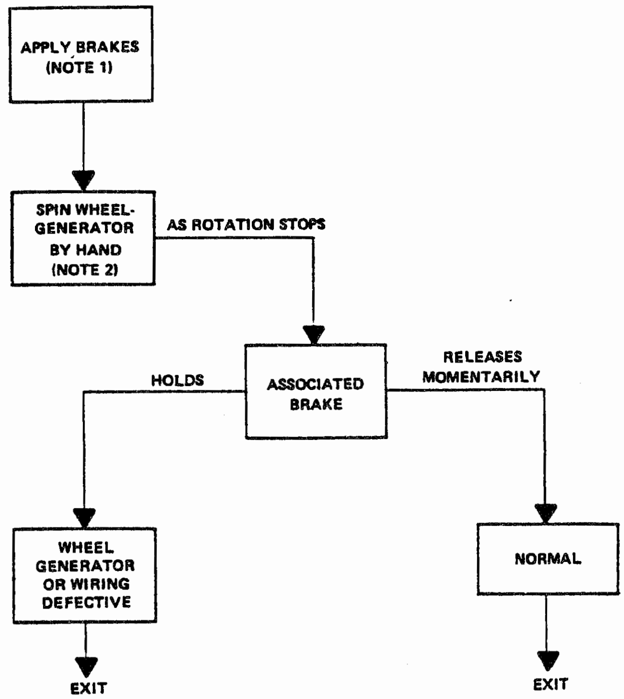
(9) Failure of brakes to release or red ANTI-SKID warning light to come on indicates problems in associated control box circuit card, or in nutcracker system.

(10) On center instrument panel, press and release A/S OFF - ANTI-SKID indicator switch (A/S OFF turn on).

(11) On center overhead panel, place EXT PWR switch in OFF and disconnect external power from aircraft.



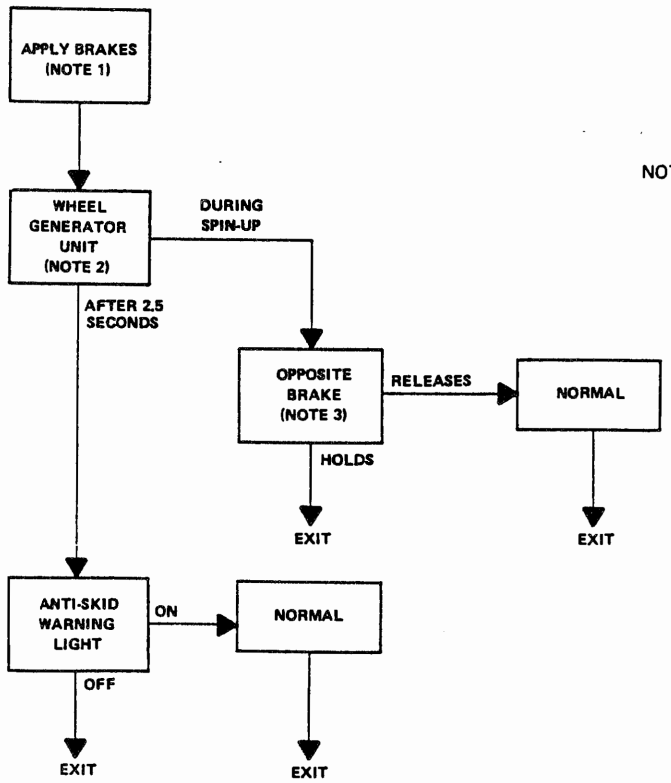
Control Box — Operational test
Figure 101.



NOTES:

1. USE SUFFICIENT PRESSURE TO SIMULATE A LOCKED-WHEEL CONDITION AND HOLD THROUGHOUT TEST.
2. A SPIN OF ONE-HALF TO ONE REVOLUTION WILL SUFFICE A CONTINUED SPIN OF AS MUCH AS 2.5 SECONDS MAY CAUSE THE RED ANTI-SKID LIGHT TO COME ON. SPIN UP IS FORWARD ROTATION OF WHEEL.

Wheel Generator Test
Figure 102.



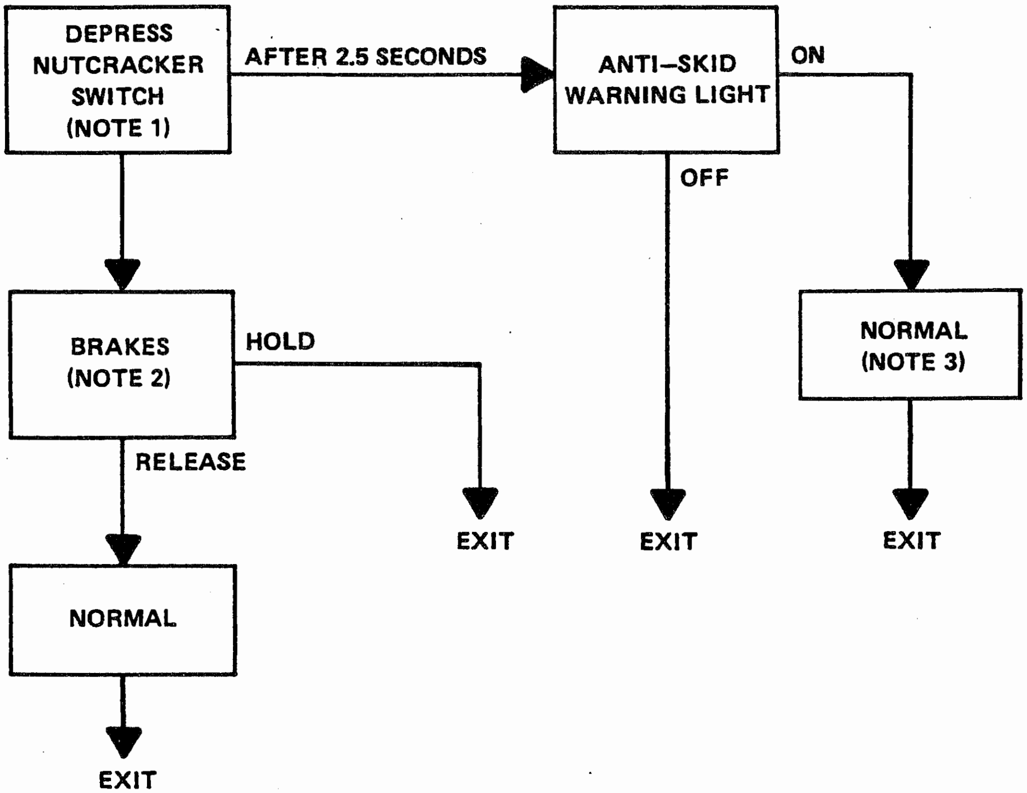
NOTES:

1. USE SUFFICIENT PRESSURE TO SIMULATE A LOCKED-WHEEL CONDITION AND HOLD THROUGHOUT TEST;

2. MAINTAIN AT LEAST 250 RPM FOR 3 SECONDS. A HAND CRANK WITH AN IMPROVISED COUPLING MAY BE USED.

3. BRAKE RELEASE SHOULD BE AS FOLLOWS:
   LEFT INBOARD BRAKE WITH RIGHT INBOARD DRIVE
   LEFT OUTBOARD BRAKE WITH RIGHT OUTBOARD DRIVE
   RIGHT INBOARD BRAKE WITH LEFT INBOARD DRIVE
   RIGHT OUTBOARD BRAKE WITH LEFT OUTBOARD DRIVE

Locked — Wheel Test
Figure 103.



NOTES:

1. EACH NUTCRACKER SWITCH IS LOCATED AT THE APEX OF THE LANDING GEAR TORSION LINKS, OR ON TOP OF THE CLEVIS OF THE UPPER TORSION LINK.

2. THE RIGHT NUTCRACKER SWITCH SHOULD RELEASE THE OUTBOARD BRAKES. THE LEFT NUTCRACKER SWITCH SHOULD RELEASE THE INBOARD BRAKES.

3. AFTER THE WARNING LIGHT COMES ON, RESET THE SYSTEM BY DEPRESSING THE A/S OFF - ANTI-SKID SWITCH TWICE.

Fail-Safe Test
Figure 104.

FAULT	PROBABLE CAUSE	REMEDY
Warning light comes on at turn on	Check pin 2 and 4 of control box if 28V dc present	Replace anti-skid control box
	If 28V dc not present	Determine cause of power loss
Warning light comes on 2.5 seconds after turn on	Incorrect resistance	Check pins 10,12,13,14 from pin to ground for 64 - 72 ohms with power OFF. If resistance is correct replace anti-skid control box.
		Check for wiring or valve at fault. Pin 10 RIB valve; Pin 12 LIB valve; Pin 13 LOB valve; Pin 14 ROB valve. Check resistance pins A-D64-72ohms isolated from ground.
		Check nutcracker switches in ground mode - Pins 27 and 31 should be at least 1 megohm above ground at test receptacle.
Warning light comes on during landing or taxi	Examine each drive cap and WST	Replace if damaged
	Check WST wiring	Repair or replace WST
		If fault repeats replace anti-skid control box.
Aircraft pulls to one side. Normal braking dry runway. Works OK with anti-skid off.	WST not working	Check brake temperature change coolest brake WST.
		If repeats change drive cap.
Flat spotted tire	Inoperative anti-skid control valve	Replace anti-skid control valve
Anti-skid warning light comes on when gear handle moved to down position.	28V dc present at pins 2 and 4 of anti-skid control box, if low voltage.	Check circuits back through gear handle and anti-skid power switch to circuit breakers.
	If voltage correct at Pins 2 and 4	Replace anti-skid control box
Temperature difference between brakes after hard or wet stop	Air in brake hydraulic system	Bleed brakes
Pulling to one side during hard or wet stop	Air in brake hydraulic system	Bleed brakes
Poor braking during hard or wet stop	Air in brake hydraulic system	Bleed brakes

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ANTI-SKID CONTROL SYSTEM — MAINTENANCE PRACTICES

1. **Drive Cap — Removal / Installation**

   (See Figure 201)

   A. Removal

      (1) Remove retaining ring from wheel.

      (2) Remove drive cap.

   B. Installation

      (1) Inspect drive cap yoke for excessive wear. If excessive wear is found replace drive cap.

      (2) Align drive cap yoke so it slides on wheel drive-shaft blade and into wheel casting.

      (3) Rotate drive cap so that drive cap wheel-lockpin seats into place.

      (4) Insert drive cap retaining ring into wheel casting and seat in retaining groove.

      NOTE: Check drive cap to ensure proper seating of drive cap lockpin.

2. **Main Landing Gear Wheel Generator — Removal / Installation**

   NOTE: The following applies to aircraft having ASC 206.

   A. Removal (See Figure 202)

   WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

      (1) Remove drive cap by removing retaining ring from wheel.

      (2) Remove attaching hardware securing wheel generator to axle.

      CAUTION: DO NOT PRY OR USE ANY TOOL ON THE DRIVE BLADE OR SHAFT THAT WILL CAUSE DAMAGE.

      (3) Grip drive shaft blade and slide wheel generator out of axle.

      (4) Disconnect electrical connector.

   B. Installation

      (1) Secure electrical connector to wheel generator and tighten.

      (2) Slide wiring and wheel generator into axle and align mounting holes.

      (3) Secure with attaching hardware as follows:

         (a) Apply a light coat of Loguic Primer + T + to male and female threads, and allows 5 minutes to dry.

         (b) Apply a light coat of Loctite Sealant #242 to male threads only.

         (c) Install adapter and hardware. Allow 24 hours to cure at room temperature or 15 minutes at 200°F.

         NOTE: Ensure a minimum 0.55 inch clearance exists between bolt ends and drive blade.

      (4) Perform Anti-Skid Control System — Operational Test.

3. **Main Landing Gear Anti-Skid Control Valve — Removal / Installation**

NOTE: The following applies to aircraft having ASC 206.

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Remove all electrical and hydraulic power from aircraft.

(2) Chock wheels and release brakes.

(3) Release pressure from brake system.

(4) Disconnect electrical connector.

(5) Disconnect hydraulic lines, Cap lines and cover in valve.

(6) Remove attaching hardware and remove valve.

(7) Remove filter and retain.

B. Installation

CAUTION: THE ANTI-SKID VALVES MUST BE INSTALLED WITH THE PRESSURE PORT (P) AND THE RETURN PORT (R1) FACING FORWARD.

(1) Position control valve and install attaching hardware.

(2) Connect hydraulic lines, ensure proper connections.

(3) Connect electrical connectors.

NOTE: Ensure proper electrical connections. If any doubt exists refer to Wiring Diagram Manual or ASC 206.

(4) Ensure that filter is installed in P-port of valve.

(5) Ensure that check valves installed in R1 and R2-ports have arrows pointing away from valve.

(6) Bleed brakes.

(7) Perform a Brake System — Operational Test, see Section 32-12-0.

(8) Perform an Anti-Skid System — Operational Test, see this Section.

4. **Anti-Skid Valve Filter — Inspection**

A. Removal

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

(1) Remove electrical and hydraulic power from aircraft.

(2) Release hydraulic pressure from brake system.

(3) Disconnect hydraulic line at pressure P-port of anti-skid valve.

(4) Remove filter from valve. Discard O-ring.

(5) Clean filter using ultrasonic cleaning unit.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES, KEEP AWAY FROM FLAMES.

NOTE: If ultrasonic cleaning unit is not available, and if filter element shows no evidence of clogging, element may be cleaned by agitating it in clean dry cleaning solvent.

B. Installation

(1) Install new O-ring on filter fitting.

(2) Install filters in anti-skid valve. Connect hydraulic line.

(3) Bleed brakes.

(4) Perform a Brake System — Operational Test, see Section 32-12-0.

(5) Perform an Anti-Skid System — Operational Test, see this Section.

5. Anti-Skid System — Operational Test

NOTE: The following applies to aircraft having ASC 206.

A. Using Cockpit Test Switch:

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: This test is to be performed on anti-skid system with a completed installation in aircraft, INBD ANTI-SKID, OUTBD ANTI-SKID, ANTI-SKID WARN, TEST, ANTI-SKID CONTROL and NUT-CRACKER circuit breakers depressed.

If test warning lights do not operate as specified in Steps A.(1) through A.(4) below a complete system check is required using Gulfstream test box 1159SEAV20153 or Aircraft Braking System test box 9560835 to isolate the defective component. Instructions are supplied with test units.

(1) With electrical power applied to aircraft, observe ANTI-SKID warning light (amber). It should be on when anti-skid control switch is OFF and out when switch is ON.

(2) Place control switch to ON.

(3) Move anti-skid test switch (nose wheel steering panel) to the left and right position while observing adjacent test light (green) warning light (red). Hold test switch in each position at least 5 seconds.

(4) Test light (green) should go OUT when test switch is returned to center position. Warning light (red) should not come ON during this test.

NOTE: Hydraulic portion of system can also be checked in conjunction with test switch. Apply pressure to all brakes. When test switch is in left position, left wheel brakes should release. When test switch is in right position, right wheel brakes should release.

B. Tweak Test:

NOTE: Do not use auxiliary pump for pressurizing system during anti-skid tests.

(1) Place anti-skid power switch to ON.

(2) Pressurize main hydraulic system.

(3) Have a man in cockpit depress brake pedals.

NOTE: A light spin of wheel generator is sufficient for test purposes if wheel generator is rotated rapidly for 2.5 to 3.0 seconds, the fail light will come ON. This is normal simulation of a locked wheel which reverts anti-skid system to manual operation. To reset system, turn anti-skid switch OFF momentarily.

(4) At same time brakes are being applied, spin (tweak) each wheel generator shaft in direction of wheel rotation.

(5) Ensure that a momentary brake release occurs as wheel generator shaft stops.

C. Fail.Safe Circuit Test

(1) Remove electrical plug at each anti-skid control valve.

(2) Ensure anti-skid warning light on switch panel comes ON at approximately 2.5 seconds after each plug is removed.

NOTE: To reset fail-safe circuits between checks, turn anti-skid switch OFF momentarily.

6. **Anti-Skid Control Box — Removal / Installation**

NOTE: The following applies to aircraft having ASC 206.

A. Removal

(1) On lower overhead panel, place the BATT switch OFF. Place the EXT PWR switch OFF. Disconnect external electrical and hydraulic power from aircraft.

(2) Loosen clamping device securing control box to mounting tray in electronic equipment rack.

(3) Carefully pull control box forward until power receptacle at rear of control is disengaged. Remove control box from mounting tray.

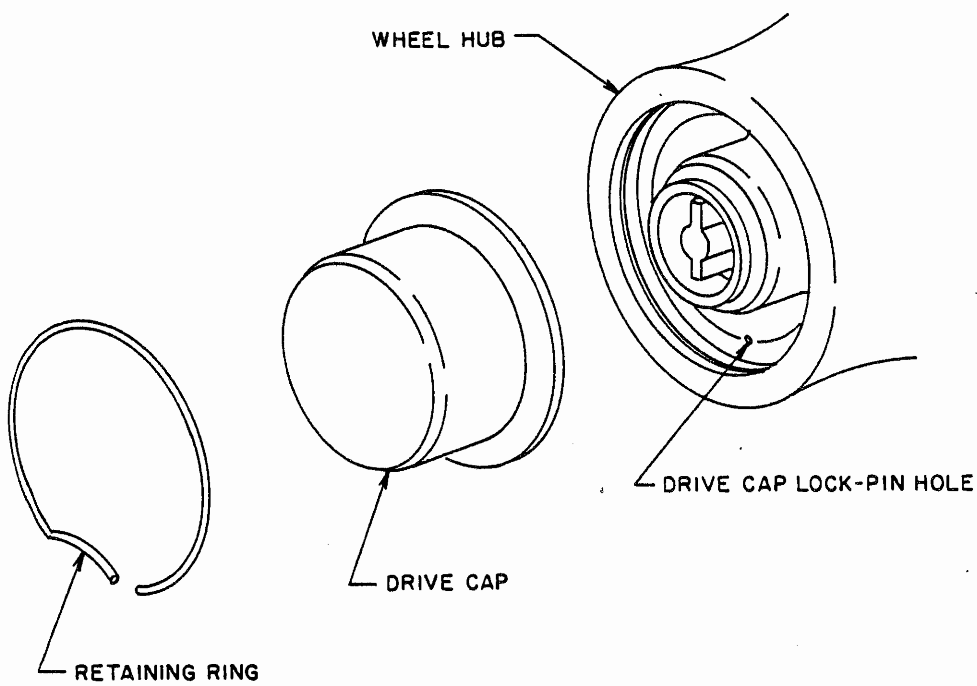
B. Installation:

(1) Place control box in mounting tray in electronic equipment rack.

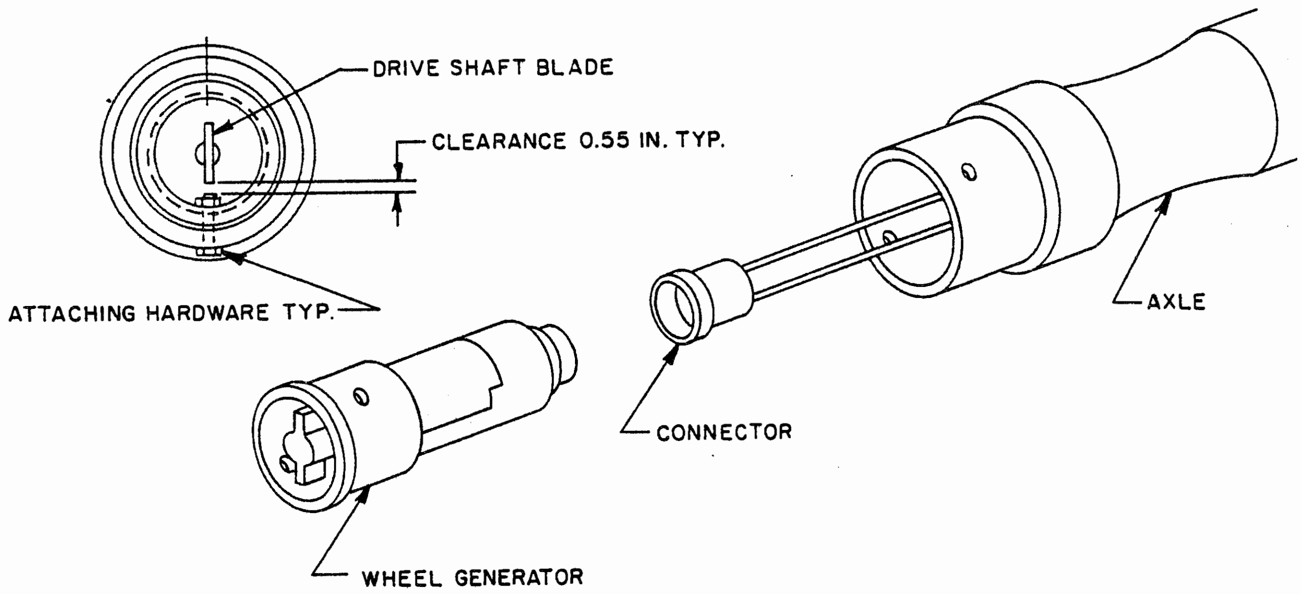
(2) Slide control box to the rear to engage power receptacle. Ensure receptacle is fully seated.

(3) Firmly secure clamping device to control box.

(4) Perform Anti-Skid System — Operational Test, see this Section.



Drive Cap
Figure 201.



Wheel Generator
Figure 202.

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LIGHTS

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LIGHTS

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LIGHTING — DESCRIPTION / OPERATION

1. **General**

This chapter describes the aircraft lighting circuitry. The coverage for interior lighting is limited to warning lights system and cockpit lighting, since all other interior lighting circuitry is completed by the aircraft outfitters. Production aircraft have provision wiring installed for future sign-lighting systems. The components of this system which are installed are: sign-lighting selector switch (located in the lower overhead panel) and sign-lighting circuit breaker (located in the left circuit breaker panel). Sign-light wiring is complete only to the mid-relay box, located under the cabin floor at Fuselage Station 193.

The aircraft is equipped with an exterior lighting system as follows: two flush mounted extendable landing lights (located in each aft wing-fuselage fillet); two taxi lights (mounted side by side on the nose landing gear strut); four wing inspection lights (one mounted on each side of the fuselage at Fuselage Station 205 and one additional light on the outboard side of each nacelle); three anti-collision lights (one located at the top forward section of the fin, one in the tail cone, and one in the fuselage bottom at Fuselage Station 503; a wing position light in each wing tip) and a white tail position light near the top of the rudder.

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WARNING LIGHTS SYSTEM — FAULT ISOLATION

1. **Warning Lights System — Fault Isolating**

   Warning light system faults usually will consist of defective lamps, capsule indicators, switches and faulty wiring. If it is determined that the lamps and circuitry are not defective, replace either right or left warning light display panel.

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WARNING LIGHTS SYSTEM — MAINTENANCE PRACTICES

1. Master Warning Lights Display Panel — Removal / Installation

CAUTION: BEFORE PERFORMING ANY OF THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. PULL APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to the master warning display panels, located in the sloping portion of each side console in the cockpit.

(2) Disconnect plugs from underside of unit.

(3) Release hardware holding panel to console.

(4) Remove panel.

B. Installation

(1) Install panel in console.

(2) Connect plug to underside of unit.

(3) Energize main and essential dc buses.

(4) Depress and hold WARN LIGHTS TEST button. All capsules in both left and right panels should light plus the two master warning lights in the eyebrow panel.

(5) While still depressing the TEST button, place NAV LIGHTS switch to ON. All lights in both master warning display panels will dim as well as the two master warning lights in the eyebrow panel.

(6) Release TEST button. Lights which normally should be out will go out, the rest will remain on.

(7) Momentarily depress the pilot WARN LIGHT RESET button. Master warning lights in eyebrow panel will go out but this will have no effect on the capsules lit in the display panels.

(8) Interrupt electrical power momentarily by placing BATT switch to OFF and the back to NORM.

(9) Master warning lights in eyebrow panel will come on.

(10) Repeat Step (7) above using the copilot WARN LIGHT RESET button, checking for the same action.

(11) De-energize electrical power.

(12) Install access panel.

2. Master Warning Lights Control Box — Removal / Installation

(Aircraft 1 - 120, 322 and 323)

A. Removal

(1) Gain access to the master warning lights control box by removing FLR-7, cabin area.

(2) Remove plugs from unit.

(3) Remove and retain mounting hardware.

(4) Remove unit.

B. Installation

(1) Install box.

(2) Connect electrical plugs and safety wire.

(3) Energize main and essential dc buses.

(4) Depress and hold WARN LIGHTS TEST button, on eyebrow panel, in cockpit. All capsules in both left and right panels should come on plus the two master warning lights in the eyebrow panel.

(5) While still depressing the TEST button, place NAV LIGHTS switch to ON. All lights in both master warning panels will dim as well as the two master warning lights in the eyebrow panel.

(6) Release TEST button. Lights which normally should be out will go out, the rest will remain on.

(7) Depress momentarily the pilots WARN LIGHT RESET button. Master warning lights in eyebrow panel will go out but this will have no effect on the capsules in the display panels.

(8) Interrupt electrical power momentarily by placing BATT switch to OFF and then back to NORM.

(9) Master warning lights in eyebrow panel will come on again.

(10) Repeat Step (7) above for using the copilots WARN LIGHTS RESET button, checking for the same action.

(11) De-energize electrical power.

(12) Install FLR-7.

INTERIOR MISCELLANEOUS LIGHTS — DESCRIPTION / OPERATION

1. **General**

A. Cockpit interior lighting consists of the following configuration.

(1) White floodlights for left and right instrument panels, left and right side consoles, overhead panel and center pedestal. Floodlights are variable intensity and range from full off to full on.

(2) Dome lighting for full illumination of cockpit.

(3) Map lights for copilot and pilot mounted on side panels and individually controlled.

(4) Red edge lighting for left and right lighting control panels, upper and lower overhead panels, pilot and copilot skirt panel, eyebrow panel and entire pedestal. Edge lighting is variable intensity and range from full on to full off.

(5) Red flood lighting for left and right instrument panels and consoles, floodlights are variable intensity and range from full on to full off.

(6) Individual red post lights for lighting of applicable aircraft.

B. Passenger Warning Light Provisions

All production aircraft have wiring installed for a future sign-lights system. The components of this system which are installed are: the sign-light selector switch located in the lower overhead panel and a SIGN LTS circuit breaker located in the left circuit breaker panel. Sign-light wiring is complete only to the mid relay box, located under the cabin floor at Fuselage Station 193 (FLR-8). From this point, circuitry can be completed to suitably placed sign units. The sign-light switch is labeled PASS WARN and has three positions associated with operator-furnished lamps. The three positions of the PASS WARN switch are:

- ST. BELTS
- OFF
- ST. BELTS - NO SMOKE

C. Emergency exit lights are located throughout the cabin. These units are self-contained, battery operated. The lights may be turned on manually by means of a switch on the unit and automatically come on when a sufficient impact force trips an internal mechanism.

NOTE: The cockpit lighting system will be covered in two separate sections, as follows:

Section 33-2-1 is for Aircraft 1 - 86 and 114 not having ASC 115.

Section 33-2-2 is for Aircraft 1 - 86 and 114 having ASC 115 and Aircraft 87 - 200, 322 and 323.

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INTERIOR MISCELLANEOUS LIGHTS — MAINTENANCE PRACTICES

1. Interior Lights — Operational Test

A. Connect a 28V external source of dc power to system.

B. Ensure all circuit breakers pertaining to interior lights are depressed.

C. Check all lights in cockpit, cabin and tail compartment for operation using appropriate switches and rheostats.

D. Close the (MA WARN) and (WARN LTS. BUS) circuit breakers.

E. Push in the WARN LIGHT TEST switch. All capsule indicators in the right and left warning light display panels should come on. The master warning lights mounted on the cockpit eyebrow panel and capsule indicator lights located on the cockpit panels should all come on.

F. Turn EMER EXT LT manual switch on and inspect lights for brilliance replace bulbs /battery if necessary.

G. Turn EMER EXT LT manual switch off.

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WARNING LIGHTS SYSTEM — DESCRIPTION / OPERATION

1. **Description**

A. The warning lights system provides a warning to crew members of malfunctions of the aircraft equipment, an unsafe operating condition which requires immediate correction or protective measures or a warning that a particular system is in operation. This system makes possible the presentation and testing of warning and indication circuits for various aircraft systems. Two arrangements are employed; one utilizes two display panels and two master lights and the other utilizes relays and indicator lights.

B. The following major components comprise the warning lights system:

Unit	No Per A/C	Location
Warning Light Control Box	1	Under cabin floor at Fuselage Station 225. (FLR-7) (Aircraft 1 - 120, 322 and 323)
Warning Light Display Panel	2	One panel is mounted on each side of cockpit on a side panel.
Capsule Indicators	60	Thirty in each warning light display panel. (Aircraft 1 - 120, 322 and 323)
	64	32 in each warning light display panel. (Aircraft 121 - 200, 322 and 323)
Master Warning Lights	2	One on either side of the cockpit eyebrow panel.
Reset Switches (WARN LT. RESET)	2	One on either side of the cockpit eyebrow panel.
Test Switches (WARN LIGHT TEST)	2	One on either side of the cockpit eyebrow panel.
Master Warning Lights Reset Relay	1	Left Fuselage Station 133 relay panel. (Aircraft 1 - 120, 322 and 323)
Master Warning Lights Dimming Relay	1	Left Fuselage Station 133 relay panel. (Aircraft 1 - 120, 322 and 323)
Warning Lights Test Relay A	1	Left Fuselage Station 133 relay panel.
Warning Lights Test Relay B	1	Left Fuselage Station 133 relay panel.
Warning Lights Test Relay C	1	Cockpit Ceiling (Behind upper overhead panel).
Warning Lights Test Relay D	1	Landing Gear Lights Resistor Box.
Windshield Heat Indicator Light Test Relay	1	Windshield Heat Junction Box (FLR-4).
Warning Lights Test Relay E	1	Fuel Filter Heat Timer Box (FLR-8).
Warning Lights Test Relay F	1	Right Fuselage Station 133 relay panel (included only for aircraft having ASC 108A, having Part I).
Landing Gear Lights Resistor Box	1	Panel on left side Fuselage Station 99 above nose gear opening.
Warn Lts. Test Power Relay	1	Left Fuselage Station 133 relay panel. (Aircraft 121 - 200 excluding 322, 323)

2. Operation

A. Aircraft 1 - 120, 322 and 323 (See Figure 1 and Figure 4)

(1) The warning light control box, two display panels and two master warning lights comprise the master warning light system. The control box consists of diodes, relays and resistors packed in a container. These are wired for passage of a warning signal (either negative or positive), testing and dimming of the display panel and master warning light bulbs and for resetting of the master warning lights.

(2) When a warning signal is received by the warning light control box, the following occurs:

(a) The signal passes through to the appropriate capsule indicator in each display panel.

(b) The signal causes the two red master warning lights, mounted on the cockpit eyebrow panel, to come on.

(3) Reset switches are provided on either side of the cockpit eyebrow panel. Pressing either reset button energizes the coil of the master warning lights reset relay. A signal is sent through the relay contacts to the control box, causing the master warning lights to go out. In each display panel, the particular capsule indicator, consisting of two lamps, remains lighted until the incoming signal ceases.

(4) Dimming resistors for the capsule indicators are located in each display panel. Dimming resistors for the master warning lights are located in the warning light control box. When dimming is desired, the navigation lights switch is moved to the ON position. This causes the navigation lights to come on and the coil of the master warning lights dimming relay to be energized. A signal is sent through the relay contacts to the control box. The control box circuitry then diverts all incoming signals through the dimming circuits.

NOTE: All capsules are pin indexed to prevent incorrect positioning.

(5) Test switches are provided on either side of the cockpit eyebrow panel. Before energizing the test system place the battery switch in the NORMAL position. Pressing either test switch button energizes the coils of warning test relays A, B, C, D, E and F (Aircraft having ASC 108A, Part I); windshield heat indicator light test relay and warning light test power relay. Through contact C of warning test relay A, and warning light test power relay, a positive and a negative signal are applied to the control box. The control box reacts as if it were receiving warning signals from all the warning circuits and all bulbs light. All remaining contacts of relays are associated with indicator lights in the cockpit panels, and are used in testing indicator bulbs. When either of the test switches are operated, relays are energized, which complete circuits from the main dc bus. Current is supplied through the relay contacts to the indicator lights. Individual mechanical dimming is provided in the indicator lights on the cockpit panels.

(6) Warning lights not tested by the warning lights test system.

Panel	Light	Test
Upper Overhead	Pitot Heat (2)	None (turn on pitot heat)
Eyebrow Panel	Propeller Amber Lights (8)	Individual Press-to-Test
	Fire Pull T handles	FIRE DET TEST Button (Instrument ac bus must be energized)
	Nacelle Fire (2)	FIRE DET TEST Button
	APU Fire	FIRE DET TEST Button
Skirt Panels	Landing Gear Handles (2)	No Test

B. Aircraft 121 - 200 (See Figure 2 and Figure 3)

(1) Two display panels and two master warning lights comprise the master warning light system. The left display panel consists of 32 capsules and diodes and relays packed in a container. These are wired for passage of a warning signal (either negative or positive), test and reset of master warning lights. The right display panel slaves from the left and provides dimming of both display panels and master warning light bulbs through two dimming relays and use of zener diodes.

(2) When a warning signal is received by the left display panel, the following occurs:

(a) The signal passes through to the appropriate capsule indicator in each display panel.

(b) The signal causes the two red master warning lights, mounted on the cockpit eyebrow panel, to come on.

(3) Reset switches are provided on either side of the cockpit eyebrow panel. Pressing either reset button energizes the coil of the particular system's reset relay. This reset relay, once energized removes the power holding the master warning lights relay energized, causing the master warning lights to go out. In each display panel, the particular capsule indicator, consisting of two lamps, remains lighted until its incoming signal ceases.

(4) Dimming for display capsules and master bulbs is accomplished by the use of two dimming relays and zener diodes located in the right display panel. When dimming is desired, the navigation lights switch is moved to the ON position. This causes the navigation lights to come on and the coils of the two dimming relays to be energized. With these two relays energized, any incoming signal to turn on the warning capsule and the master warning bulbs must pass through a zener diode (acting as a resistor) before passing to ground, causing the bulbs to come on less brightly.

(5) Test switches (WARN LIGHT TEST) are provided on either side of the cockpit eyebrow panel. To energize the test system, the battery switch must be in the NORMAL position. Pressing either test switch button energizes the coils of warning test relays A, B, C, D, E and F, the windshield heat indicator light test relay and the warning light test power relay. Power from either test switch also energizes the one negative capsule test relay and causes it to supply a ground, so that the negative capsule on each display panel (outside battery sw), and all positive capsules in each display panel become illuminated. The main dc bus supplies current for all testing of bulbs.

(6) Warning lights not tested by the warning lights test system.

Panel	Light	Test
Upper Overhead	Pitot Heat (2)	None (turn on pitot heat)
Eyebrow Panel	Propeller Amber Lights (8)	Individual Press-to-Test
	Fire Pull T-handles	FIRE DET TEST Button (Instrument ac bus must be energized)
	Nacelle Fire (2)	FIRE DET TEST Button
	APU Fire	FIRE DET TEST Button
Skirt Panels	Landing Gear Handles (2)	No Test

3. Emergency DC Operation

A. All positive warning signals which receive power from the essential dc bus turn on the respective capsules on each console.

B. The two master warning lights on the eyebrow panel do not operate.

C. The one negative warning signal capsule does not operate.

D. The warning lights test function of the warning system is inoperative.

E. Those positive warning lights which are turned on (located on the console display panel) are dimmed when the navigation lights switch is placed in the ON position.

AC INSTR BUS
AC MAIN BUS
AC EQUIP BUS

L GEN OFF
R GEN OFF
L AC GEN OFF
R AC GEN OFF
BAT FAIL
L GEN HOT
R GEN HOT
L AC GEN HOT
R AC GEN HOT

RADIOS HOT
BLOWER HOT
L OIL PRESS
R OIL PRESS
L FUEL PRESS
R FUEL PRESS

L W/M PRESS
R W/M PRESS
L GEARBOX PRESS
R GEARBOX PRESS
CABIN PRESS
L FUEL QUAN
R FUEL QUAN
CABIN OXYGEN ON
DOORS UNSAFE
IGNITION ON
AUTOPILOT
OUTSIDE BAT SW

Master Warning Light Display Panel (right Side)
(Aircraft 1-120, 322 and 323)
Figure 1.

AC INSTR BUS
AC MAIN BUS
AC EQUIP BUS
L GEN OFF
R GEN OFF
L AC GEN OFF
R AC GEN OFF
L FUEL PRESS
R FUEL PRESS
L W/M PRESS
R W/M PRESS
BAT FAIL
BLOWER HOT
CABIN PRESS
L GEN HOT
R GEN HOT
L AC GEN HOT
R AC GEN HOT
L FUEL QUAN
R FUEL QUAN

RADIOS HOT
L OIL PRESS
R OIL PRESS
CABIN OPEN
IGNITION ON
AUTO PILOT
L GEARBOX PRESS
R GEARBOX PRESS
OUTSIDE BAT SW

Master Warning Light Display Panel (right Side)
(Aircraft 121 - 200 and 114)
Figure 2.

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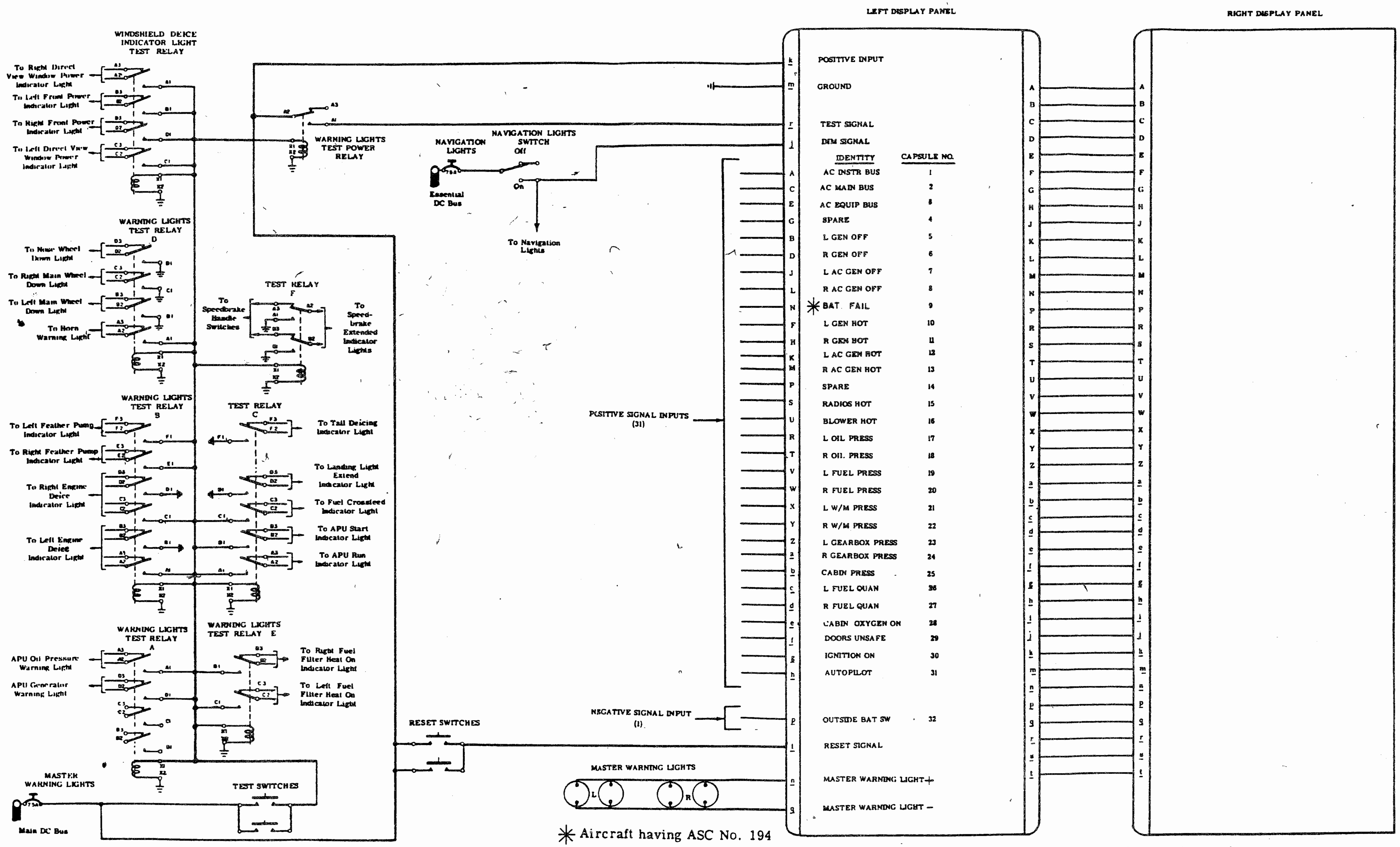
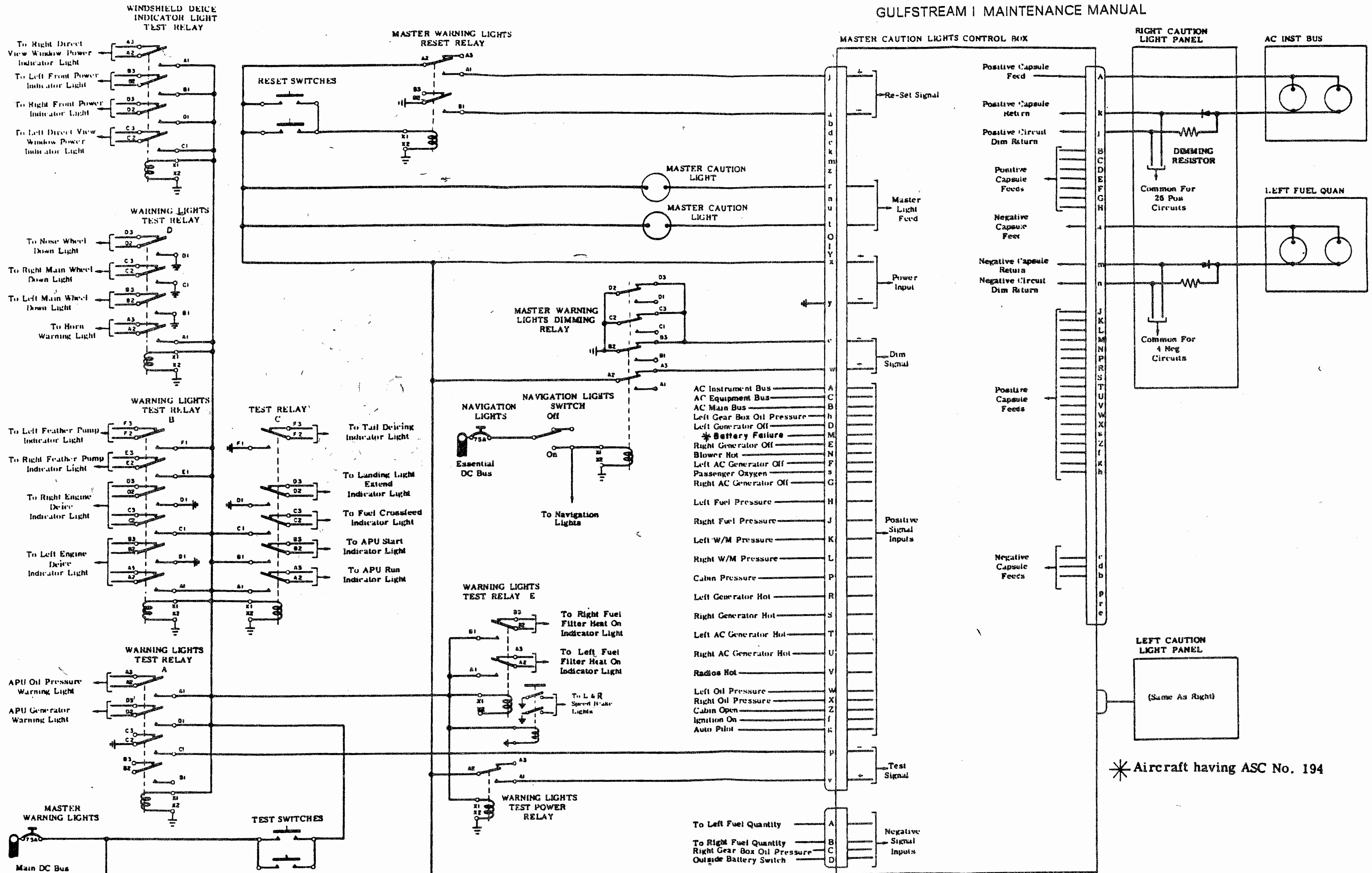


Warning Lights — Electrical Schematic
(Aircraft 121 - 200, 322 and 323)
Figure 3.



Warning Lights — Electrical Schematic
(Aircraft 1 - 120, and 114)
Figure 4.

INTERIOR LIGHTS (AIRCRAFT 1 - 86, AND 114 NOT HAVING ASC 115) — DESCRIPTION / OPERATION

1. **Description**

A. Console, Flight Instrument Panel, Overhead and Pedestal White Floodlights. (See Figure 1 and Figure 2)

(1) The copilots right side console receives white floodlight illumination from three assemblies. Each assembly contains a 6 cp, No. 1495 bulb. The lights are controlled with the SIDE CONSOLE PANEL WHITE FLOOD rheostat on the right lighting control panel when the WHT. OVERRIDE switch, located on the lower overhead panel, is in the center position. The pilots left side console receives white floodlight illumination from three assemblies. Each assembly contains a 6 cp, No. 1495 bulb. The lights are controlled with the SIDE CONSOLE PANEL WHITE FLOOD rheostat on the left lighting control panel when the WHT. OVERRIDE switch is in the inactive position. Power for both the pilots and copilots console white floodlights, come from the main dc bus. The circuits are protected by the CONSOLE FLD. 5 ampere circuit breaker.

(2) The right flight instrument panel receives white floodlight illumination from three assemblies. These assemblies each contain a 6 cp, No. 1495 bulb. The lights are controlled with the INSTRUMENT PANEL WHITE FLOOD rheostat in the right lighting control panel when the WHT. OVERRIDE switch is in the inactive position.

The left flight instrument panel receives white illumination from three assemblies each containing a 6 cp, No. 1495 bulb. These lights are controlled with the INSTRUMENT PANELS WHITE FLOOD rheostat on the left lighting control panel when the WHT. OVERRIDE switch is in the inactive position.

Power for both left and right flight instrument panel white floodlights comes from the main dc bus. The circuits are protected by the INST. FLD. 5 ampere circuit breaker.

(3) The overhead panel receives white floodlight illumination from two floodlight assemblies mounted overhead in the cockpit. Each floodlight assembly contains a 20 watt, No. 1385 bulb. These lights are controlled by the OVERHEAD FLOOD rheostat, located in the center portion of the upper overhead panel, when the WHT. OVERRIDE switch is in the inactive position.

Power for both floodlights comes from the essential dc bus. The circuit is protected by the DOME FLD. 5 amp circuit breaker.

(4) The center pedestal receives white floodlight illumination from one floodlight assembly mounted overhead in the cockpit, which contains a 20 watt, No. 1385 bulb. This light is controlled by the PEDESTAL FLOOD rheostat, located in the center portion of the upper overhead panel, when the WHT. OVERRIDE switch is in the inactive position.

Power for the pedestal floodlight comes from the essential dc bus. The circuit is protected by the DOME FLD. 5 amp circuit breaker.

B. Dome Light

A dome light containing a No. 308 bulb, and a dome light switch are mounted overhead in the cockpit. The dome light, which provides general illumination of the cockpit, is controlled by the switch. Power is from the essential dc bus, through the DOME FLD. 5 amp circuit breaker.

C. Map Lights

Two map lights are installed above, and to the left of each front window. The right map light is controlled by a MAP FLOOD switch located on the right lighting control panel. The left map light is controlled by a MAP FLOOD switch located on the left lighting control panel. An adjustment lens which can be rotated for various spot light focuses is installed in each map light. Each light has its own brightness adjustment. Power is from the main dc bus through the MAP LTS. 5 amp circuit breaker. A No. 313 bulb is used in each map light.

D. White Override Switch

When the WHT. OVERRIDE switch is in the inactive position, the intensity of all cockpit white floodlights (except dome and map lights) is controlled by their individual rheostats. The WHT. OVERRIDE switch is a four pole switch mounted in the lower overhead panel. When the switch is placed in the ALL ON position, 3 poles complete a circuit to bypass the associated rheostats of various white floodlights, causing them to come on full intensity.

The fourth pole of the switch in the ALL ON position energizes the floodlight on relay which in turn completes circuits through its closed contacts, bypassing associated rheostats, allowing the remainder of the cockpit white floodlights to come on full intensity. It is advisable under certain flight conditions (lightning storms) to place the switch in the ALL ON position. This allows all white lights under its control to come on full intensity which tends to minimize the effect of bright flashes. Returning the switch to the inactive position de-energizes the relay and allows all white floodlights to return to individual rheostat control. When the switch is placed in the ALL OFF position a circuit is completed to energize the No. 1 and 2 floodlight out relays. These relays when energized deactivate the circuits to all cockpit white floodlights (except dome and map lights). This prevents inadvertent application of white light, while flying at night under red lighting conditions. (See Figure 1 and Figure 2)

E. Console, Flight Instrument Panel, Overhead and Pedestal Red Lights

(1) The copilot console receives red light illumination from two red floodlight assemblies. The copilot lighting control panel and right skirt panel receives red edge lighting from a total of 19 edge lights. Each floodlight assembly contains a 6 cp, No. 1495 bulb. The edge lights contain No. 9906-1 bulbs. These lights are controlled by the SIDE CONSOLE PANEL RED EDGE AND FLOOD rheostat on the right lighting control panel.

The pilot console receives red light illumination from two red floodlight assemblies. The pilot lighting control panel and left skirt panel receive red edge lighting from a total of 18 edge lights. Each floodlight assembly contains a 6 cp, No. 1495 bulb. The edge lights contain No. A-9906-1 bulbs. These lights are controlled with the SIDE PANEL RED EDGE AND FLOOD rheostat on the left lighting control panel.

Power for both the pilot and copilot console red floodlights and edge lights comes from the main dc bus. The circuits are protected by the CONSOLE EDGE 5 amp circuit breaker.

(2) The right flight instrument panel receives red floodlight illumination from five floodlight assemblies. Four post light assemblies in the same circuit illuminate the lower portion of the center instrument panel. Two floodlight assemblies contain a No. 1495 bulb. The remaining three contain No. 313 bulbs. The post lights contain No. 327 bulbs. These lights are controlled with INSTRUMENT PANELS RED FLOOD rheostat on the right lighting control panel.

The left flight instrument panel receives red floodlight illumination from five floodlight assemblies. Two floodlight assemblies contain a No. 1495 bulb, the remaining three contain No. 313 bulbs. These lights are controlled by the INSTRUMENT PANELS RED FLOOD rheostat located on the left lighting control panel. Also in this circuit is the 24 volt lighting feed intended to be used in the customer installed standby magnetic compass.

Power for both left and right flight instrument panel red floodlights comes from the essential dc bus. The right flight instrument panel red floodlight circuit is protected by the INST. EDGE 5 amp circuit breaker. The left flight instrument panel red floodlight circuit is protected by the INST. EDGE 5 amp circuit breaker.

(3) The eyebrow and the lower overhead panels are edge lighted with 43 edge lights. Each edge light contains a No. A-9906-1 bulb. These lights are controlled by the EYEBROW and LOWER OVERHEAD EDGE rheostat, located in the center position of the upper overhead panel.

The upper overhead panel is edge lighted with 54 edge lights. Each edge light contains a No. A-9906-1 bulb. These lights are controlled with the OVERHEAD EDGE rheostat in the center portion of the upper overhead panel.

Power for all of the overhead red edge lights comes from the main dc bus. The circuit is protected by the CENTER EDGE 7.5 amp circuit breaker.

(4) The forward pedestal and elevator trim wheel is edge lighted with 23 edge lights. Each edge light contains a No. 327 bulb. These lights are controlled by the PEDESTAL EDGE rheostat, located in the center portion of the upper overhead panel.

A PEDESTAL RADIO EDGE rheostat is located in the center portion of the upper overhead panel. This unit is capable of controlling up to 35 edge lights. Circuits to the pedestal terminal panel are provided for customer-supplied radio controls. This circuit also provides edge lighting for the rudder and aileron trim knobs.

Power for all pedestal red edge lights comes from the main dc bus. The circuit is protected by the CENTER EDGE 7.5 ampere circuit breaker.

F. Emergency DC Operation

(1) The following cockpit lights are operative in emergency:

(a) Pedestal and overhead flood lights (white).

(b) Dome Light (white).

(c) Left and right instrument panel red lights.

(d) Standby compass light.

(2) All other cockpit lights are inoperative in emergency.

(3) The WHT. OVERRIDE switch is inoperative in emergency.

2. Major Components and Locations

A. Rheostats - on associated lighting panels (pilot, copilot and upper overhead).

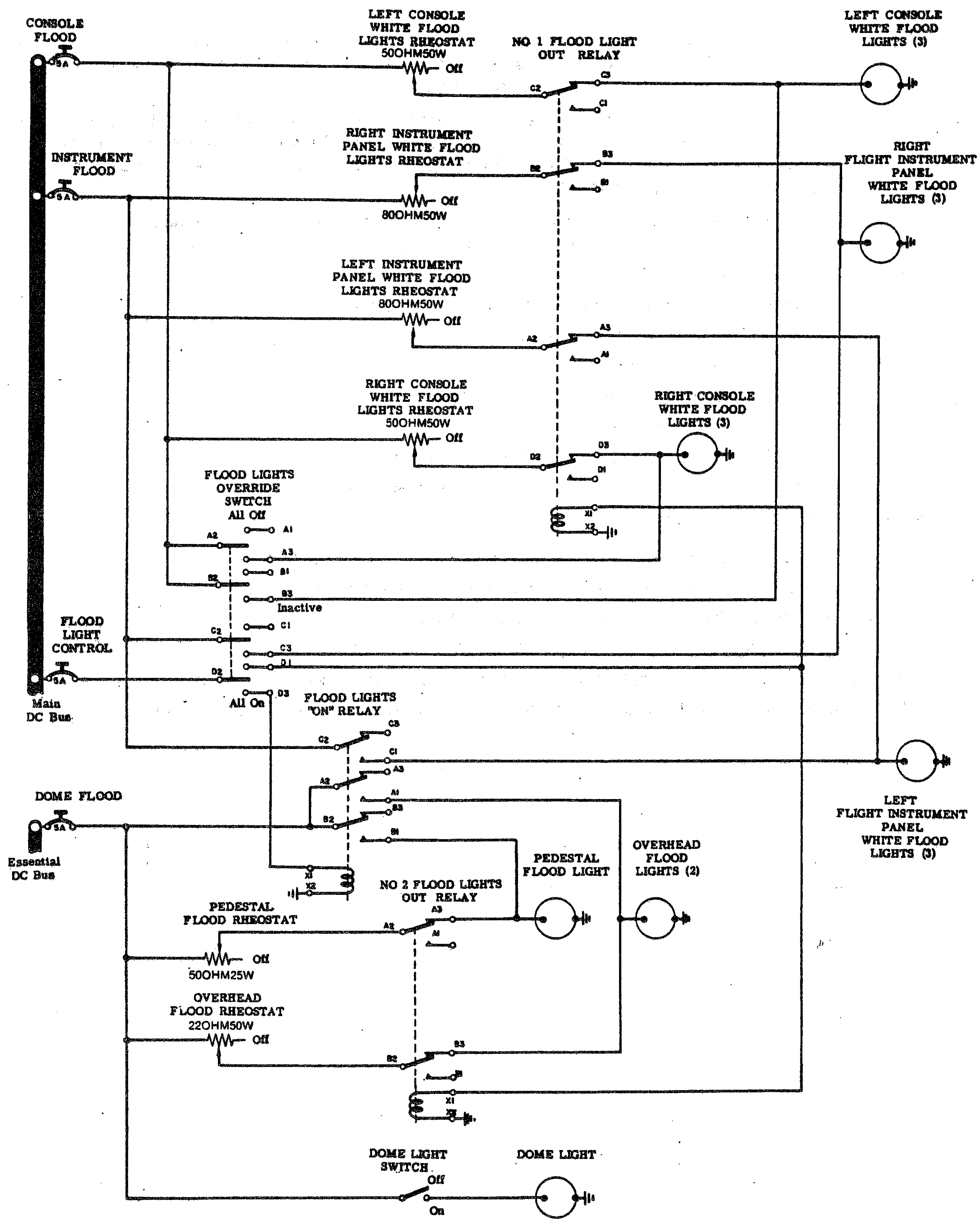
B. Map Light Switch - on respective pilot and copilot lighting control panels.

C. Dome Light and Switch - overhead in cockpit ceiling.

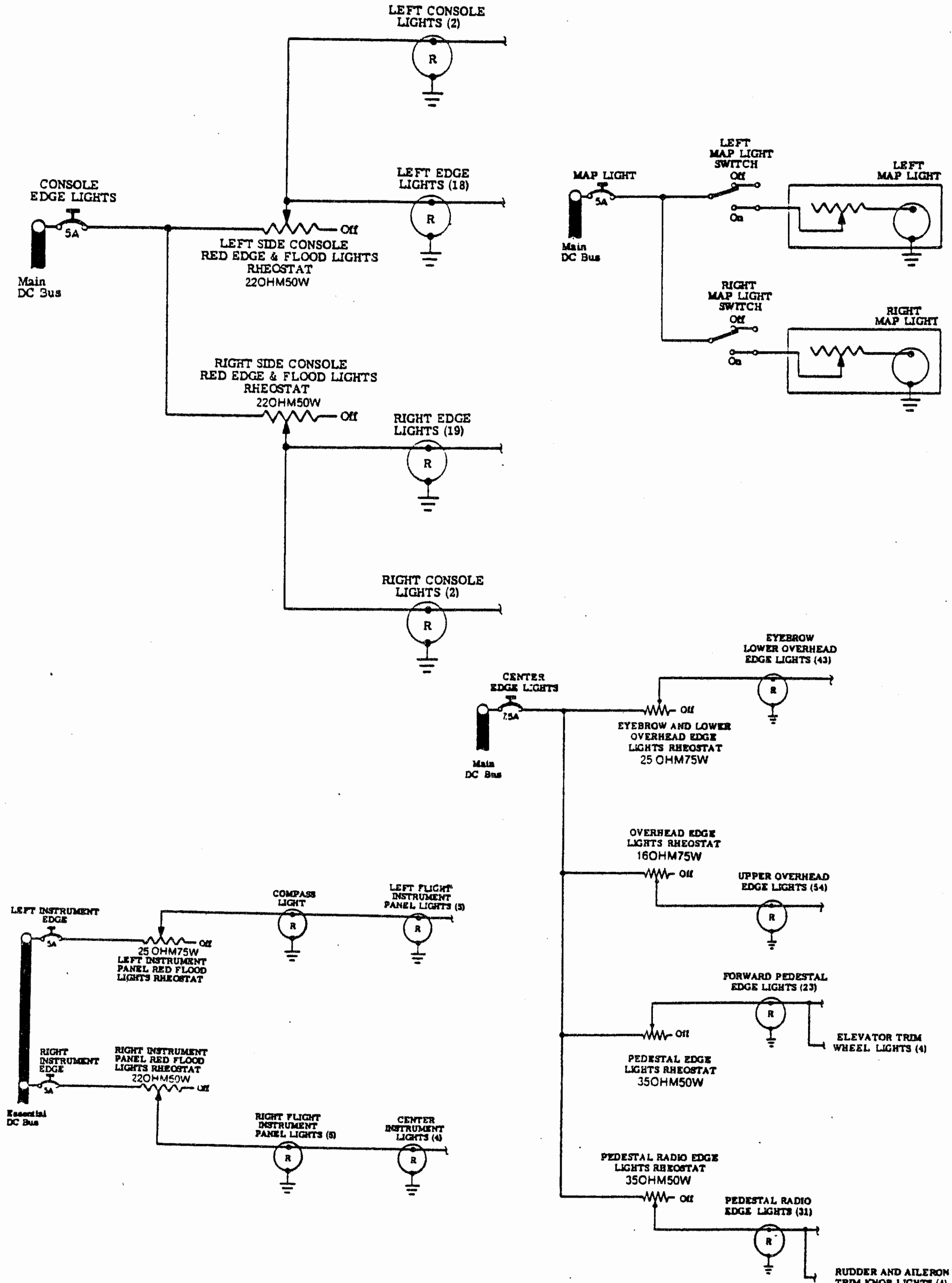
D. Flood Light ON Relay - left Fuselage Station 133 relay panel.

E. Flood Light OUT Relays No. 1 and No. 2 - left Fuselage Station 133 relay panel.

F. All associated circuit breakers are located on the left Fuselage Station 133 circuit breaker panel.



Cockpit Lights — Schematic
Figure 1.



Instrument/Edge Lights — Schematic
Figure 2.

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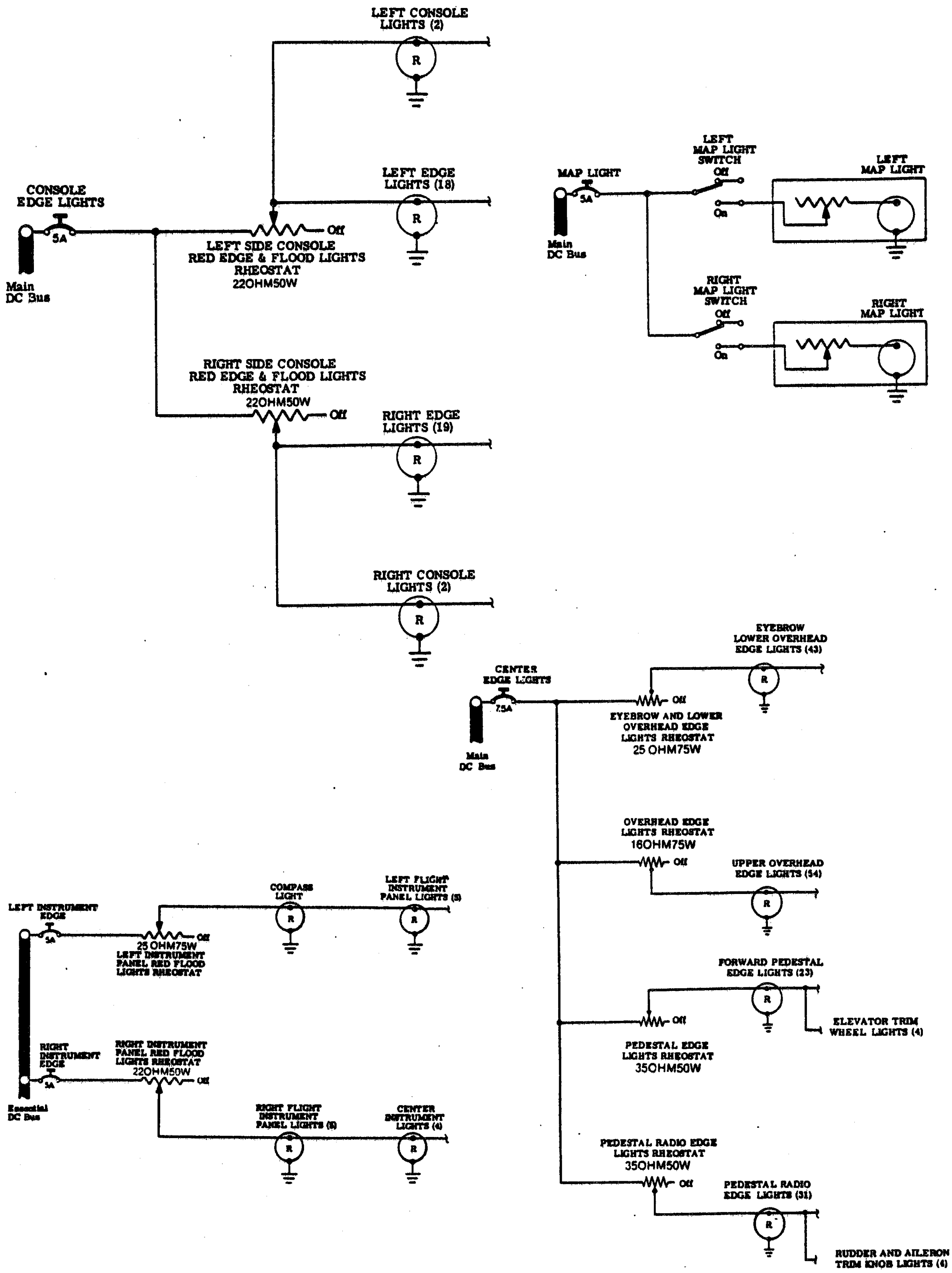

Interior Lights — Schematic
Figure 2.

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INTERIOR LIGHTS (AIRCRAFT 1 - 86, AND 114 NOT HAVING ASC 115) — MAINTENANCE PRACTICES

1. **General Maintenance Precautions**

   A. Edge lighting panels are internally wired and extreme caution should be used when drilling or altering holes in these panels, or when applying paint. Obtain detailed print of panel for exact wire routing before drilling any holes.

   B. Application of paint in the engraved grooves will prevent the light from shining through engravings when required, defeating the purpose of the panels.

   C. All edge lighting panels bulb replacements must be GRIMES No. A-9906-1 or equivalent, since a special combination bulb-socket is used. A No. 327 type will not fit, except in pedestal edge lighting system which uses 327 bulbs.

   D. Edge lighting panel bulbs are installed from the aft side of the panel, requiring the removal of the panel for bulb replacement, except for pedestal edge light panels.

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INTERIOR LIGHTS (AIRCRAFT 87 - 200, 322 AND 323 HAVING ASC 115) — DESCRIPTION / OPERATION

1. **Description**

   A. Console, Flight Instrument Panel, Overhead and Pedestal White Floodlights. (See Figure 1 thru Figure 3)

      (1) The copilots right side console receives white floodlight illumination from three assemblies. Each assembly contains a 6 cp, No. 1495 bulb. The lights are controlled with the SIDE CONSOLE PANEL WHITE FLOOD rheostat on the right lighting control panel when the WHT. OVERRIDE switch located on the upper overhead panel, is in the center position. The pilots left side console receives white illumination from three assemblies. Each assembly contains a 6 cp, No. 1495 bulb. The lights are controlled with the SIDE CONSOLE PANEL WHITE FLOOD rheostat on the left lighting control panel when the WHT. OVERRIDE switch is in the inactive position. Power for both the pilots and copilots console white floodlights comes from the main dc bus. The circuits are protected by the CONSOLE FLD. 5 amp circuit breaker.

      (2) The right flight instrument panel receives white floodlight illumination from four assemblies. These assemblies each contain a No. 313 bulb. The lights are controlled with the INSTRUMENT PANEL FLOOD WHITE FLOOD rheostat in the right lighting control panel when the WHT. OVERRIDE switch is in the inactive position.

      The left flight instrument panel receives white floodlight illumination from four assemblies each containing a No. 313 bulb. These lights are controlled with the INSTRUMENT PANEL FLOOD WHITE FLOOD rheostat on the left lighting control panel when the WHT. OVERRIDE switch is in the inactive position. Power for both left and right flight instrument panel white floodlights comes from the main dc bus. The circuits are protected by the INST. FLD. 5 amp circuit breaker.

      (3) The overhead panel receives white floodlight illumination from two assemblies mounted overhead in the cockpit. Each floodlight assembly contains a 20 watt, No. 1385 bulb. These lights are controlled by the OVERHEAD FLOOD rheostat, located in the center portion of the upper overhead panel, when the WHT. OVERRIDE switch is in the inactive position.

      Power for both floodlights comes from the essential dc bus. The circuit is protected by the DOME FLD. 5 amp circuit breaker.

      (4) The center pedestal receives white floodlight illumination from one assembly mounted overhead in the cockpit, which contains a 20 watt, No. 1385 bulb. This light is controlled by the PEDESTAL FLOOD rheostat, located in the center portion of the upper overhead panel, when the WHT. OVER-RIDE switch is in the inactive position.

      Power for the pedestal floodlight comes from the essential dc bus. The circuit is protected by the DOME FLD. 5 amp circuit breaker.

   B. Dome Light

      A dome light containing a No. 308 bulb, and a dome light switch are mounted overhead in the cockpit. The dome light, which provides general illumination of the cockpit, is controlled by the switch. Power is from the essential dc bus, through the DOME FLD. 5 amp circuit breaker.

   C. Map Lights

      The map lights and map light controls are located directly above the left and right lighting control panel. The map light fixture is an "adjustable gooseneck" type and utilizes a No. 327 bulb. Power is from the main dc bus through the MAP LTS. 5 amp circuit breaker. Map light switch is pushbutton on-off type located aft of the fixture.

   D. White Override Switch

      When the WHT. OVERRIDE switch is in the center position, the intensity of all cockpit white floodlights (except dome and map lights) is controlled by individual rheostats. The WHT. OVERRIDE switch is a four pole switch mounted in the lower overhead panel. When the switch is placed in the ALL ON position, 3 poles complete a circuit to bypass the associated rheostats of various white floodlights, causing them to come on full intensity.

The fourth pole of the switch in the ALL ON position energizes the FLOODLIGHT ON RELAY which in turn completes circuits through its closed contacts, bypassing associated rheostats, allowing the remainder of the cockpit white floodlights to come on full intensity. It is advisable under certain flight conditions (lightning storms) to place the switch in the ALL ON position. This allows all white lights under its control to come on full intensity which tends to minimize the effect of bright flashes. Returning the switch to the inactive position de-energizes the relay and allows all white floodlights to return to individual rheostat control. When the switch in placed in the ALL OFF position a circuit is completed to energize the No. 1 and No.2 floodlight out relays. These relays deactivate the circuits to all cockpit white floodlights (except dome and map lights). This prevents inadvertent application of white light, while flying at night under red lighting conditions. (See Figure 1 thru Figure 3)

E. Console, Flight Instrument Panel, Center Instrument Panel, Overhead and Pedestal Red Lights.

(1) The copilots console receives red light illumination from two red light assemblies. The copilots lighting control panel and right skirt panel receives red edge lighting from a total of 19 edge lights. Each floodlight assembly contains a 6 cp, No. 1495 bulb. The edge lights contain No. A-9906-1 bulbs. These lights are controlled by the SIDE CONSOLE PANEL RED EDGE AND FLOOD rheostat on the right lighting control panel.

The pilots console receives red light illumination from two red light assemblies. The pilots lighting control panel and left skirt panel receives red edge lighting from a total of 20 edge lights. Each floodlight assembly contains a 6 cp, No. 1495 bulb. The edge lights contain No. 9906-1 bulbs. These lights are controlled with the SIDE CONSOLE PANEL RED EDGE AND FLOOD rheostat on the left lighting control panel.

Power for both the pilots and the copilots console red floodlights and edge lights comes from the main dc bus. The circuits are protected by the CONSOLE EDGE 5 amp circuit breaker.

(2) Lighting is provided for individual instruments on the engine instrument panels. The FLAPS position and FUEL DATUM indicators on the center instrument panel are individually lighted. These lights are controlled by the RED INSTRUMENT LIGHTS FLIGHT INSTR rheostat on the right lighting control panel. These lights utilize No. 327 bulbs. Power for these lights is supplied by the essential dc bus and is protected by the R INST EDGE 5 amp circuit breaker on the left Fuselage Station 133 circuit breaker panel.

(3) The remaining 16 engine instruments (8 on the right engine instrument panel and 8 on the left engine instrument panel) are each lighted by a lamp located directly above each instrument. These lights are controlled by the RED INSTRUMENT LIGHTS ENGINE INSTR rheostat located on the left lighting control panel. These lights utilize No. 327 bulbs. Power for these lights is supplied by the main dc bus and is protected by the CONSOLE EDGE 5 amp circuit breaker, located on the left Fuselage Station 133 circuit breaker panel.

(4) Lighting can be provided for up to 11 instruments on the left flight instrument panel and up to 10 instruments on the right flight instrument panel. Panel lights on the left flight instrument panel are controlled by the RED INSTRUMENT LIGHTS FLIGHT INSTR rheostat on the left lighting control panel. Power for these lights is supplied by each assembly; the essential dc bus and is protected by the L INST EDGES 5 amp circuit breaker located on the left Fuselage Station 133 circuit breaker panel. Panel lights on the right flight instrument panel are controlled by the RED INSTRUMENT LIGHTS FLIGHT INSTR rheostat, on the right lighting control panel. Power for these lights is supplied by the essential dc bus through the R and L INST EDGE LTS circuit breaker, respectively.

(5) The right flight instrument panel receives red floodlight illumination from four floodlight assemblies, each containing a No. 313 bulb. These lights controlled with RED INSTRUMENT LIGHTS FLIGHT INSTR rheostat on the right LIGHTING CONTROL PANEL.

The left flight instrument panel receives red floodlight illumination from four floodlight assemblies. Each assembly contains a No. 313 bulb. These lights are controlled by the RED INSTRUMENT LIGHTS FLIGHT INSTR rheostat located on the left lighting control panel. The internal light contained in the standby magnetic compass is also in this circuit.

Power for both left and right flight instrument panel red floodlights comes from the essential dc bus. The right flight instrument panel red floodlight circuit is protected by the RIGHT INSTRUMENT

EDGES 5 amp circuit breaker. The left flight instrument panel red floodlight circuit is protected by the LEFT INSTRUMENT EDGE 5 amp circuit breaker.

(6) The eyebrow and the lower overhead panels are edge lighted with 43 edge lights. Each edge light contains a NO. A-9906-1 bulb. These lights are controlled by the EYEBROW and LOWER OVERHEAD EDGE rheostat, located in the center position of the upper overhead panel.

The upper overhead panel is edge lighted with 54 edge lights. Each edge light contains a No. A-9906-1 bulb. These lights are controlled with the OVERHEAD EDGE rheostat in the center portion of the upper overhead panel.

Power for all of the overhead red edge lights comes from the main dc bus. The circuit is protected by the CENTER EDGE 7.5 amp circuit breaker.

(7) The forward pedestal and elevator trim wheel in edge lighted with 27 edge lights. Each edge light contains a No. A-9906-1 bulb. These lights are controlled by the PEDESTAL EDGE rheostat, located in the center portion of the upper overhead panel.

The PEDESTAL RADIO EDGE rheostat is located in the center portion of the upper overhead panel. This unit is capable of controlling up to 31 edge lights. Circuits to the pedestal terminal panel are provided so that as customer-supplied radio controls are installed, all that is necessary is to run wires to this point. This circuit also provides edge lighting for the rudder and aileron trim knobs.

Power for all pedestal red edge lights comes from the main dc bus. The circuit is protected by the CENTER EDGE 7.5 amp circuit breaker.

F. Emergency DC Operation

(1) The following cockpit lights are operative in emergency.

(a) Pedestal and overhead flood lights (white)

(b) Dome light (white)

(c) Left and right flight instrument panel red post lights.

(d) Standby compass light.

(2) All other cockpit lights are inoperative in emergency.

(3) The WHT. OVERRIDE switch is inoperative in emergency.

2. Major Components And Locations

A. Rheostats - on associated lighting panels (pilots, copilots and upper overhead).

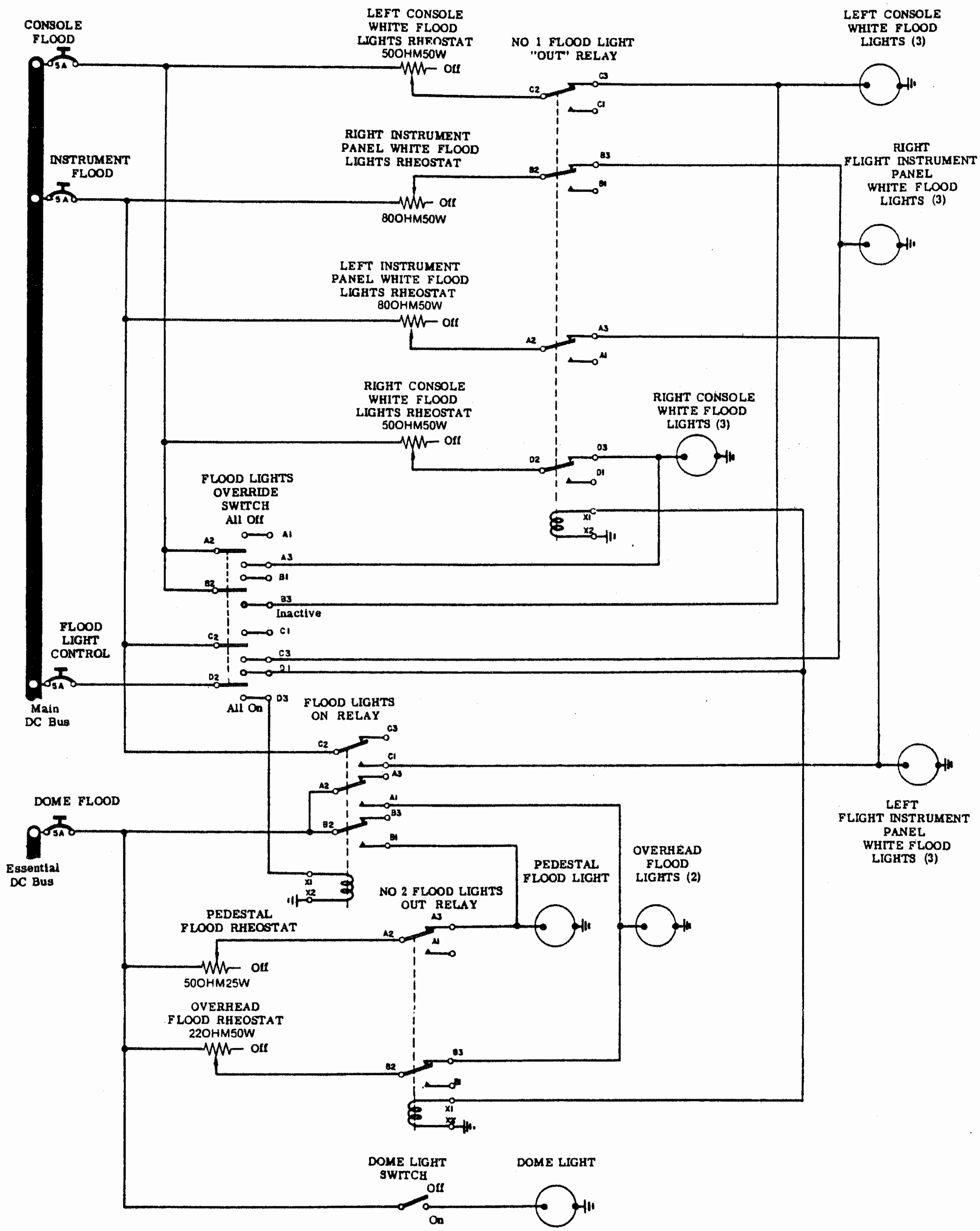
B. Map Light Switch - above respective pilots and copilots lighting control panels.

C. Dome Light and Switch - overhead in cockpit ceiling.

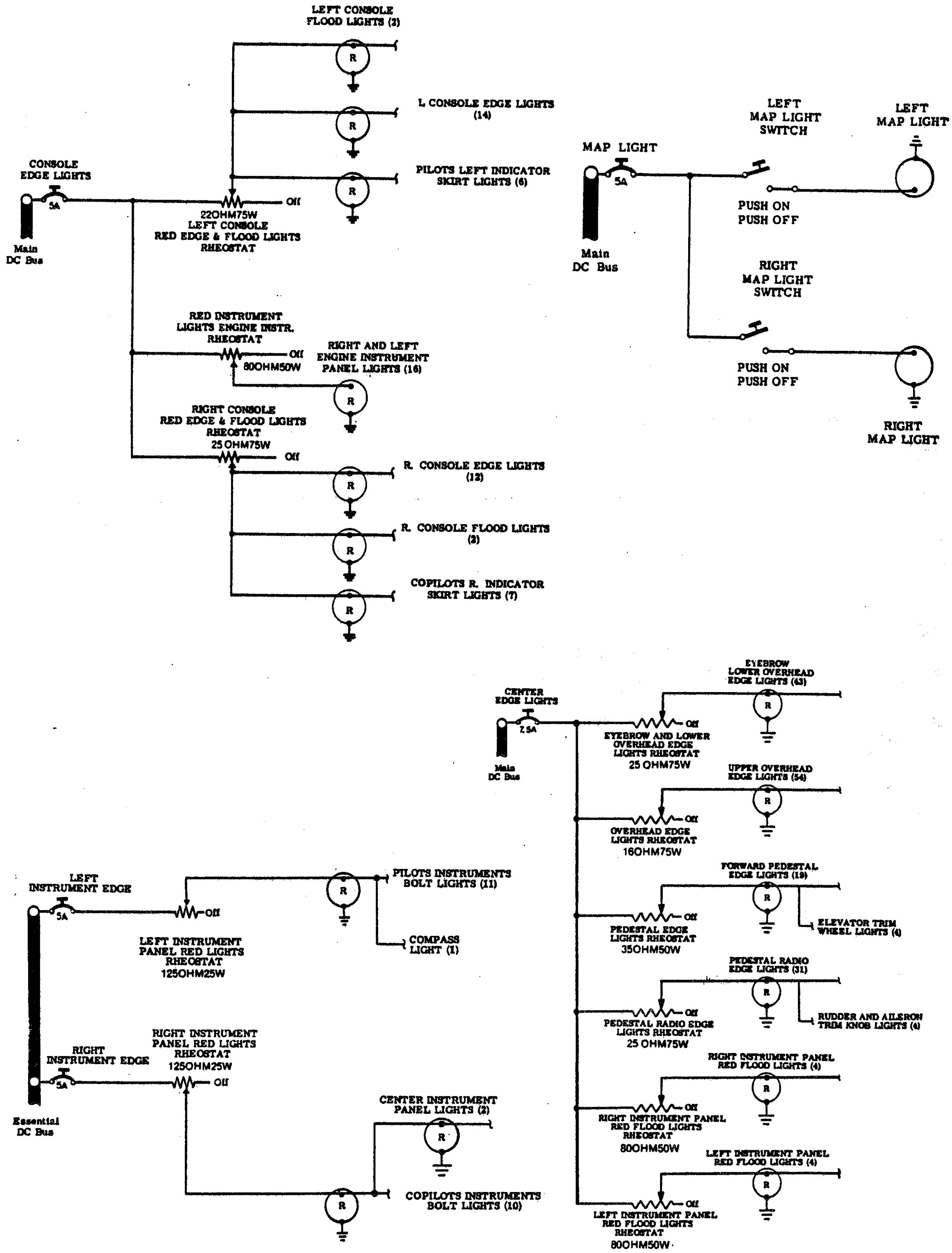
D. Flood Light ON Relay - left Fuselage Station 133 relay panel.

E. Flood Light OUT Relays No. 1 and No. 2 - left Fuselage Station 133 relay panel.

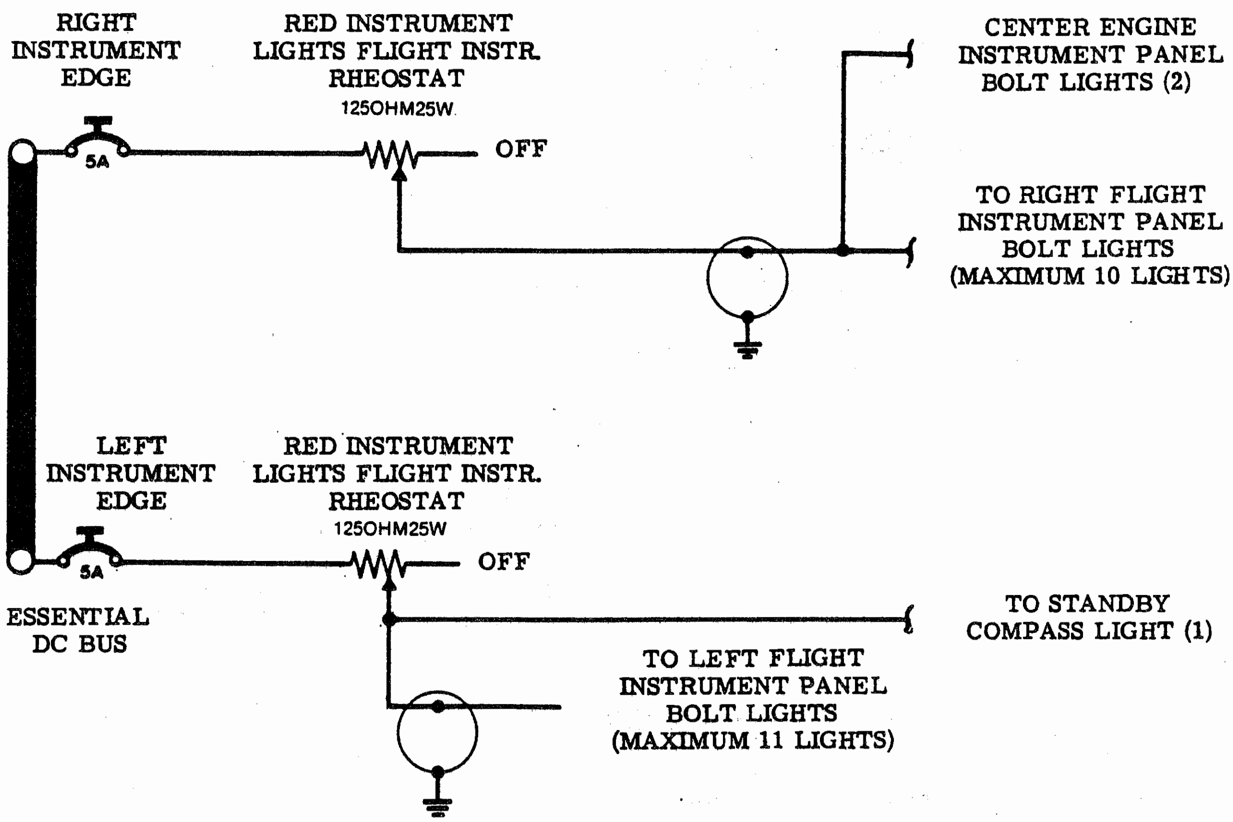
F. All associated circuit breakers are located on the left Fuselage Station 133 circuit breaker panels.



Cockpit Lights — Schematic
Figure 1.



Cockpit Lights — Schematic
Figure 2.



Cockpit Lights — Schematic
Figure 3.

INTERIOR LIGHTS (AIRCRAFT 87 - 200, 322 AND 323 HAVING ASC 115) — MAINTENANCE PRACTICES

1. **General Maintenance Precautions**

   A. Edge lighting panels are internally wired, and extreme caution should be used when drilling or altering holes in these panels, or when applying paint. Obtain detailed print of panel for exact wire routing before drilling any holes.

   B. Application of paint in the engraved groove will prevent the light from shining through engravings when required, defeating the purpose of the panels.

   C. All edge lighting panels bulb replacements must be GRIMES No. A-9906-1 or equivalent, since a special combination bulb-socket is used. A No. 327 type will not fit, except in pedestal edge light system which uses 327 bulbs.

   D. Edge lighting panel bulbs must be installed from the aft side of the panel, requiring the removal of the panel for bulb replacement, except for pedestal edge light panels.

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EMERGENCY INTERIOR EXIT LIGHTS — DESCRIPTION / OPERATION

1. Emergency Exit Lights

Self-contained, battery operated emergency exit lights are located above the main door and stair and at other strategic places in the cabin. These lights come on manually by using a switch at each fixture or automatically when a sufficient impact force trips the mechanism. The "g" sensing device is contained in a small cylinder within the light unit. To be effective, this device must always be mounted with the red dot forward. It is possible to adopt standard fixtures for left or right mounting. Activation of the sensing unit takes place when it is subjected to a force of 1 1/2 or more "g"s for at least 1/30 of a second. Since the emergency exit lights are powered by internal batteries their availability is entirely independent of aircraft electrical power. These batteries should be periodically checked and replaced as necessary.

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EMERGENCY INTERIOR EXIT LIGHTS — MAINTENANCE PRACTICES

1. **Emergency Exit Lights — Removal / Installation**

   A. Removal

      (1) Gain access to light(s).

      (2) Remove cover screws.

      (3) Remove batteries.

      (4) Remove container.

   B. Installation

      (1) Install container.

      (2) Install Type D batteries.

      (3) Check position of inertia cylinder, red dot must face forward.

      (4) Turn manual switch ON and inspect lights for brilliance, replace bulbs if necessary.

      (5) Turn manual switch OFF.

      (6) Inspect area for presents of foreign objects.

      (7) Replace cover plate.

2. **Emergency Exit Light Battery Forward/Aft — Removal / Installation**

   A. Removal

      (1) Remove cover plate, retain plate and hardware.

      (2) Remove and discard batteries.

      (3) Inspect battery case for signs of leakage or corrosion. If required, replace with new assembly.

   B. Installation

      (1) Install new Industrial Type D batteries.

      (2) Check bulbs and batteries with MANUAL switch.

      (3) Inspect area for presents of foreign objects.

      (4) Install cover plate using hardware previously removed.

3. **Emergency Exit Lights Inertia Switch — Inspection**

   A. Remove and retain cover.

   B. Check inertia switch cylinder; red dot must face forward.

   C. Inspect area for presents of foreign objects.

   D. Install cover.

   E. Perform Emergency Exit Lights - Operational Test, see this Section.

4. **Emergency Exit Lights — Operational Test**

   A. Turn Emergency Exit Lights manual switch on and inspect lights for brilliance, replace bulbs/battery if necessary.

   B. Turn Emergency Exit Lights manual switch off.

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EXTERIOR LIGHTING — DESCRIPTION / OPERATION

1. **General**

The exterior lighting system consists of position lights, anti-collision lights, landing lights, taxi lights and wing inspection lights. All exterior lights are controlled by switches located in the lower overhead panel. Power for all exterior circuits, except the navigation lights, is from the main dc bus. Power for the navigation lights is supplied form the essential dc bus.

2. **Fuselage / Nacelle Wing Inspection Lights**

The wing inspection light system provides for illumination of the leading edge of each wing, so the icing and other general conditions of the leading edges can be observed. One fuselage lighting unit is mounted on each side of the fuselage at Fuselage Station 205. These serve to light the inboard section of each wing and the inboard side of each nacelle. One nacelle lighting unit is mounted on the outboard side of each nacelle, and serves to light the outer sections of the leading edges. All four lights are controlled by the wing switch located in the lower overhead panel in the cockpit. Power supply for these lights is from the main dc bus through the 7.5 amp wing inspection (WING INSP) lights circuit breaker. (See Figure 2) As a result the lights are inoperative during emergency operation.

NOTE: The nacelle units have special prismatic lenses which are indexed to deflect the light up, aft and outboard onto the wing leading edge. Do not interchange right and left lenses. Fuselage units have clear lenses.

3. **Landing Lights**

One retractable landing light is mounted in each aft wing-fuselage fillet. Four switches (LEFT ON/OFF and RIGHT ON/OFF, and LEFT EXTEND/RETRACT and RIGHT EXTEND/RETRACT), which are located in the lower overhead panel, provide control over these lights. One switch turns the landing light on and the other switch controls the light position. These lamps are sealed beam units each rated at 600 watts and draw 22 amps each.

NOTE: On aircraft having ASC 208, extend and retract functions for both Landing Lights are controlled by one switch. The location of the right landing light switch is used for the ID light switch.

Both landing lights operate in the same manner. When the light switch is set in the ON position, the landing light relay, located in the nacelle relay box, is energized. (See Figure 1) Through the relay contacts, power is supplied to the lamp from the main dc bus through a 35 amp current limiter, which is located in the nacelle relay box.

The landing lights are flush mounted and are extendable through a 95° arc. By setting the switch to EXTEND, the landing light housing can be extended to any position within the specified limits. As soon as either landing light begins to extend, an indicator light mounted in the cockpit lower overhead panel beneath the switches is lighted, providing visible indication that a landing light is not fully retracted.

CAUTION: THE LANDING LIGHT IS NOT AUTOMATICALLY TURNED OFF WHEN LIGHT IS FULLY RETRACTED. ENSURE TO TURN LANDING LIGHT SWITCH TO OFF.

The landing lights are totally inoperative in emergency operation, however, the cockpit indicator light does operate.

NOTE: The landing lights may be extended or retracted at speeds up to 290 knots IAS.

4. **Taxi Lights**

Two taxi lights are mounted on the nonsteerable portion of the nose landing gear to provide illumination of the area directly in front of the aircraft during taxiing. These lamps are each rated at 150 watts and draw 0.5 amps each. The taxi light control switch (TAXI ON/OFF) is located in the lower overhead panel.

When the taxi light control switch in set to ON, the taxi light relay located in the mid relay box is energized. Power is supplied to the taxi lights from the main dc bus through the 25 amp TAXI LT. circuit breaker and the relay contacts. (See Figure 1)

CAUTION: THE TAXI LIGHTS ARE NOT AUTOMATICALLY TURNED OFF WHEN THE NOSE GEAR IS RETRACTED. ENSURE TO TURN TAXI LIGHT SWITCH OFF WHEN NOT IN USE.

Taxi lights are inoperative in emergency operation.

5. Navigation Lights

A. Anti-collision Lights

There are three red, rotating-type anti-collision lights which contain identical motor driven mechanisms. The top anti-collision light is installed at the top forward part of the fin, the aft anti-collision light is installed in the tail cone and the bottom anti-collision light is installed in the fuselage bottom at Fuselage Station 503. All three anti-collision lights have filters incorporated in their motor circuits for the suppression of radio noise. The bottom and the aft anti-collision lights are of identical construction. The top anti-collision light is larger, and is a vibration-resistant type. Power for these lights is supplied by the main dc bus through three 7.5 amp circuit breakers. The lights are controlled by the anti-collision lights switch (ANTI-COLL., ALL ON/BOTTOM ONLY) located in the lower overhead panel.

Placarded switch positions are ALL ON, OFF, and BOTTOM ONLY. When the anti-collision lights switch is set to ALL ON, all three lights will come on and rotate. When the switch is set to BOTTOM ONLY, the aft and bottom anti-collision lights operate. (See Figure 3) Anti-collision lights do not operate in emergency operation.

B. Position Lights.

Three position lights are located as follows: One green light is installed in the right wing tip; one red light is installed in the left wing tip; one white light is installed in the top aft part of the fin. The position lights are controlled by the NAV ON/OFF lights switch located in the lower overhead panel. When the switch is in the ON position, power is supplied from the main dc bus to the position lights through the 7.5 amp navigation lights (NAV.) circuit breaker. (See Figure 3) When the system is activated, all three lights come on and the warning lights dimming relay is energized causing the master caution display lights to dim.

C. Emergency dc Operation.

Navigation lights are available for operation in emergency.

6. Identification Lights

Two ID lights are provided on aircraft having ASC 208. These ID lights are located in the chin cowling of each engine. Power is provided for ID lights by the 28V dc bus via the MAIN DC BUS SUPPLY circuit breakers and routed through the ID LIGHT CONTROL, LEFT ID LIGHT and RIGHT ID LIGHT circuit breakers. The ID LIGHTS control switch will energize the ID lights control relay which closes individual contacts allowing power from LEFT and RIGHT ID LIGHT circuit breakers to turn on the ID lights. Lights may be controlled individually by circuit breakers. (See Figure 4)

Control switch for ID lights is located in the LANDING LIGHTS section of the overhead panel and is placarded ID LIGHTS. This switch replaces the RIGHT EXTENDED/RETRACT SWITCH for aircraft landing lights. Both landing lights are controlled by one switch.

EXTERIOR LIGHTING — MAINTENANCE PRACTICES

1. **Exterior lights — Operational Test**

A. Connect a 28V external source of dc power to system.

B. Position/Anti-collision Lights

(1) Depress TOP AC, AFT AC, BOTTOM AC, NAV, MA WARN and WARN LTS BUS circuit breakers.

(2) Set anti-collision lights switch to ALL ON. Top, bottom and aft anti-collision lights should operate.

(3) Set anti-collision lights switch to BOTTOM ONLY. Bottom and aft anti-collision lights should operate. Top collision lights should go off.

(4) Set anti-collision lights switch to OFF. The three anti-collision lights should go off.

(5) Set NAV lights switch to ON. The left and right wing tip lights and the tail light should burn continuously at maximum intensity. The cockpit warning lights should become dimmed and burn at reduced intensity.

C. Landing/Taxi Lights

(1) Depress the TAXI LT CONT, TAXI LT PWR, L LDG LT CONT and R LDG LT CONT circuit breakers.

(2) Inspect right and left landing light current limiters located in the right and left nacelles.

(3) Set left landing light switch to ON. Left landing light should come on.

(4) Set left landing position switch to EXTEND. Left light housing should extend fully. As soon as landing light begins to extend, the indicator light mounted in the cockpit lower overhead panel beneath the switches should come on.

(5) Set left landing light switch to OFF. Left landing light should go off.

(6) Set left landing light position switch to RETRACT. Left light housing should retract flush with skin and warning light at left end of lower overhead panel should go out.

(7) Set right landing light switch to ON. Right landing light should come on.

(8) Set right landing light position switch to EXTEND. Right light housing should extend fully. As soon as landing light begins to extend, the indicator light in the cockpit lower overhead panel beneath the switches should come on.

(9) Set right landing light switch to OFF. Right landing light should go off.

(10) Set right landing light position switch to RETRACT. Right light housing should retract flush with skin and warning light at left end of lower overhead panel should go out.

(11) Set taxi light control switch to ON. Both taxi lights mounted on the nose landing gear should come on.

(12) Set taxi lights control switch to OFF. Both taxi lights should go off.

D. Fuselage/Nacelle Wing Inspection Lights

(1) Close the wing inspection lights (WING INSP.) circuit breaker.

(2) Set wing inspection lights switch to ON. The four lights should come on.

(3) Set wing inspection lights switch to OFF. The four lights should go off.

(4) Wing inspection lights are bayonet type sealed-beam bulbs. When changing bulbs ensure to install correct lens, because various lenses control direction of light.

E. Identification Lights

NOTE: Applies to aircraft having ASC 208 only.

(1) Close ID LIGHT CONT, R ID LT PWR, L ID LT PWR and MAIN DC BUS FEEDER (A and B) circuit breakers.

(2) Select ID LIGHTS switch to ON, both ID Lights should come on.

(3) Select ID LIGHTS switch to OFF, both ID Lights should go off.

F. Strobe Light

(1) Engage strobe lights circuit breakers and turn aircraft power on. Check operation of lights.

(2) Turn strobe light switch to OFF.

2. Anti-collision Bulbs — Replacement

A. Lower Fuselage Light.

(1) Loosen screw that holds clamp around lens of light.

(2) Remove clamp and red lens form aircraft.

(3) Twist bulbs and lift them out.

(4) Inspect for foreign objects.

(5) Replace bulbs, lens and clamp.

(6) Tighten clamp with screw.

B. Aft Light

(1) Remove Plexiglas tail cap cover.

(2) Loosen screw holding clamp around lens.

(3) Remove clamp and lens from aircraft.

(4) Twist bulbs and lift them out.

(5) Inspect for foreign objects.

(6) Replace bulbs, lens and clamp.

(7) Tighten clamp with screw.

(8) Inspect area for foreign objects.

(9) Install Plexiglas tail cap cover.

C. Top Light.

WARNING: TO PREVENT INJURY TO MAINTENANCE PERSONNEL, PLACE A PLACARD READING "DO NOT MOVE RUDDER" IN THE COCKPIT BEFORE STARTING WORK.

CAUTION: 1/4 INCH SPACER INSERTS ARE USED TO PREVENT OVER-TIGHTENING SCREWS AND BREAKING PLEXIGLAS. THESE SHOULD NOT BE DISCARDED.

(1) Remove screws from Plexiglas fin cover to gain access to light.

(2) Remove screws which secure cover plate to light using a short common screwdriver.

(3) Remove lens, twist bulbs and lift them out.

(4) Replace bulbs and lens and install coverplate.

(5) Inspect area for foreign objects.

(6) Replace Plexiglas fin cover.

3. Navigation Lights — Replacement

A. Wing Tip Lights.

(1) Remove Plexiglas wing tip cap cover.

(2) Flip spring clip to one side and lift colored cover off.

(3) Twist bulb and lift it out.

(4) Replace bulb.

(5) Inspect area for presents of foreign objects.

(6) Replace colored cover and spring clip.

(7) Inspect area for foreign objects.

(8) Install wing tip cap cover.

B. Tail Light.

(1) Flip spring clip to one side.

(2) Remove clear cover.

(3) Twist bulb and lift it out.

(4) Inspect area for foreign objects.

(5) Replace bulb and clear cover.

(6) Replace spring clip.

4. Landing Lights — Replacement

A. Remove screws which hold rim to frame.

B. Extract sealed beam bulb and disconnect electrical connection.

C. Replace bulb.

D. Inspect area for foreign objects.

E. Replace rim and install screws.

5. Anti-collision Light Assembly — Removal / Installation

A. Removal

(1) Ensure anti-collision light switch is in OFF position.

(2) Disengage anti-collision light circuit breaker.

CAUTION: WHEN REMOVING TOP ANTI-COLLISION LIGHT PLACE A PLACARD READING "DO NOT MOVE RUDDER" IN THE COCKPIT BEFORE STARTING WORK. PLACE GUST LOCK IN LOCKED POSITION.

1/4 INCH SPACER INSERTS ARE USED TO PREVENT OVER-TIGHTENING SCREWS AND BREAKING PLEXIGLAS. THEY SHOULD NOT BE DISCARDED.

(3) Remove screws and angle plates that secure Plexiglas fin cover.

(4) Remove screws securing Plexiglas tail cap cover.

(5) Remove screws that secure mounting plate to fuselage. Removing mounting plate scribe location mark to facilitate installation.

(6) Disconnect electrical leads.

(7) Remove screws that secure anti-collision light assembly and remove assembly.

(8) Loosen and remove retaining ring and lens from old unit. Remove and discard lamps.

B. Installation

(1) Clean old zinc chromate paste or tape form mounting areas.

(2) Install assembly on mounting plate and connect electrical leads.

NOTE: Bottom anti-collision light must be sealed between assembly and mounting plate to prevent collection of moisture in light assembly.

(3) Apply electric power to aircraft and engage anti-collision light circuit breaker.

(4) Position anti-collision light switch to ON, to ensure light operation before installing Plexiglas cover.

(5) Turn off electrical power.

(6) Inspect area for foreign objects.

(7) Install top and aft anti-collision light Plexiglas cover.

(8) Apply a thin layer of zinc chromate to top cover mounting screw holes.

(9) Position Plexiglas cover over mounting and install spacers.

(10) Install angle plate and secure with screws hand tight.

(11) Inspect area for presents of foreign objects.

(12) Install bottom anti-collision light Plexiglas cover.

NOTE: Mounting plate for bottom anti-collision light is part of pressure vessel. Care must be exercised to seal plate thoroughly using zinc chromate tape. Install plate and screws. Clean excess zinc chromate tape from fuselage.

(13) Depress TOP AC, AFT AC, BOTTOM AC, MA WARN and WARN LTS BUS circuit breakers.

(14) Set anti-collision lights switch to ALL ON. Top, bottom and aft anti-collision lights should operate.

(15) Set anti-collision lights switch to BOTTOM ONLY. Bottom and aft anti-collision lights should operate. Top collision lights should go off.

(16) Set anti-collision lights switch to OFF. The three anti-collision lights should go off.

6. Landing Light — Removal / Installation

A. Removal

(1) Place landing light switch to OFF position.

(2) Place landing light RETRACT/EXTEND switch to OFF position.

(3) Disengage landing lights, left and right circuit breakers.

(4) Remove landing light fairing cover. Disconnect antenna lead and remove fairing cover.

(5) Remove landing light screws securing light to its mount.

(6) Disconnect electrical connector. Remove landing light assembly.

B. Installation

(1) Connect electrical connector and install landing light assembly.

(2) Turn power on engage landing light control circuit switch breakers and select ENTEND on landing light switch, light should extend fully. Turn switch ON, light should come on. Turn light OFF. Turn switch to RETRACT. Light should retract. Ensure switch light operation before installing and fairing.

(3) Connect antenna lead, and install fairing.

(4) Return system to normal.

7. Strobe Light Power Supply — Removal / Installation

A. Removal

WARNING: HIGH VOLTAGE IS INVOLVED IN THE CIRCUIT BETWEEN POWER SUPPLY AND LIGHT ASSEMBLY. ALTHOUGH A BLEED OFF RESISTOR IS INCORPORATED IN THE POWER SUPPLY CIRCUIT THE STROBE LIGHT SYSTEM SWITCH SHOULD BE PLACED IN THE OFF POSITION AND A MINIMUM PERIOD OF TEN MINUTES ALLOWED TO ELAPSE BEFORE CABLES ARE DISCONNECTED AT POWER SUPPLY, OR AT LIGHT ASSEMBLY OR BEFORE THESE UNITS ARE HANDLED OR, DISASSEMBLED IN ANY WAY.

(1) De-energize electrical power.

(2) Disengage strobe light circuit breakers.

(3) Locate and remove screws securing power supply to chassis.

(4) Disconnect electrical plugs.

(5) Remove power supply.

B. Installation

(1) Install unit in chassis.

(2) Connect electrical plug.

(3) Engage strobe lights circuit breakers and turn aircraft power on. Check operation of lights.

(4) Turn strobe light switch to OFF.

8. Strobe Light Timer — Removal / Installation

A. Removal

(1) Ensure electrical power is off.

(2) Pull strobe light circuit breakers.

(3) Locate timer and remove screws securing timer to chassis.

(4) Disconnect electrical plugs.

(5) Remove timer.

B. Installation

(1) Install timer on chassis.

(2) Connect electrical plugs.

(3) Return system to normal configuration.

(4) Check operation of timer.

9. Strobe Light — Removal / Installation

A. Removal

WARNING: HIGH VOLTAGE IS INVOLVED IN THE CIRCUIT BETWEEN POWER SUPPLY AND LIGHT ASSEMBLY. ALTHOUGH A BLEED OFF RESISTOR IS INCORPORATED IN THE POWER SUPPLY CIRCUIT THE STROBE LIGHT SYSTEM SWITCH SHOULD BE PLACED IN THE OFF POSITION AND A MINIMUM PERIOD OF TEN MINUTES ALLOWED TO ELAPSE BEFORE CABLES ARE DISCONNECTED AT POWER SUPPLY, OR AT LIGHT ASSEMBLY OR BEFORE THESE UNITS ARE HANDLED OR, DISASSEMBLED IN ANY WAY.

(1) De-energize electrical power.

(2) Disengage strobe light circuit breakers.

(3) Remove Plexiglas cover. Extreme care should be taken to avoid damaging Plexiglas.

(4) Disconnect electrical connection to light. Remove mounting screws from light assembly. Remove light assembly.

B. Installation

(1) Install light assembly. Mount unit with hardware.

(2) Connect electrical plug.

(3) Before installing Plexiglas cover, engage strobe lights circuit breakers and turn aircraft power on. Check operation of lights.

(4) Turn strobe light switch to OFF, pull strobe light circuit breaker and turn aircraft power OFF.

CAUTION: DO NOT USE A SCREWGUN TO INSTALL SCREWS.

(5) Inspect area for foreign objects.

(6) Install Plexiglas cover. Do not overtighten screws.

10. ID Light — Removal / Installation

A. Removal

(1) De-energize electrical power.

(2) Remove chin cowling.

(3) Remove and retain lower bolts. Light assembly should now slip out to the side of the bracket.

(4) Remove retainer from light housing.

(5) Remove old lamp.

B. Installation

(1) Place new lamp in housing and install retainer.

(2) Slip housing into place on mounting bracket and secure with bolts.

(3) Close ID LIGHT CONT, R ID LT PWR, L ID LT PWR and MAIN DC BUS FEEDER (A and B) circuit breakers.

(4) Select ID LIGHTS switch to ON, both ID Lights should come on.

(5) Select ID LIGHTS switch to OFF, both ID Lights should go off.

(6) Inspect area for foreign objects and security of all attachments.

(7) Install chin cowling.

LIST OF EFFECTIVE PAGES

CHAPTER 34

NAVIGATION

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** Pages added by Revision 49
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CHAPTER 34

NAVIGATION

NAVIGATION — DESCRIPTION / OPERATION

1. General

A. Instruments

Instruments, their systems, installation and operation differ in equipment installed. Refer to appropriate manufacturer handbook for information concerning maintenance, fault isolation, installation and operation of equipment.

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PITOT AND STATIC PRESSURE SYSTEMS — DESCRIPTION / OPERATION

1. General

(See Figure 1)

The pitot system obtains its pressure from heads of the pitot tubes. The static system obtains its pressure from two static vents (left and right) located on fuselage just forward of horizontal stabilizer. From this source, static air is teed together and piped forward to pilot static selector valve. From the static selector valve air is directed by a manifold to pilot airspeed indicator, altimeter and rate of climb indicator. The copilot static or atmospheric source of air is admitted through its own pair of individual flush static ports and is identical in operation to the pilot static system. (See Figure 1)

As an alternate means of obtaining static pressure if the ports become blocked, static selector valves on pilots and copilots instrument panels, permit ambient air, forward of the cockpit nose section, to be used as the source of static pressure. The pilot and copilot may select an alternate source of static air by individual selection of a valve mounted on the extreme outboard portion of each flight instrument panel. The pilot selector valve controls only the pilot static instruments; the copilot selector valve controls only the copilot static instruments. The drains for the alternate system are located at Fuselage Station 75, left and right side, just forward of the instrument panel. (See Figure 1)

Minor deformations inside the static ports are acceptable and should not have any affect on airspeed and altimeter readings, nor should any autopilot functions, as controlled by the Air Data Computers (ADC), be adversely affected. This internal deformation should not be confused with deformations exterior to the static port. Contour of the airframe in the vicinity of the static port should be smooth and clean. Minor deformations within an inch or two, forward or aft, can cause severe errors in the static pressure input to aircraft systems. It is recommended that maintenance personnel use caution when inserting and removing VPT static port adaptors to preclude damage (internal or external) to the static ports.

For Aircraft 1 - 35 and 114; the recommended source for auto pilot static is located on copilot supply line Tee connection under FLR-2. For Aircraft 36 - 200, 322 and 323 excluding 114, the recommended source for autopilot static is located at the drains at Fuselage Station 135 and is tapped off copilot supply line.

The pitot heads are equipped with electrical heating elements for icing protection. Indication of heat being supplied to each pitot heater is indicated by two green lights located by the pitot heat DE-ICING switch on the lower left corner of the upper overhead panel. See to Chapter 30 for complete details on pitot heat system.

2. Airspeed Warning System

NOTE: The following information is applicable to Aircraft 1 - 124, 322 and 323 having ASC 152 and Aircraft 125 - 200.

A. Description

An aural speed warning device is incorporated to alert the crew when the MAXIMUM OPERATING SPEED (Vmo or Mmo) is reached.

The aural speed warning system installed is a sealed diaphragm operated device known as the Speed Warning Sensor, which receives pitot and static information from the copilot airspeed indicator system and translates this information to close an internally incorporated relay when the Vmo speed is reached. Electrically the relay completes a circuit to a cricket type warning horn which then alerts the crew. The system also includes a test button mounted adjacent to the horn which when depressed, completes the horn circuit for testing purposes. (See Figure 2)

Power for this system comes from essential dc bus through the SPEED WARN circuit breaker on pilot circuit breaker panel. It is fully operative in emergency dc.

B. Location of Components;

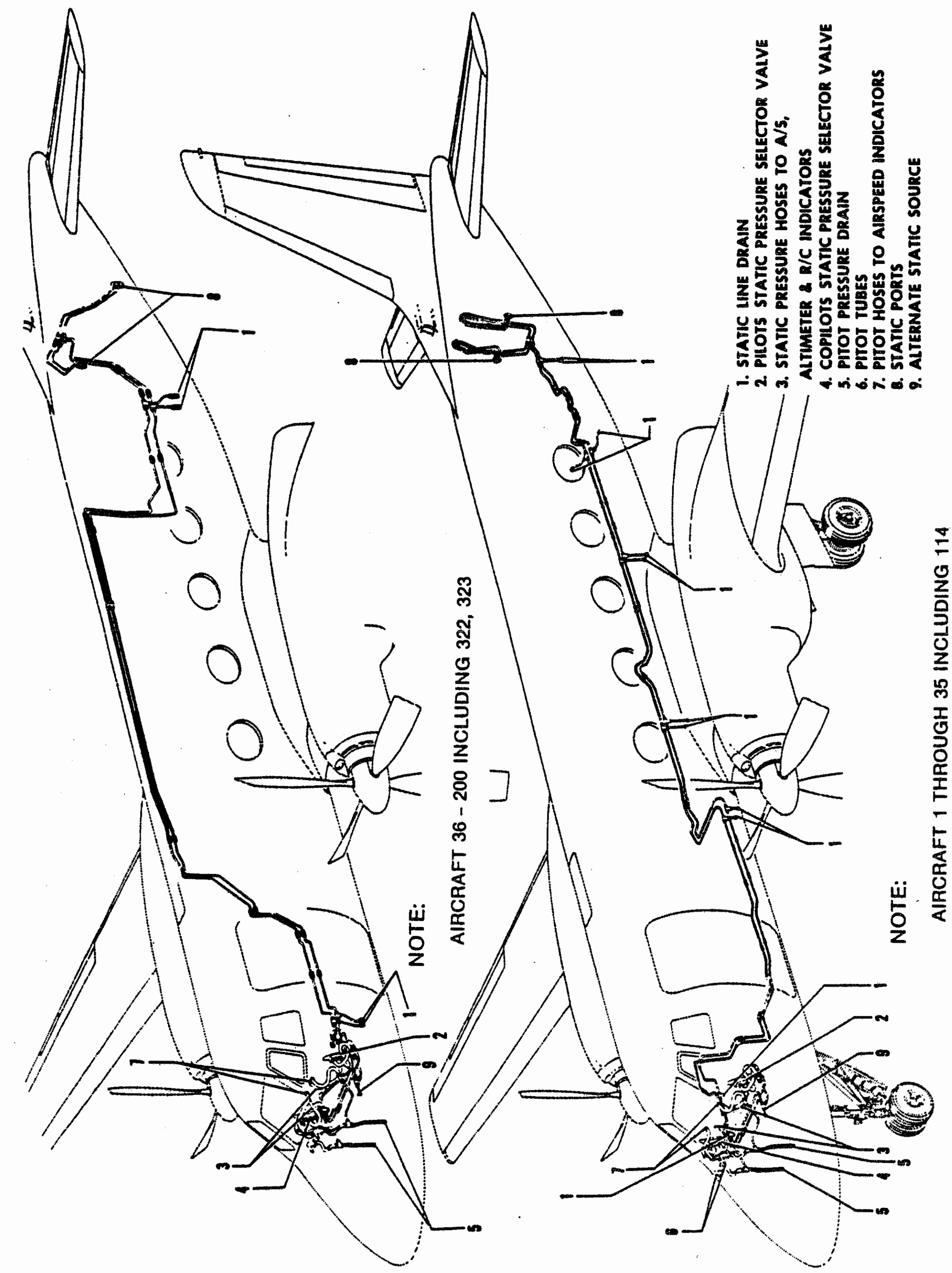
UNIT	NO PER A/C	LOCATION
Speed Warning	1	In cockpit, forward of copilot console, at approximately Fuselage Station 92.
Speed Warning Horn	1	Upper overhead panel, cockpit.
Speed Warning Test Switch	1	Upper overhead panel, adjacent to horn.
SPEED WARN Circuit Breaker	1	Pilot circuit breaker panel, cockpit.

C. Detailed Description of Operation (See Figure 3)

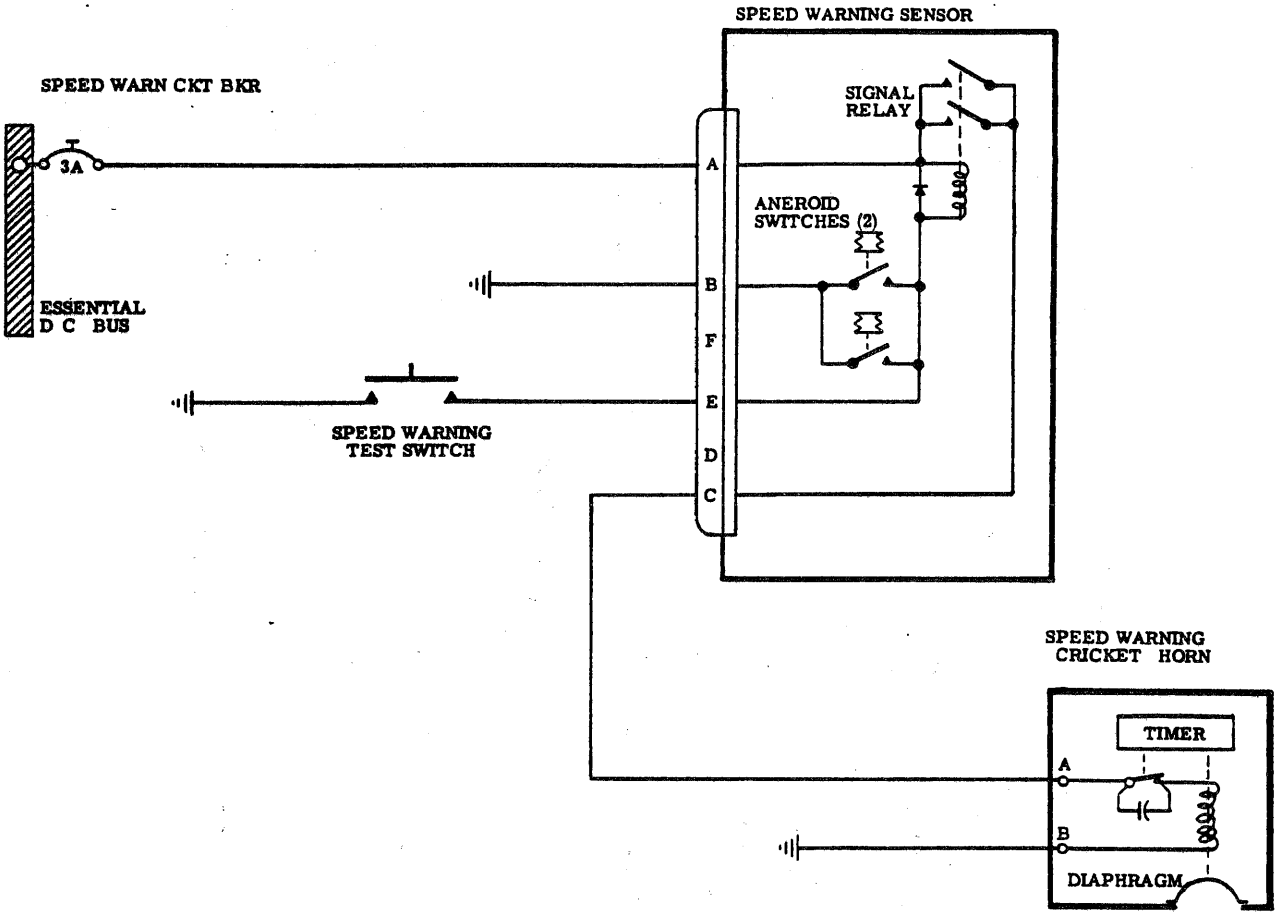
Pitot pressure is obtained from the copilots pitot head. the sensor pitot line is teed into the line forward of the copilots console panel. This pitot pressure is applied to the Pt port of the speed warning sensor. Static pressure is obtained from the copilots normal static system (tail ports) through a tee connector forward of the copilots console.

This line terminates at the P port of the speed warning sensor unit. The sensor, through internal diaphragm operated devices, evaluates the airspeed information and when the device senses 290 to 296 knots below 12,000 ft., or MACH 0.54 to MACH 0.55 above 12,000 ft., the internal relay closes. Power from the essential dc bus, through the SPEED WARN circuit breaker is applied through the contacts of the sensor relay to the horn, causing it to sound.

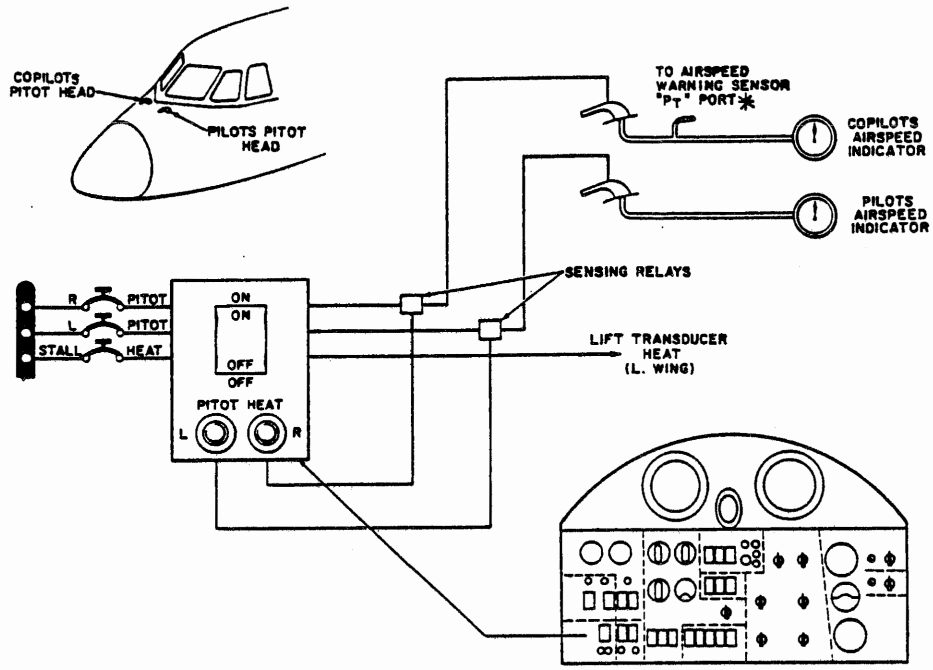
When the situation is resolved, the relay breaks and silences the horn. A test switch is located adjacent to the horn. When depressed, the switch applies a ground to cause the relay to close and, if power is applied, the horn will sound indicating that the system is operative. Releasing the test switch restores the system to normal operation. The sensor is mounted on a bracket attached to structure, forward of the copilot console, and the electrical connection to it is by means of an electrical plug. The horn basically consists of a cricket type diaphragm, mechanically connected to an electrical solenoid which is vibrated by the action of an internal timer and an electrical contact. Application of electrical power to the horn from the sensor causes the diaphragm to oscillate, emitting cricket like sounds.



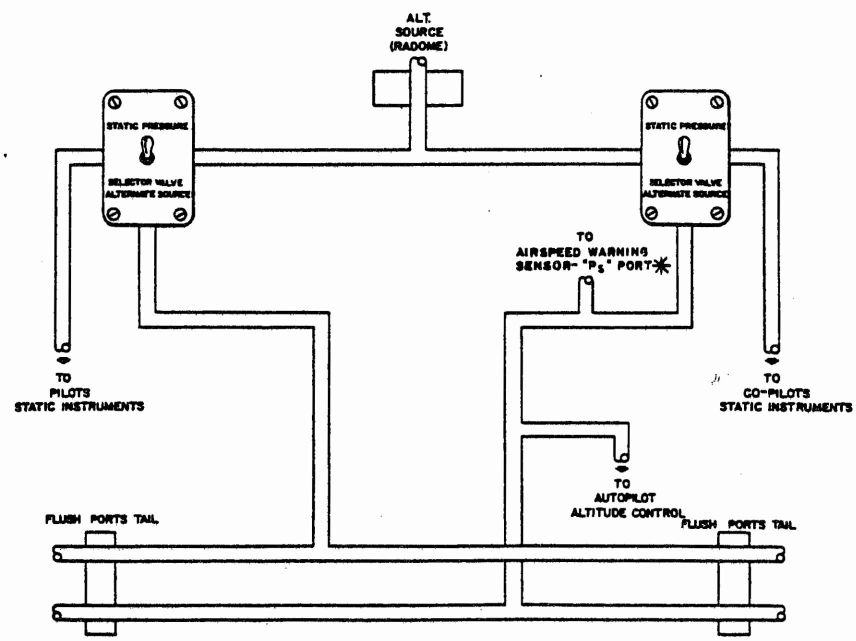
Pitot/Static Installation
Figure 1.



Airspeed Warning System — Electrical Schematic
Figure 2.



*AIRCRAFT 1 - 124 INCLUDING 322,
323 HAVING ASC 152 AND
AIRCRAFT 125 - 200



Pitot/Static System — Functional Schematic
Figure 3.

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PITOT AND STATIC PRESSURE SYSTEMS — FAULT ISOLATION

FAULT	POSSIBLE CAUSE	CORRECTION
Pitot head does not heat.	Defective heating element.	Replace pitot head.
	Loose electrical connection.	Tighten connection.
	Faulty wiring.	Repair or replace wiring.
Incorrect or erratic readings.	Leaky tubing.	Inspect tubing and tighten fittings. Perform leak test.
	Defective indicator.	Replace indicator.
	Ice or water in lines.	Drain and purge system. (See this chapter.
No reading obtainable.	Obstruction in pitot head.	Clean pitot head.
	Obstruction in system tubing.	Purge system. (See this Chapter)
	Ice or water in lines.	Drain and purge system. (See this Chapter)
	Obstruction in static ports.	Clean and purge system.

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PITOT AND STATIC PRESSURE SYSTEMS — MAINTENANCE PRACTICES

1. **Pitot / Static Pressure Systems — Servicing**

   A. Pitot Pressure Lines

   WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

   (1) Install nose gear ground safety locks.

   (2) Open nose wheel well door utilizing the ground service valve and install ground safety door lock. (See Section 12-0)

   (3) In nose wheel well remove pilots pitot drain line cap at Fuselage Station 61.50 left side.

   (4) Drain all accumulated water from this line and install cap.

   (5) Remove copilots drain line cap at Fuselage Station 62.50 right side.

   (6) Drain all accumulated water from this line and install cap.

   (7) Inspect pitot heads. Ensure that drain holes on bottom of head are open.

   (8) Check system for tightness of connection.

   B. Static Pressure Lines.

   (1) Disconnect all static air lines from pilot instrument panel.

   CAUTION: ENSURE THAT ALL STATIC CONNECTIONS TO INSTRUMENTS ARE DISCONNECTED.

   (2) Use shop air to blow lines clear (approximately 90 pounds).

   (3) Disconnect all static air lines from copilots instrument panel.

   (4) Disconnect static air lines from autopilot amplifier or altitude control.

   (5) Disconnect all static lines from cabin instruments.

   CAUTION: ENSURE THAT ALL INSTRUMENTS AND UNITS ARE DISCONNECTED FROM COPILOTS STATIC AIR LINES.

   (6) Use shop air to blow lines clear.

   (7) Blow alternate static air lines clear.

   (8) Reconnect all lines and check system for tightness of all connections.

2. **Static Pressure System — Draining**

   NOTE: Removable AN-4 caps are provided on the normal static drain locations (low points in the system), to facilitate draining and purging. (See Figure 201)

   A. Drain static air lines. On Aircraft 1 - 35 and 114 the location of drain fittings are as follows:

   (1) Alternate static air line at Fuselage Station 75 - Pilots disconnect bracket.

   (2) Alternate static air line at Fuselage Station 75 - Copilots disconnect bracket.

   (3) Pilots static air line at Fuselage Station 93 - right console.

   (4) Pilots static air line at Fuselage Station 257 - left side below floor.

   (5) Copilots static air line at Fuselage Station 257 - left side below floor.

   (6) Pilots static air line at Fuselage Station 320 - right side through access hole in skin.

   (7) Copilots static air line at Fuselage Station 320 - right side through access hole in skin.

   (8) Pilots static air line at Fuselage Station 416 - left side below floor.

   (9) Copilots static air line at Fuselage Station 416 - left side below floor.

   (10) Pilots static air line at Fuselage Station 525 - left side below floor.

(11) Copilots static air line at Fuselage Station 525 - left side below floor.

(12) Pilots static air line at Fuselage Station 616 1/2 - left side below floor.

(13) Copilots static air line at Fuselage Station 616 1/2 - left side below floor.

B. On Aircraft 36 - 200, 322 and 323, the locations of the drain fittings are as follows:

(1) Alternate static air line at Fuselage Station 75 - Pilots disconnect bracket.

(2) Alternate static air line at Fuselage Station 75 - Copilots disconnect bracket.

(3) Pilots static air lines at Fuselage Station 135 - right side above floor, by the outflow valves.

(4) Copilots static air line at Fuselage Station 135 - right side above floor, by the outflow valves.

(5) Pilots static air line at Fuselage Station 556 - right side above floor (baggage compartment).

(6) Copilots static air line at Fuselage Station 556 - right side above floor (baggage compartment).

3. Pitot Head — Removal / Installation

A. Removal

(1) Gain access to disconnect and cap electrical and air line connections.

(2) Remove mounting hardware and pitot head.

B. Installation

(1) Connect air lines and electrical connections to pitot head.

(2) Energize main and essential dc bus.

NOTE: Do not leave pitot heat switch ON for more than 20 seconds.

(3) Select pitot heat switch to ON; observe that appropriate indicator light comes on. Immediately touch pitot head lightly (do not grab). The pitot head will get extremely hot in a very short time.

(4) De-energize main and essential dc bus.

(5) Perform pitot system portion of Pitot System - Leak Check/Functional Test, this Section.

4. Overspeed Warning System — Functional Test

A. Perform copilots Pitot/Static System - Leak Check/Functional Test before Overspeed Warning System - Functional Test, this Section.

B. Apply power to aircraft.

C. Depress SPEED WARNING TEST switch, horn should sound; release TEST switch, horn should silence.

D. Apply a pressure source to copilots pitot head until speed warning horn sounds; note copilot airspeed indicator, horn should sound between 290 and 296 knots. Aircraft having ASC 240 (GIC), horn will sound between 260 and 266 knots.

CAUTION: WHEN APPLYING VACUUM, OBSERVE COPILOTS AIRSPEED INDICATOR. IF POINTER MOVES OFF ZERO, STOP AND CHECK ALL LINES FOR LEAKAGE.

E. Apply a vacuum equivalent to 30,000 feet on master altimeter to both copilots pitot and static system; simultaneously at 30,000 feet the difference between master and copilots altimeter should not exceed 360 feet.

F. Close off shutoff valve downstream of pitot connection and slowly open bleed valve downstream of same connection until master airspeed indicator reads between MACH 0.54 - 0.55, with power on, overspeed aural warning horn should sound, indicating that warning system is operating properly at high altitude.

G. Slowly reduce vacuum to zero at a rate not to exceed 1000 feet per minute.

H. Apply a vacuum equivalent to 20,000 feet to copilots pitot and static system simultaneously, and close shutoff valve; slowly open bleed valve until airspeed indicator reads between 246 and 251 knots, aural warning should actuate.

I. Slowly reduce vacuum to zero at a rate not to exceed 1000 feet per minute.

5. Airspeed Warning Sensor — Removal / Installation

A. Removal

(1) Gain access to, disconnect and cap air lines and electrical plug.

(2) Remove mounting hardware and remove sensor.

B. Installation

(1) Ensure adequate insulation exists between skin of aircraft and sensor.

(2) Install and secure sensor.

(3) Connect air lines and electrical plug to sensor.

(4) Perform Copilot Pitot/Static System - Leak Check/Functional Test this Section.

(5) Perform Overspeed Warning - Functional Test, this Section.

(6) Inspect area for presents of foreign objects, security of all attachments.

(7) Install access covers previously removed.

6. Speed Warning "Cricket" Horn — Removal / Installation

A. Removal

(1) Gain access on upper overhead panel and disconnect electrical leads.

(2) Remove mounting hardware and horn.

B. Installation

(1) Install and secure warning horn.

(2) Connect electrical leads.

(3) Energize ESS dc bus and check "CRICKET" horn by pushing OVERSPEED TEST button.

7. Pitot/Static System — Leak Check / Functional Test

CAUTION: DO NOT APPLY VACUUM TO PITOT OR STATIC LINES WHEN PERFORMING THIS TEST.

A. Pitot System.

(1) Ensure that all equipment / units are installed properly and connections are tight.

(2) Connect suitable pressure source tester to pilot/copilot pitot head to produce indication of 290 to 296 knots on master airspeed indicator. Aircraft airspeed indicators should be within 5 knots of each other.

(3) Close pressure source.

(4) After one minute indicated air speed shall not decrease more than 5 knots.

(5) Slowly decrease pressure to zero and disconnect pressure source.

(6) If leak is detected isolate section of system until leak is found.

B. Static Systems.

CAUTION: DO NOT APPLY PRESSURE TO EITHER PITOT OR STATIC LINE WHEN PERFORMING THIS TEST.

(1) Ensure that all equipment / units are installed properly and connections are tight.

(2) Ensure that selector valve pertaining to system under test is in static pressure position (locked position).

(3) Seal port opening in flush static fitting on one side of aircraft only.

(4) Using test fitting or water drain connection, connect vacuum source tester to open port on opposite side of aircraft and to applicable pitot head.

(5) Interconnect both systems into a single vacuum pump.

CAUTION: WHEN APPLYING VACUUM TO SYSTEM, OBSERVE AIRCRAFT AIRSPEED INDICATOR. IF POINTER MOVES OFF ZERO POSITION, STOP TEST AND CHECK FOR LEAKS.

(6) Apply vacuum to both pitot and static system simultaneously until master altimeter indicates 15,500 feet. Difference between master altimeter and aircraft altimeter should not exceed 107 feet.

(7) Close vacuum source. After one minute indicated altitude on master altimeter shall not fall below 15,400 feet.

(8) Slowly reduce vacuum to zero at a rate of no more than 1,000 feet per minute.

(9) Disconnect vacuum source.

8. Alternate Static System — Leak Check / Functional Test

CAUTION: DO NOT APPLY PRESSURE TO EITHER PITOT OR STATIC LINE WHEN PERFORMING THIS TEST.

A. Move selector valve (No.1 or No.2) pertaining to system under test to alternate source position. Valve not being used must remain in locked position.

B. Using test fitting or water drain connection, connect vacuum source tester to alternate static source fitting in nose section of aircraft and to applicable pitot head.

C. Interconnect both systems to single vacuum pump.

CAUTION: WHEN APPLYING VACUUM TO SYSTEM, OBSERVE AIRCRAFT AIRSPEED INDICATOR. IF POINTER MOVES OFF ZERO POSITION, STOP TEST AND CHECK FOR LEAKS.

D. Apply vacuum to both pitot and static system simultaneously until master altimeter indicates 15,500 feet. Difference between master altimeter and aircraft altimeter should not exceed 107 feet.

E. Close vacuum source. After one minute indicated altitude on master altitude shall not fall below 15,400 feet.

F. Slowly reduce vacuum to zero at rate of no more than 1,000 feet per minute.

G. Disconnect vacuum source.

9. Gyro Horizon Power Supply (No.1, No.2 and No.3) — Inspection

CAUTION: REMOVE 10 AMP FUSE BEFORE REMOVING OR INSTALLING PS-823 COVERS.

NOTE: Field maintenance of the Model PS-823 power supply will be limited to fuse replacement, battery changing and performance test.

To perform inspection of the horizon system, consult J.E.T. Electronics Manual TP-200.

A. Remove battery packs and remove covers.

B. Inspect individual battery cells for indications of venting or leaking. Venting is indicated by a visible rupture near the top center of the cell or the appearance of corrosion in the same area.

C. Test battery by connecting a 10K 1/2 watt resistor and dc volt meter to PS.823 connector pins (+) and (−). The voltmeter shall read 20 - 28V dc.

D. Connect a 7 ohm, 150 watt resistor to battery pack terminals. When the terminal voltage decreases to 20V dc disconnect 7 ohm load.

E. Recharge battery pack by connecting a 200 milliampere current supply. Adjust supply to provide approximately 200ma and charge battery pack for 16 hours minimum.

10. Compass Index Error — Adjustment

NOTE: This adjustment determines and cancels out index error due to longitudinal alignment of flux wave as follows.

A. Place compass system in operation and allow at least 5 minutes for gyro to come up to speed and to slave to magnetic north. Ensure that "N-S" and "E-W" adjustments on compensator on flux valve are at zero.

B. With engines running and as much equipment operating on aircraft as would be operating in flight, see Aircraft Flight Manual for operating engines. Position aircraft on a compass rose and turn it to each of the four cardinal headings.

C. Record differences in reading between compass indicator dial and compass rose as + or −, depending on whether compass indicator readings are greater or less than compass rose readings. Allow sufficient time for compass indicator dial to settle out at each heading before taking readings.

D. Add the error algebraically and divide by four; the result is the index error.

E. Loosen screws holding flux valve flange to its mounting surface and rotate it to cancel out index error. If error is positive, flange should be rotated counterclockwise (giving a − reading on flange), as observed from above the unit. If error is negative, rotate flange clockwise (giving a + reading on the flange). Amount of rotation should equal index error.

F. Tighten mounting screws and check readings at four cardinal headings. Calculate index error to ensure it is zero. If it is not zero, readjust flux valve flange until this error is cancel led. Any remaining error in excess of 1° which are caused by extraneous magnetic fields should be counteracted by using adjustment screws mounted on top of flux valve.

G. Record readings as shown below:

No.1 COMPASS			
N 0	E 90	S 180	W 270

No.2 COMPASS			
N 0	E 90	S 180	W 270

11. Compass Flux Valve (No.1 and No.2) — Adjustment / Compass Swing

NOTE: This adjustment compensates thin flux for effects of residual magnetism and is done in the following manner:

A. Place aircraft on a north (0°) heading. Allow sufficient time for compass indicator dial to settle down before and after adjustments. Correct for all error by means of "N-S" adjusting screw on flux valve.

B. Turn aircraft to east (90°) heading. Correct for all error by means of "E-W" adjusting screw.

C. Turn aircraft to south (180°) heading. Take out half the error , if any, with "N-S" adjusting screw.

D. Place aircraft on a west (270°) heading. Take out half the error, if any, with "E-W" adjusting screw.

E. Flux valve should now be fully compensated. As a check, swing aircraft on 30° increments and note readings on dial. All headings should be within 1° of magnetic heading. If errors noted are not tolerable, repeat compass swing procedure and readjust flux valve to a higher degree of accuracy.

F. Record readings as shown below:

	N 0	30	60	E 90	120	150	S 180	210	240	W 270	300	330
No.1												
No.2												

12. Standby Compass — Calibration

WARNING: ANY TIME MAINTENANCE CHECKS ARE TO BE PERFORMED IN CONNECTION WITH PITOT OR ANGLE OF ATTACK PROBE HEATERS, ENSURE SWITCHES ARE RETURNED TO OFF. IF THIS WARNING IS NOT ADHERED TO AND PERSONNEL COME INTO CONTACT WITH HEATER SURFACE, SERIOUS BURNS MAY RESULT.

CAUTION: ENSURE PITOT TUBE COVERS ARE REMOVED BEFORE APPLYING HEAT TO PROBES.

EXTENDED APPLICATION OF FULL VOLTAGE TO PITOT HEATERS CAN CAUSE PREMATURE FAILURE.

A. PROCEDURE

NOTE: Standby compass calibration to be accomplished with the APU running, radios, windshield heat, pitot heat and other normal flight required systems ON.

Pitot heat and windshield heat should only be on during time when actual compass readings are being taken.

(1) Swing aircraft to the four cardinal headings using pilots master compass as a standard.

(2) Adjust "E-W" and "N-S" adjustments on standby compass as required for cardinal headings.

(3) Swing aircraft to the four cardinal headings to verify standby compass headings are within a tolerance of ± 5°

(4) Swing aircraft to every 15° increment and record heading on compass calibration card.

13. Compass Flux Valve — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Gain access to flux valve through access cover 21-UPR-8 (Aircraft 1 - 93 not having ASC 125) or 21-UPR-11 (Aircraft having ASC 125 Aircraft 94 - 200, 322 and 323. (See Figure 202)

(3) Disconnect electrical leads.

(4) Remove mounting screws and unit.

B. Installation

(1) Install and secure flux valve.

(2) Connect electrical leads.

(3) Perform Compass Index Error — Adjustment, this Section.

(4) Perform Compass Flux Valve — Adjustment/Compass Swing, this Section.

(5) Inspect area for presents of foreign objects, security of all attachments.

(6) Install access panel.

14. Altimeter — Test

A. Reference: FAR Part 43, Appendix E, Sections 43.3 and 43.5; FAR Part 91, Section 91.411; FAR Part 145, Section 145.47; Advisory Circular 43-2 and U.S. Standard Atmosphere, 1962.

B. General: Altimeter and static system tests are required by Section 91.411 of FARS. Equipment, materials, and required tests for test equipment are specified in Section 145.47 of FARS. Persons authorized to perform altimeter and static systems tests are identified in Section 91.411 of FARS (Aircraft manufacturer or certified repair station).

C. Test Equipment: Following test equipment is acceptable for testing altimeters:

(1) Mercurial barometers with accuracies specified in and maintained in accordance with FAA Advisory Circular 43-2.

(2) High accuracy portable test equipment (with appropriate correction card) maintained in accordance with FAR 145.47(b). Calibration checks of test equipment in accordance with following schedule provides satisfactory level of performance:

(a) Each 30 days, after initial calibration, equipment will be checked for accuracy against barometer described in C.(1) or an altimeter (with appropriate correction card) which has been calibrated, within past 30 days, against barometer described in C.(1).

(b) Each day equipment will be checked for accuracy, at station pressure, using an aneroid or mercurial barometer. If equipment is not used daily, "before" test may be substituted for daily test.

D. Altimeter tests are performed in accordance with FAR Part 43, Appendix E. If altimeter test is to be performed with instrument installed in aircraft, following guidelines should be observed.

NOTE: Fixed frequency ac must be available to the altimeter, also all ADC function switches must be in the normal position.

(1) Static leak test should be conducted first to ensure that static system leaks do not influence altimeter indications. this Section.

(2) Permit altimeter to stabilize after a flight before performing test.

(3) Use portable test equipment or barometric test equipment as described in Step C.(1).

(4) When vibration is applied to instrument, ensure that it is not of a magnitude which might mask a sticky altimeter.

(5) Observing the precautions listed in FAA Advisory Circular No. AC 43-203B, test altimeter system as specified in FAR Part 43 Appendix E.

(6) Person making altimeter test, record on altimeter date of actual altimeter test and maximum, altitude to which altimeter was tested.

15. Altimeter — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Gain access to and disconnect and cap air lines to altimeter indicator.

(3) Remove mounting screws and remove indicator.

B. Installation

CAUTION: IF CALIBRATION DATE OF ALTIMETER BEING INSTALLED EXCEEDS 24 MONTHS, RECALIBRATE BEFORE INSTALLATION.

(1) Install and service indicator.

(2) Connect air lines to indicator.

(3) Perform Static System - Leak Check/Functional Test, this Section.

(4) Set indicator to correct barometric pressure.

(5) Record last calibration date of altimeter being installed.

16. ATC Transponder — Removal / Installation

NOTE: A ATC-600A Test Set and VPT-10 (or equivalent) Test Set will be required for these procedures;

A. Removal

(1) Remove power from aircraft.

(2) Gain access to transponder and disconnect antenna coax.

(3) Remove transponder.

B. Installation

(1) Install transponder and connect antenna coax.

(2) Ensure all appropriate circuit breakers are set.

(3) Apply electrical power to aircraft.

(4) On ATC control head, select ON, altitude reporting OFF and enter "1000" for ATC code.

CAUTION: PLACING TEST SET ANTENNA CLOSER THAN 15" TO AIRCRAFT ATC ANTENNA WILL RESULT IN DAMAGE TO TEST SET.

(5) Place ATC-600A test set antenna approximately 21" from aircraft ATC antenna.

(6) Set up ATC-600A tester for testing ATC systems.

NOTE: Do not use emergency codes 7700, 7770 and 7600 for normal or test operations.

(7) Place mode selector on ATC-600A to A/C code and check the following;

- Code 1000 is displayed in test set window.
- Binary code indicator lamp A1 comes on.
- F2 framing pulse lamp is off.

(8) Measure minimum trigger level as follow;

- Set XPDR SIG control fully counterclockwise.
- Set up test antenna until % reply indicator just begins to read 100% (see CAUTION above).
- Rotate XPDR SIG control clockwise until % reply indicator reads 90% reply.
- Note reading on XPDR SIG level scale. Reading should be between -69 dbm and -77 dbm.

NOTE: An additional ±3 dbm tolerance is allowed when using portable test equipment due to possible antenna coupling errors.

- Activate altitude reporting at the ATC control head and verify that the minimum trigger level observed in Step above differs no more than 1 dbm.

(9) Perform frequency test as follows;

NOTE: Pilots code 0000 is to be used for this Step only. After completing frequency test, reset pilots code to 1000.

- On ATC-600A, select FREQ/POWER switch to POWER. Observe indication on power meter, peak power should indicate 250 - 500 watts.
- Select FREQ/POWER switch to FREQ. Set pilots code 0000 into the transponder control head, deselect altitude reporting on control head.
- Adjust ATC-600A FREQ gain control for a mid-scale FREQ/PWR meter indication. Rotate the XMTR FREQ control for a peak FREQ/PWR meter indication. At the peak, read the deviation in MHz from 1090 MHz directly from the XMTR FREQ control dial. Transmitter frequency must be 1090 MHZ ±3 MHz.

(10) Perform SLS test as follows;

- Push SLS switch on ATC-600A to 0 bd. Numeric display blanks out, binary lamps go out, XPDR % reply meter drops to 0% (1% maximum) and the PWR/FREQ meter drops to 0.
- Push SLS switch to 9 db (ON). The % reply must be 90% minimum.

(11) Perform identification pulse test as follows;

- Press IDENT switch on ATC control head, identification pulse indicator on ATC-600A should light momentarily then go out.
- Press IDENT switch on ATC control column, identification pulse indicator on ATC-600A should light momentarily then go out.

(12) Perform correspondence test as follows;

- Connect pitot/static tester to aircraft.
- Select mode selector on ATC-600A to A/C ALT.
- Select altitude reporting on ATC control head.
- Set aircraft altimeter to 29.92.
- Operate altimeters to altitudes listed in Table below and verify cockpit displayed altitude differs no more than 125 feet of ATC transmitted altitude displayed on ATC-600A.

PULSE POSITION (0 to 1 in pulse position denotes absence or presence of pulse, respectively)											
Range Increments (Feet)	D_2	D_4	A_1	A_2	A_4	B_1	B_2	B_4	C_1	C_2	C_4
- 50 to + 50	0	0	0	0	0	0	1	1	0	1	0
950 to 1050	0	0	0	0	0	1	1	0	0	1	0
1050 to 1150	0	0	0	0	0	1	1	0	1	1	0
1250 to 1350	0	0	0	0	0	1	1	1	1	0	0
1750 to 1850	0	0	0	0	0	1	0	1	0	0	1
2550 to 2650	0	0	0	0	0	1	0	0	0	1	1
2750 to 2850	0	0	0	0	1	1	0	0	0	0	1
6750 to 6850	0	0	0	1	1	0	0	0	0	0	1
14,750 to 14,850	0	0	1	1	0	0	0	0	0	0	1
30,750 to 30,850	0	1	1	0	0	0	0	0	0	0	1

Data Correspondence Test Table

17. ATC Transponder — Test (FAR 91.413)

A. Remove power from aircraft.

B. Gain access to transponder and disconnect antenna coax.

C. Remove transponder, see ATC Transponder - Removal/Installation, this Section.

D. Perform test inspection as required by (FAR Part 43, Appendix F)

E. Install transponder, see ATC Transponder - Removal/Installation, this Section.

18. Static Selector Valve — Removal / Installation

A. Removal

(1) Remove selector valve mounting screws.

(2) Disconnect and cap air line to selector valve and remove valve.

B. Installation

(1) Install and secure selector valve.

(2) Connect air line to valve.

(3) Perform static portion of Pitot/Static - Leak Check/Functional Test, this Section.

19. Standby Compass Indicator — Removal / Installation

A. Removal

(1) Remove hardware securing compass housing.

(2) Disconnect electrical lead and remove housing containing compass.

(3) Remove hardware securing compass to housing and compass.

B. Installation

(1) Install aluminum foil on compass body to minimize magnetic standby compass defections caused by electrical field effects as follows:

(a) Prepare one piece of aluminum foil 3 1/2 inches wide by 18 inches long; fold lengthwise into a double strip 1 3/4 inches wide.

(b) Wrap double strip of foil tightly around compass body two and one-half times, positioning so that extra one-half turn is squarely on top of compass body and edge of strip is in contact with square mounting flange of compass; secure foil in position with scotch tape.

(c) Prepare another strip of foil 1 3/4 inches wide by 18 inches long; do not double this strip.

(d) Wrap single strip of foil directly over double strip, two and one-half times positioned so that extra one-half turn is at bottom of compass body; secure foil with scotch tape in at least two places.

(e) Wrap scotch tape completely around outside of foil and compass case; when completed, foil wrapping should be securely attached to compass body.

(f) Apply lacing cord ties over tape for added security.

(2) Check all compass and compass bracket attaching hardware for magnetic properties; it is important that all attaching hardware be of nonmagnetic materials.

(3) Install unit in housing and secure with nonmagnetic hardware.

(4) Connect electrical lead to fitting on compass.

(5) Reinstall compass and compass bracket using nonmagnetic hardware.

(6) Perform Standby Compass - Calibration, this Section.

(7) Check operation of compass edge light.

20. Rate of Climb Indicators (Pneumatic or Electrical) — Removal / Installation

A. Removal

(1) Disconnect air line and/or electrical connector from indicator.

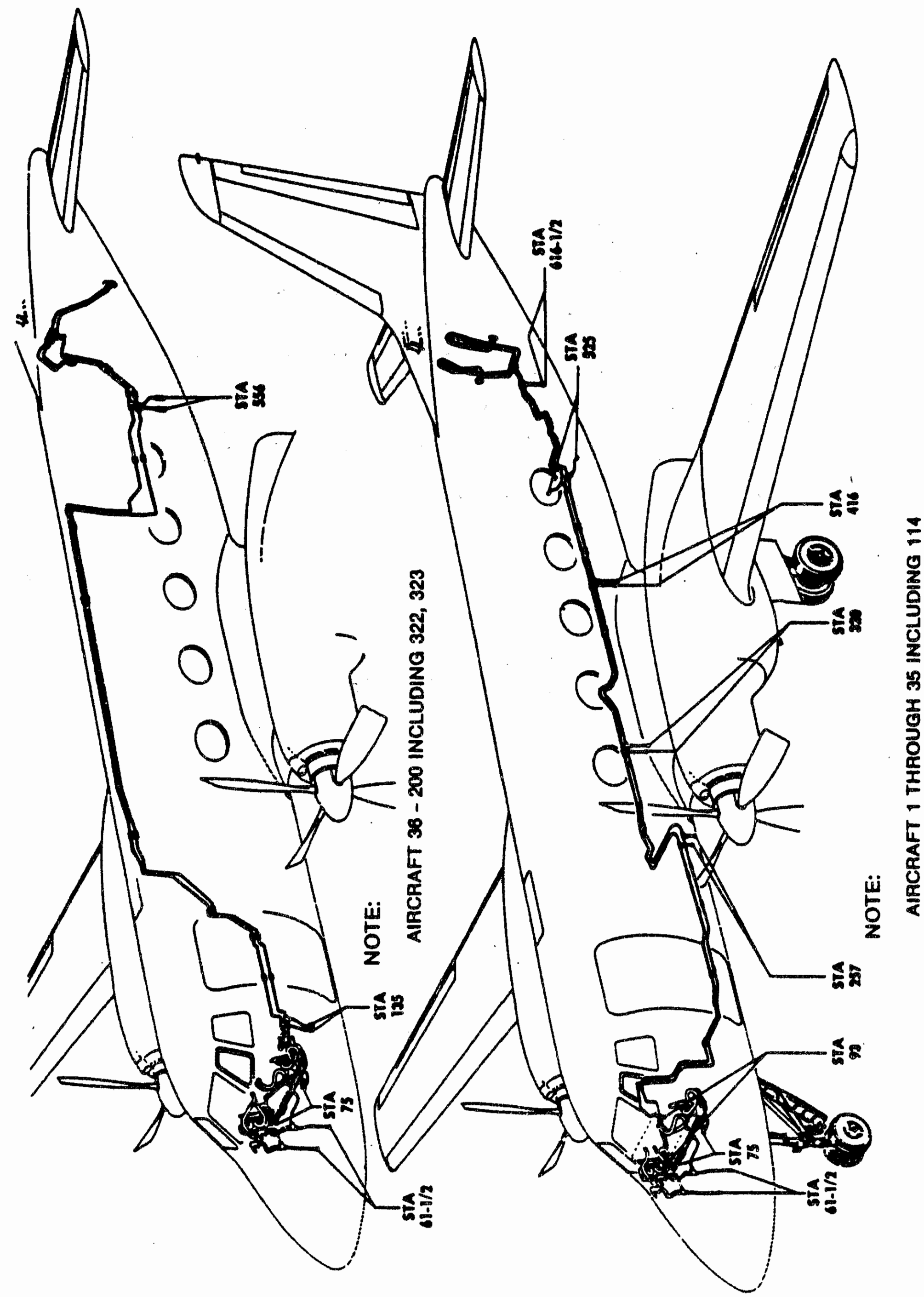
(2) Remove mounting screws and remove indicator.

B. Installation

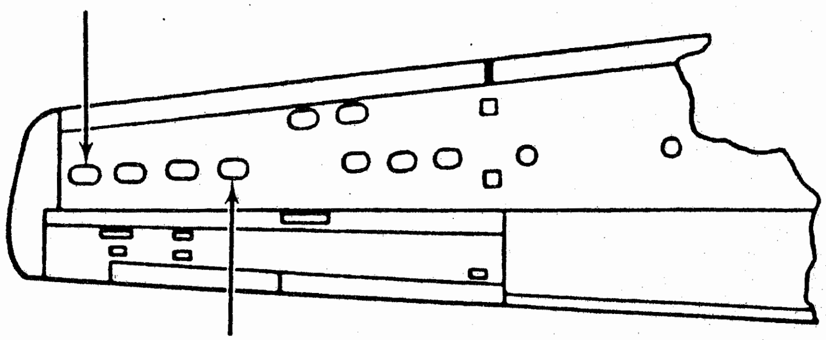
(1) Install and secure indicator with mounting screws.

(2) Connect air line and/or electrical connector to indicator.

(3) If pneumatic type, perform static portion of Pitot/Static - Leak Check/Functional Test, this Section.



Pitot/Static System Drain Points
Figure 201.



Flux Valve Location
Left Wing (View Looking Down) Right Wing (Opposite)
Figure 202.

5. **Airspeed Warning Sensor — Removal / Installation**

A. Removal

(1) Gain access to, disconnect and cap air lines and electrical plug.

(2) Remove mounting hardware and remove sensor.

B. Installation

(1) Ensure adequate insulation exists between skin of aircraft and sensor.

(2) Install and secure sensor.

(3) Connect air lines and electrical plug to sensor.

(4) Perform Overspeed Warning — Functional Test, this Section.

(5) Install access covers previously removed.

6. **Speed Warning "Cricket" Horn — Removal / Installation**

A. Removal

(1) Gain access to and disconnect electrical leads.

(2) Remove mounting hardware and horn.

B. Installation

(1) Install and secure warning horn.

(2) Connect electrical leads.

(3) Energize ESS dc bus and check "CRICKET" horn by pushing OVERSPEED TEST button.

7. **Pitot/Static System — Leak Check / Functional Test**

NOTE: Do not apply vacuum to pitot static lines when performing this test

A. Pitot System.

(1) Ensure that all equipment / units are installed properly and connections are tight.

(2) Connect suitable pressure source tester to pilot/copilot pitot head to produce indication of 290 to 296 knots on master airspeed indicator. Aircraft airspeed indicators should be within 5 knots of each other.

(3) Close pressure source.

(4) After one minute indicated air speed shall not decrease more than 5 knots.

(5) Slowly decrease pressure to zero and disconnect pressure source.

(6) If leak is detected isolate section of system until leak is found.

B. Static Systems.

NOTE: Do not apply pressure to either pitot or static line when performing this test.

(1) Ensure that all equipment / units are installed properly and connections are tight.

(2) Ensure that selector valve pertaining to system under test is in static pressure position (locked position).

(3) Seal port opening in flush static fitting on one side of aircraft only.

(4) Using test fitting or water drain connection, connect vacuum source tester to open port on opposite side of aircraft and to applicable pitot head.

(5) Interconnect both systems into a single vacuum pump.

CAUTION: WHEN APPLYING VACUUM TO SYSTEM, OBSERVE AIRCRAFT AIRSPEED INDICATOR. IF POINTER MOVES OFF ZERO POSITION, STOP TEST AND CHECK FOR LEAKS.

(6) Apply vacuum to both pitot and static system simultaneously until master altimeter indicates 15,500 feet. Difference between master altimeter and aircraft altimeter should not exceed 107 feet.

(7) Close vacuum source. After one minute indicated altitude on master altimeter shall not fall below 15,400 feet.

(8) Slowly reduce vacuum to zero at a rate of no more than 1,000 feet per minute.

(9) Disconnect vacuum source.

8. **Alternate Static System — Leak Check / Functional Test**

A. Move selector valve (No.1 or No.2) pertaining to system under test to alternate source position. Valve not being used must remain in locked position.

B. Using test fitting or water drain connection, connect vacuum source tester to alternate static source fitting in nose section of aircraft and to applicable pitot head.

C. Interconnect both systems to single vacuum pump.

CAUTION: WHEN APPLYING VACUUM TO SYSTEM, OBSERVE AIRCRAFT AIRSPEED INDICATOR. IF POINTER MOVES OFF ZERO POSITION, STOP TEST AND CHECK FOR LEAKS.

D. Apply vacuum to both pitot and static system simultaneously until master altimeter indicates 15,500 feet. Difference between master altimeter and aircraft altimeter should not exceed 107 feet.

E. Close vacuum source. After one minute indicated altitude on master altitude shall not fall below 15,400 feet.

F. Slowly reduce vacuum to zero at rate of no more than 1,000 feet per minute.

G. Disconnect vacuum source.

9. **Gyro Horizon Power Supply (No.1, No.2 and No.3) — Inspection**

CAUTION: REMOVE 10 AMP FUSE BEFORE REMOVING OR INSTALLING PS-823 COVERS.

NOTE: Field maintenance of the Model PS-823 power supply will be limited to fuse replacement, battery changing and performance test.

To perform inspection of the horizon system, consult J.E.T. Electronics Manual TP-200.

A. Remove battery packs and remove covers.

B. Inspect individual battery cells for indications of venting or leaking. Venting is indicated by a visible rupture near the top center of the cell or the appearance of corrosion in the same area.

C. Test battery by connecting a 10K 1/2 watt resistor and dc volt meter to PS.823 connector pins (+) and (−). The voltmeter shall read 20 - 28V dc.

D. Connect a 7 ohm, 150 watt resistor to battery pack terminals. When the terminal voltage decreases to 20V dc disconnect 7 ohm load.

E. Recharge battery pack by connecting a 200 milliampere current supply. Adjust supply to provide approximately 200ma and charge battery pack for 16 hours minimum.

10. **Ground Proximity Warning System — Operational Test**

A. Place battery switch to NORMAL and energize essential ac bus.

B. Turn essential and main radio master switch on.

C. Ensure GPWS ac and dc, No.1 NAV REC and RADIO ALT ac circuit breakers are depressed.

D. Ensure GPWS INHIBIT switch is in normal and safety wired.

E. Ensure GPWS FLAP OVERIDE switch is in normal with the guard closed.

F. Press GPWS test switch, both pilots and copilots pull-up annunciator lights should come on and aural warning pull-up, etc. should be heard.

G. Press radar altimeter test button, the radar altitude should indicate 50 feet after 2 - 3 seconds and then indicate 8888 after an additional 2 seconds. The decision height (DH) warning lights on both altimeter and pilots attitude indicator should come on.

H. Turn radio master switch off.

I. De-energize essential ac bus.

J. Turn battery switch off.

11. Compass Index Error — Adjustment

NOTE: This adjustment determines and cancels out index error due to longitudinal alignment of flux wave as follows.

A. Place compass system in operation and allow at least 5 minutes for gyro to come up to speed and to slave to magnetic north. Ensure that "N-S" and "E-W" adjustments on compensator on flux valve are at zero.

B. With engines running and as much equipment operating on aircraft as would be operating in flight, position aircraft on a compass rose and turn it to each of the four cardinal headings.

C. Record differences in reading between compass indicator dial and compass rose as + or −, depending on whether compass indicator readings are greater or less than compass rose readings. Allow sufficient time for compass indicator dial to settle out at each heading before taking readings.

D. Add the error algebraically and divide by four; the result is the index error.

E. Loosen screws holding flux valve flange to its mounting surface and rotate it to cancel out index error. If error is positive, flange should be rotated counterclockwise (giving a − reading on flange), as observed from above the unit. If error is negative, rotate flange clockwise (giving a + reading on the flange). Amount of rotation should equal index error.

F. Tighten mounting screws and check readings at four cardinal headings. Calculate index error to ensure it is zero. If it is not zero, readjust flux valve flange until this error is cancel led. Any remaining error in excess of 1° which are caused by extraneous magnetic fields should be counteracted by using adjustment screws mounted on top of flux valve.

G. Record readings as shown below:

No.1 COMPASS			
N 0	E 90	S 180	W 270

No.2 COMPASS			
N 0	E 90	S 180	W 270

12. Compass Flux Valve (No.1 and No.2) — Adjustment / Compass Swing

NOTE: This adjustment compensates thin flux for effects of residual magnetism and is done in the following manner:

A. Place aircraft on a north (0°) heading. Allow sufficient time for compass indicator dial to settle down before and after adjustments. Correct for all error by means of "N-S" adjusting screw on flux valve.

B. Turn aircraft to east (90°) heading. Correct for all error by means of "E-W" adjusting screw.

C. Turn aircraft to south (180°) heading. Take out half the error , if any, with "N-S" adjusting screw.

D. Place aircraft on a west (270°) heading. Take out half the error, if any, with "E-W" adjusting screw.

E. Flux valve should now be fully compensated. As a check, swing aircraft on 30° increments and note readings on dial. All headings should be within 1° of magnetic heading. If errors noted are not tolerable, repeat compass swing procedure and readjust flux valve to a higher degree of accuracy.

F. Record readings as shown below:

	N 0	30	60	E 90	120	150	S 180	210	240	W 270	300	330
No.1												
No.2												

13. Standby Compass — Calibration

WARNING: ANY TIME MAINTENANCE CHECKS ARE TO BE PERFORMED IN CONNECTION WITH PITOT OR ANGLE OF ATTACK PROBE HEATERS, ENSURE SWITCHES ARE RETURNED TO OFF. IF THIS WARNING IS NOT ADHERED TO AND PERSONNEL COME INTO CONTACT WITH HEATER SURFACE, SERIOUS BURNS MAY RESULT.

CAUTION: ENSURE PITOT TUBE COVERS ARE REMOVED BEFORE APPLYING HEAT TO PROBES.

EXTENDED APPLICATION OF FULL VOLTAGE TO PITOT HEATERS CAN CAUSE PREMATURE FAILURE.

A. PROCEDURE

NOTE: Standby compass calibration to be accomplished with the APU running, radios, windshield heat, pitot heat and other normal flight required systems ON.

Pitot heat and windshield heat should only be on during time when actual compass readings are being taken.

(1) Swing aircraft to the four cardinal headings using pilots master compass as a standard.

(2) Adjust "E-W" and "N-S" adjustments on standby compass as required for cardinal headings.

(3) Swing aircraft to the four cardinal headings to verify standby compass headings are within a tolerance of $\pm$ 5°

(4) Swing aircraft to every 15° increment and record heading on compass calibration card.

14. Compass Flux Valve — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Gain access to flux valve through access cover 21-UPR-8 (Aircraft 1 - 93 not having ASC 125) or 21-UPR-11 (Aircraft having ASC 125 Aircraft 94 - 200, 322 and 323.

(3) Disconnect electrical leads.

(4) Remove mounting screws and unit.

B. Installation

(1) Install and secure flux valve.

(2) Connect electrical leads.

(3) Perform Compass Index Error — Adjustment, this Section.

(4) Perform Compass Flux Valve — Adjustment / Compass Swing, this Section.

(5) Install access panel.

15. Altimeter — Test

A. Reference: FAR Part 43, Appendix E, Sections 43.3 and 43.5; FAR Part 91, Section 91.411; FAR Part 145, Section 145.47; Advisory Circular 43-2 and U.S. Standard Atmosphere, 1962.

B. General: Altimeter and static system tests are required by Section 91.411 of FARS. Equipment, materials, and required tests for test equipment are specified in Section 145.47 of FARS. Persons authorized to perform altimeter and static systems tests are identified in Section 91.411 of FARS (Aircraft manufacturer or certified repair station).

C. Test Equipment: Following test equipment is acceptable for testing altimeters:

(1) Mercurial barometers with accuracies specified in and maintained in accordance with FAA Advisory Circular 43-2.

(2) High accuracy portable test equipment (with appropriate correction card) maintained in accordance with FAR 145.47(b). Calibration checks of test equipment in accordance with following schedule provides satisfactory level of performance:

(a) Each 30 days, after initial calibration, equipment will be checked for accuracy against barometer described in C.(1) or an altimeter (with appropriate correction card) which has been calibrated, within past 30 days, against barometer described in C.(1).

(b) Each day equipment will be checked for accuracy, at station pressure, using an aneroid or mercurial barometer. If equipment is not used daily, "before" test may be substituted for daily test.

D. Altimeter tests are performed in accordance with FAR Part 43, Appendix E. If altimeter test is to be performed with instrument installed in aircraft, following guidelines should be observed.

NOTE: Fixed frequency ac must be available to the altimeter, also all ADC function switches must be in the normal position.

(1) Static leak test should be conducted first to ensure that static system leaks do not influence altimeter indications.

(2) Permit altimeter to stabilize after a flight before performing test.

(3) Use portable test equipment or barometric test equipment as described in Step C.(1).

(4) When vibration is applied to instrument, ensure that it is not of a magnitude which might mask a sticky altimeter.

(5) Observing the precautions listed in FAA Advisory Circular No. AC 43-203B, test altimeter system as specified in FAR Part 43 Appendix E.

(6) Person making altimeter test - record on altimeter date of actual altimeter test and maximum - altitude to which altimeter was tested.

16. Altimeter — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Gain access to and disconnect and cap air lines to indicator.

(3) Remove mounting screws and remove indicator.

B. Installation

CAUTION: IF CALIBRATION DATE OF ALTIMETER BEING INSTALLED EXCEEDS 24 MONTHS, RECALIBRATE BEFORE INSTALLATION.

(1) Install and service indicator.

(2) Connect air lines to indicator.

(3) Perform Static System — Leak Check / Functional Test, this section.

(4) Set indicator to correct barometric pressure.

(5) Record last calibration date of altimeter being installed.

17. ATC Transponder — Removal / Installation

NOTE: A ATC-600A Test Set and VPT-10 (or equivalent) Test Set will be required for these procedures;

A. Removal

(1) Remove power from aircraft.

(2) Gain access to transponder and disconnect antenna coax.

(3) Remove transponder.

B. Installation

(1) Install transponder and connect antenna coax.

(2) Ensure all appropriate circuit breakers are set.

(3) Apply electrical power to aircraft.

(4) On ATC control head, select ON, altitude reporting OFF and enter "1000' for ATC code.

CAUTION: PLACING TEST SET ANTENNA CLOSER THAN 15″ TO AIRCRAFT ATC ANTENNA WILL RESULT IN DAMAGE TO TEST SET.

(5) Place ATC-600A test set antenna approximately 21″ from aircraft ATC antenna.

(6) Set up ATC-600A tester for testing ATC systems.

NOTE: Do not use emergency codes 7700, 7770 and 7600 for normal or test operations.

(7) Place mode selector on ATC-600A to A/C code and check the following;

- Code 1000 is displayed in test set window.
- Binary code indicator lamp A1 comes on.
- F2 framing pulse lamp is off.

(8) Measure minimum trigger level as follow;

- Set XPDR SIG control fully counterclockwise.
- Set up test antenna until % reply indicator just begins to read 100% (see CAUTION above).
- Rotate XPDR SIG control clockwise until % reply indicator reads 90% reply.
- Note reading on XPDR SIG level scale. Reading should be between -69 dbm and -77 dbm.

  NOTE: An additional ± 3 dbm tolerance is allowed when using portable test equipment due to possible antenna coupling errors.

- Activate altitude reporting at the ATC control head and verify that the minimum trigger level observed in Step above differs no more than 1 dbm.

(9) Perform frequency test as follows;

NOTE: Pilots code 0000 is to be used for this Step only. After completing frequency test, reset pilots code to 1000.

- On ATC-600A, select FREQ/POWER switch to POWER. Observe indication on power meter, peak power should indicate 250 - 500 watts.
- Select FREQ/POWER switch to FREQ. Set pilots code 0000 into the transponder control head, deselect altitude reporting on control head.
- Adjust ATC-600A FREQ gain control for a mid-scale FREQ/PWR meter indication. Rotate the XMTR FREQ control for a peak FREQ/PWR meter indication. At the peak, read the deviation in MHz from 1090 MHz directly from the XMTR FREQ control dial. Transmitter frequency must be 1090 MHZ ± 3 MHz.

(10) Perform SLS test as follows;

- Push SLS switch on ATC-600A to 0 bd. Numeric display blanks out, binary lamps go out, XPDR % reply meter drops to 0% (1% maximum) and the PWR/FREQ meter drops to 0.
- Push SLS switch to 9 db (ON). The % reply must be 90% minimum.

(11) Perform identification pulse test as follows;

- Press IDENT switch on ATC control head, identification pulse indicator on ATC-600A should light momentarily then go out.
- Press IDENT switch on ATC control column, identification pulse indicator on ATC-600A should light momentarily then go out.

(12) Perform correspondence test as follows;

- Connect pitot/static tester to aircraft.
- Select mode selector on ATC-600A to A/C ALT.
- Select altitude reporting on ATC control head.
- Set aircraft altimeter to 29.92.
- Operate altimeters to altitudes listed in Table below and verify cockpit displayed altitude differs no more than 125 feet of ATC transmitted altitude displayed on ATC-600A.

PULSE POSITION (0 to 1 in pulse position denotes absence or presence of pulse, respectively)											
Range Increments (Feet)	D_2	D_4	A_1	A_2	A_4	B_1	B_2	B_4	C_1	C_2	C_4
- 50 to + 50	0	0	0	0	0	0	1	1	0	1	0
950 to 1050	0	0	0	0	0	1	1	0	0	1	0
1050 to 1150	0	0	0	0	0	1	1	0	1	1	0
1250 to 1350	0	0	0	0	0	1	1	1	1	0	0
1750 to 1850	0	0	0	0	0	1	0	1	0	0	1
2550 to 2650	0	0	0	0	0	1	0	0	0	1	1
2750 to 2850	0	0	0	0	1	1	0	0	0	0	1
6750 to 6850	0	0	0	1	1	0	0	0	0	0	1
14,750 to 14,850	0	0	1	1	0	0	0	0	0	0	1
30,750 to 30,850	0	1	1	0	0	0	0	0	0	0	1

Data Correspondence Test Table

18. ATC Transponder — Test

A. Remove power from aircraft.

B. Gain access to transponder and disconnect antenna coax.

C. Remove transponder.

D. Perform test inspection as required by (FAR 91.413)

E. Install transponder, see ATC Transponder — Removal / Installation this Section.

19. Static Selector Valve — Removal / Installation

A. Removal

(1) Remove selector valve mounting screws.

(2) Disconnect and cap air line to selector valve and remove valve.

B. Installation

(1) Install and secure selector valve.

(2) Connect air line to valve.

(3) Perform static portion of Pitot/Static — Leak Check / Functional Test, this Section.

20. Standby Compass Indicator — Removal / Installation

A. Removal

(1) Remove hardware securing compass housing.

(2) Disconnect electrical lead and remove housing containing compass.

(3) Remove hardware securing compass to housing and compass.

B. Installation

(1) Install aluminum foil on compass body to minimize magnetic standby compass defections caused by electrical field effects as follows:

(a) Prepare one piece of aluminum foil 3 1/2 inches wide by 18 inches long; fold lengthwise into a double strip 1 3/4 inches wide.

(b) Wrap double strip of foil tightly around compass body two and one-half times, positioning so that extra one-half turn is squarely on top of compass body and edge of strip is in contact with square mounting flange of compass; secure foil in position with scotch tape.

(c) Prepare another strip of foil 1 3/4 inches wide by 18 inches long; do not double this strip.

(d) Wrap single strip of foil directly over double strip, two and one-half times positioned so that extra one-half turn is at bottom of compass body; secure foil with scotch tape in at least two places.

(e) Wrap scotch tape completely around outside of foil and compass case; when completed, foil wrapping should be securely attached to compass body.

(f) Apply lacing cord ties over tape for added security.

(2) Check all compass and compass bracket attaching hardware for magnetic properties; it is important that all attaching hardware be of nonmagnetic materials.

(3) Install unit in housing and secure with nonmagnetic hardware.

(4) Connect electrical lead to fitting on compass.

(5) Reinstall compass and compass bracket using nonmagnetic hardware.

(6) Perform Standby Compass — Calibration, this Section.

(7) Check operation of compass edge light.

21. Rate of Climb Indicators (Pneumatic or Electrical) — Removal / Installation

A. Removal

(1) Disconnect air line and/or electrical connector from indicator.

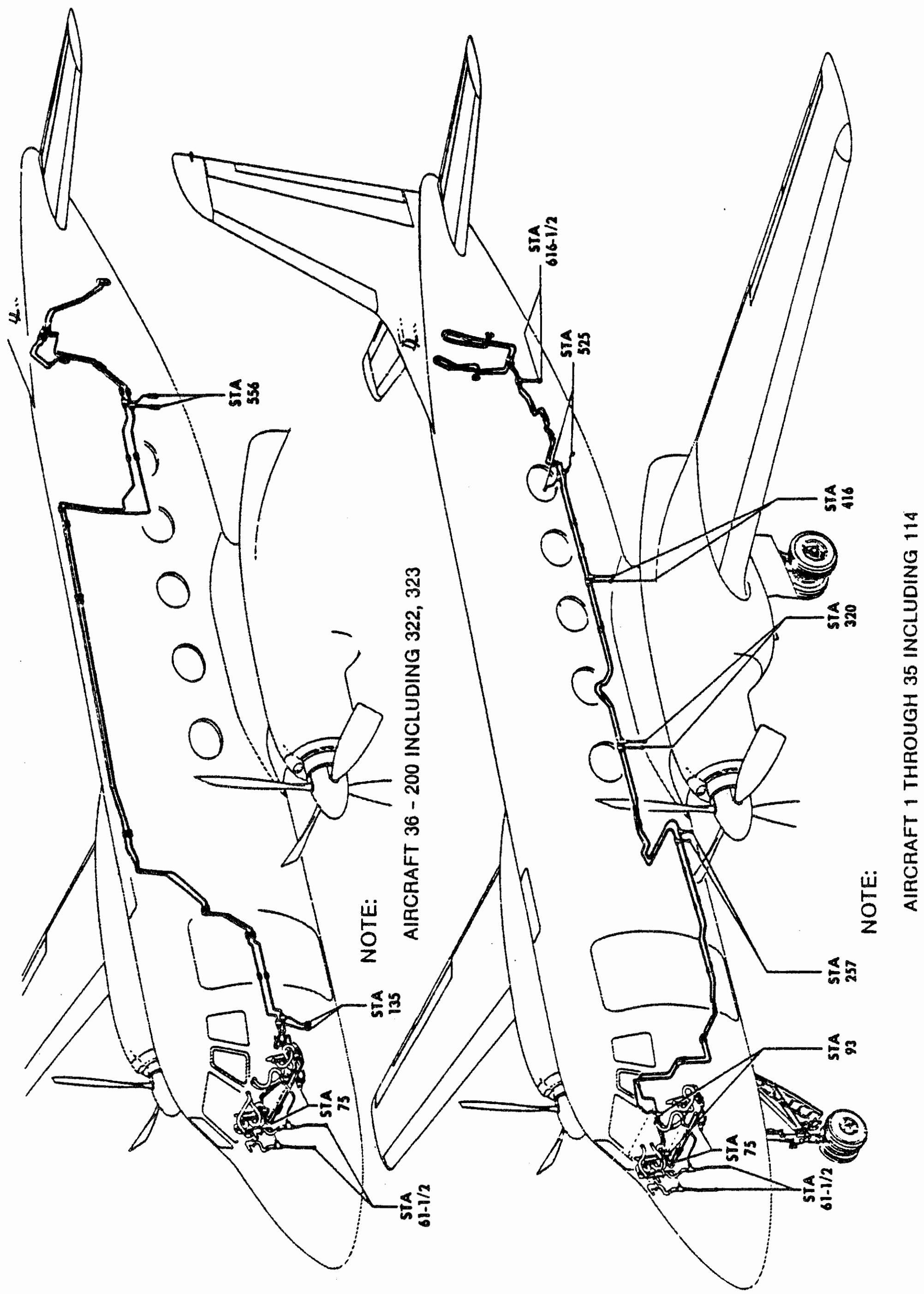
(2) Remove mounting screws and remove indicator.

B. Installation

(1) Install and secure indicator with mounting screws.

(2) Connect air line and/or electrical connector to indicator.

(3) If pneumatic type, perform static portion of Pitot/Static — Leak Check / Functional Test, this Section.



Pitot/Static System Drain Points
Figure 201.

LIST OF EFFECTIVE PAGES

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OXYGEN

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CHAPTER 35

OXYGEN

OXYGEN SYSTEM — DESCRIPTION / OPERATION

1. General

A. Aircraft 1 - 59 and 114 (See Figure 1)

The aircraft utilizes an 1800 psi high pressure oxygen system which provides pilot and copilot with a diluter-demand type of oxygen supply. Provisions have also been made, through a shutoff valve, for future incorporation of a low-pressure (17 to 42 psi), continuous flow, supply system for the passenger compartment. Installation of an automatic, continuous flow, regulator and wall outlets located as required by compartment arrangement, is required to complete the passenger compartment oxygen system. (This is a function of the furnishing agency.) Usual installation consists of a plug-in type of outlet as each seat with disposable masks. One or two outlets, (including a therapeutic outlet), are in the lavatory. Therapeutic outlet flow is rated at several times the normal flow and is utilized in case passenger illness or respiratory problems arise. Unless the cabin installation includes special types of outlets and masks with built-in demand type regulators, standard plug-in masks used will cause oxygen to flow only when plugged in. This flow is at a continuous rate, regardless of whether the mask is on or not provided the supply and cabin shutoff valves in the cockpit are both on. Removing the mask plug will shutoff the flow at that outlet. Major components of the oxygen system installed in the basic production aircraft are listed in the following table.

Unit	No. Per A/C	Location
Filler Valve	1	Nosewheel well, left side, forward of Fuselage Station 63.
Oxygen Cylinder	1	Forward of cockpit bulkhead Fuselage Station 63.
Diluter-Demand Regulator	2	One each in the aft section of pilots and copilots side consoles at Fuselage Station 133.
Pressure Gage	1	Pilots outboard skirt panel.
Flow Indicator	2	One on the pilots outboard skirt panel adjacent to the pressure gage, and one on the copilots outboard skirt panel adjacent to the normal hydraulic pressure gage.
Cabin Shutoff Valve	1	Aft section of copilots side console at Fuselage Station 133, adjacent to the diluter-demand regulator.
Oxygen Supply Shutoff Valve	1	Lower left skirt of pilots panel.

The oxygen supply from the oxygen cylinder is routed with a common rigid tubing outlet line, through a cockpit oxygen shutoff valve to each diluter-demand regulator. The oxygen supply shutoff valve is provided in the system to meet FAA requirements. The oxygen supply shutoff valve acts as an internal master shutoff for the entire aircraft oxygen supply (both passenger and crew).

NOTE: The cabin shutoff valve will stop all oxygen flow to the passengers when closed, but will not affect the crew supply.

Oxygen supply shutoff valve should remain closed at all times until oxygen is required in flight. Flexible hoses, attached to each regulator, provide the oxygen mask connections for each flight crew member. A 0 to 2000 psi oxygen cylinder pressure gage located on the pilots outboard skirt panel indicates the available oxygen supply in the system. Oxygen is routed from the gage to the crew distribution system and also to the passenger distribution system.

NOTE: Do not attempt to read cylinder pressure gage until after the oxygen supply shutoff valve is open, since the gage is downstream of the valve. Opening valve slowly will bring cylinder pressure into the gage and the rest of the system, this is the only way to check cylinder pressure. (See Figure 1)

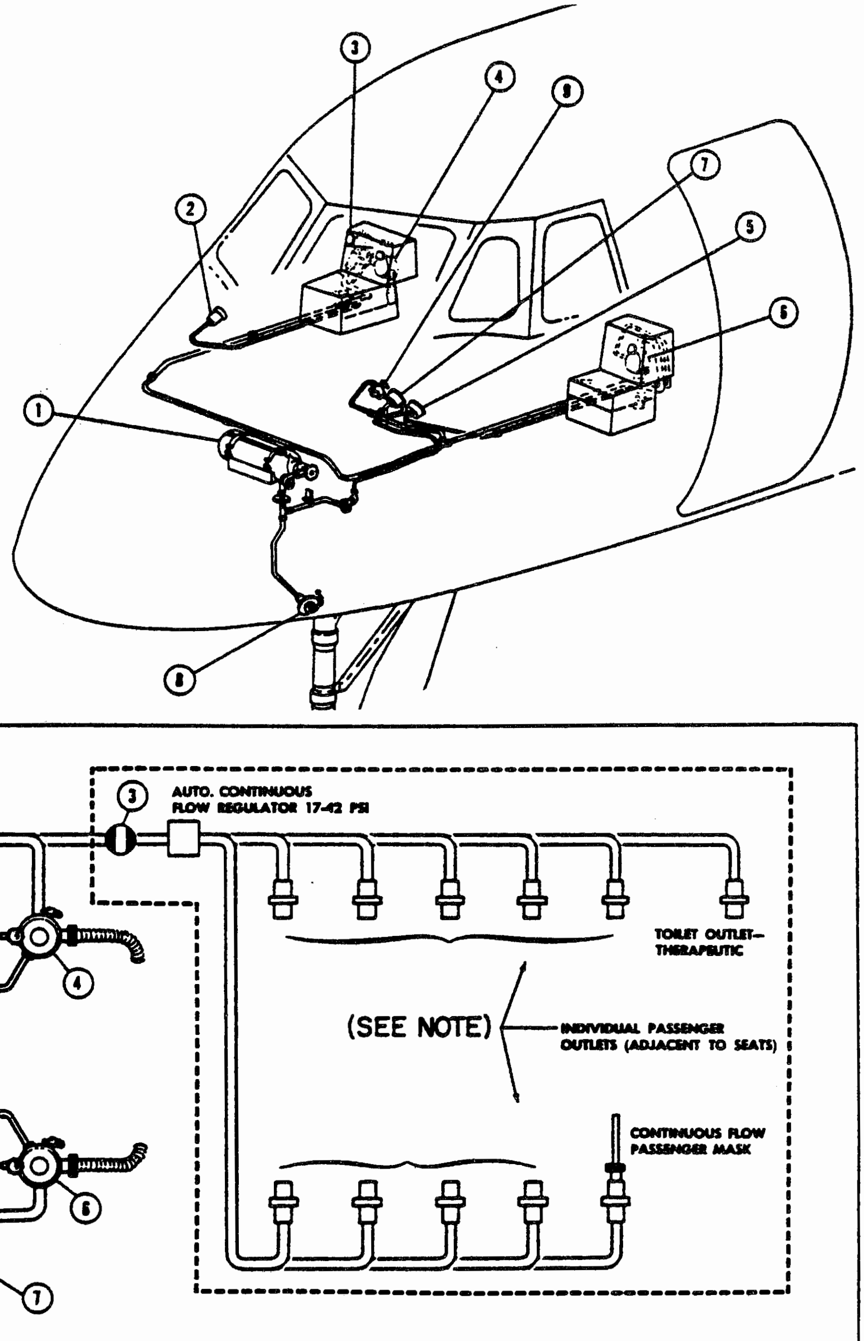
A blinker type flow indicator is connected to each diluter-demand regulator to provide a visual indication of the proper operation of the regulator.A T-fitting connects one outlet of the oxygen supply shutoff valve to the diluter-demand regulator on the copilots console and the cabin shutoff valve. The cabin oxygen shutoff valve outlet is capped on delivery. The passenger oxygen system, when installed, uses this outlet as its oxygen supply source. It serves no other purpose in the production installation.

A. Aircraft 60 - 200, 322 and 323, excluding 114, (See Figure 2)

The oxygen system installed in Aircraft 60 - 200, 322 and 323 excluding 114, the same as the system installed in earlier production models, except for the omission of the cabin shutoff valve aft of the copilots side console at Fuselage Station 133. The oxygen flow is directly from the oxygen supply, through a cockpit oxygen shutoff valve to the diluter-demand regulators, and capped at the copilots regulator T-fitting. This enables the user to have the furnishing agency install any approved type of shutoff valve, automatic or manual as desired.

B. Aircraft having ASC 200A have an access door on the lower left nose section for the filler valve. The filler valve is enclosed in the wheel well to protect the filler valve from contamination. Refer to Chapter 12 for service instructions.

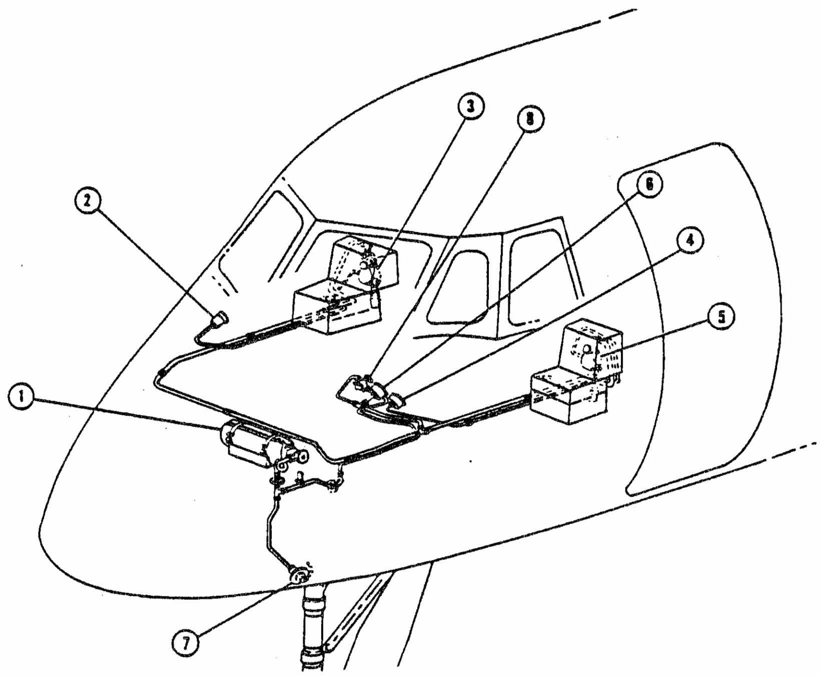
1. Oxygen Bottle
2. Copilots Flow Indicator
3. Cabin Shutoff Valve
4. Copilots Demand Regulator
5. Pilots Flow Indicator
6. Pilots Demand Regulator
7. Pressure Gage
8. Filler Valve
9. Cockpit Shutoff Valve

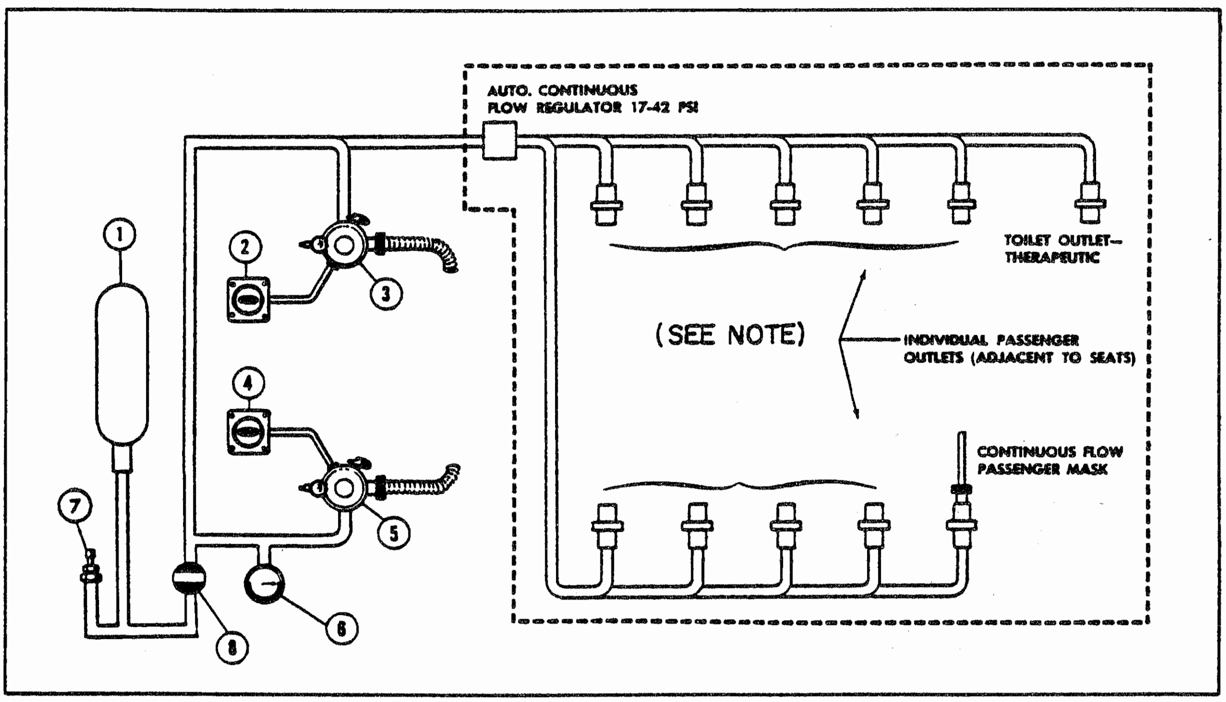


NOTE: Installation shown within dotted lines is typical only and is not provided by manufacturer. This equipment will be installed by distributor at purchaser's request to meet individual requirements.

Oxygen System — Aircraft 1 - 59 and 114
Figure 1.

1. Oxygen Bottle
2. Copilots Flow Indicator
3. Copilots Demand Regulator
4. Pilots Flow Regulator
5. Pilots Demand Regulator
6. Pressure Gage
7. Filler Valve
8. Cockpit Shutoff Valve





NOTE: Installation shown within dotted lines is typical only and is not provided by manufacturer. This equipment will be installed by distributor at purchaser's request to meet individual requirements.

Oxygen System — Aircraft 60 - 200, 322 and 323 Excluding 114
Figure 2.

OXYGEN SYSTEM — FAULT ISOLATION

Fault	Possible Cause	Correction
Loss of system pressure.	Leaking or damaged cylinder.	Replace cylinder.
	Shutoff valve(s) closed. (Aircraft 1 - 59 Including 114 only.)	Open valve(s).
	System not charged.	Replenish oxygen supply.
	Defective pressure gage.	Replace gage.
	Defective diluter-demand regulator.	Replace regulator.
	Leak in system.	Check all valves, tubing and connections for leaks with soap solution.
Insufficient oxygen at high altitude.	Restricted lines.	Examine tubing for dents, clogging, etc. Replace if necessary.
	Mask does not fit properly.	Refit or replace mask.
	Defective diluter-demand regulator.	Replace regulator.
	Breathing tube damaged.	Replace tube.
	Defective oxygen cylinder.	Replace cylinder.
	Leak in system.	Check all valves, tubing and connections for leaks with soap solution.
No oxygen available at regulator.	Shutoff valve on cylinder is OFF.	Open valve by turning counterclockwise. (Accessible only on ground through access opening LH side.)
	Oxygen supply shutoff valve in cockpit is OFF.	Open valve. (Pilot outboard skirt panel.)

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OXYGEN SYSTEM — MAINTENANCE PRACTICES

1. General

Hands, clothing and tools must be free of oil, grease and dirt when working with oxygen equipment. Only special, nonsparking tools made of brass or berilium should be used.

WARNING: TRACES OF OIL, GREASE OR ORGANIC MATTER NEAR COMPRESSED OXYGEN MAY RESULT IN SPONTANEOUS COMBUSTION AND AN EXPLOSION.

Never use a mixture containing oil, grease or any other hydrocarbon on any connection, packing, gage or other part of oxygen equipment. Apply Recto-Seal No. 15 to all pipe thread connections: apply the compound sparingly and carefully only to the first three male threads from the end of the fitting. No compound or other lubricant shall be used on coupling sleeves or outside of a tubing flare.

Ensure all parts of the gages, tubing, valves and etc. are free from oil, grease or other foreign matter.

When lines are disconnected, close them immediately. Use clean, lint free caps to exclude air and foreign matter. Masking tape or similar items are not suitable.

CAUTION: NEVER FILL, DRAIN OR OTHERWISE DISCHARGE OXYGEN FROM AN OXYGEN SYSTEM WHEN THE AIRCRAFT IS IN A HANGAR.

DO NOT, UNDER ANY CIRCUMSTANCES, BLOW AIR THROUGH OR INTO OXYGEN EQUIPMENT.

High pressure oxygen cylinders may be replaced when the aircraft is inside a hangar, but should never be connected to the aircraft system or turned on in a hangar.

Type 321 corrosion resistant steel, 52SO aluminum alloy, and type N copper lines in the oxygen pressure system should be replaced at any time that they are found to be kinked, twisted, deeply scored or nicked, Care should be used when replacing cylinders to prevent unnecessary bending of the cylinder supply copper tubing.

Use only Aviators Gaseous Breathing Oxygen, from green-colored cylinders.

CAUTION: NEVER TIGHTEN OXYGEN SYSTEM FITTING WITH OXYGEN SYSTEM PRESSURE APPLIED.

2. Oxygen Cylinder — Maintenance Precautions

Handle cylinders carefully at all times. Never use cylinders as supports. Store them where they will not be knocked over or struck by other objects.

Spare cylinders should be stored in a cool place out of direct sunlight. Cylinders containing oxygen should be stored well away from other gas cylinders, highly inflammable materials, radiators or any source of heat.

When cylinders aboard aircraft are replaced, dust caps must be installed on the removed cylinder. Valves on discharged cylinders should be kept tightly closed, even if the cylinders are considered to be empty. Any remaining oxygen should be kept in the cylinder to prevent moisture laden air from entering. Should it ever be necessary to empty a cylinder, this should be done in the open atmosphere (outdoors).

Whenever possible, maintain at least 100 psi oxygen pressure in cylinders at all times to prevent accumulation of moisture due to condensation. When the aircraft ascends from warm temperatures at ground level to cold temperatures at high altitudes, the gage will show a drop in cylinder pressure. This does not indicate a loss of oxygen, but only a decrease in pressure caused by the drop in temperature.

NOTE: Particular attention of both crew and maintenance personnel should be directed to the fact that if the cylinder pressure falls below 50 psi the cylinder must be replaced and appropriate test made before it is put back in service.

A small indentation or an extensive rust spot on the thin wall of a cylinder may materially reduce its strength. Any cylinder so marred should be removed from service for an inspection by the oxygen supplier.

Do not clamp cylinder in a vise. The cylinder must not be scratched or marred.

WARNING: OPERATE ALL OXYGEN VALVES CAREFULLY USING CAUTION DURING OPENING AND CLOSING. VALVES SHOULD BE OPENED AND CLOSED SLOWLY. FAILURE TO OBSERVE THESE SAFETY PRECAUTIONS COULD RESULT IN A FLASH FIRE OR EXPLOSION.

Operate all oxygen valves carefully, opening and closing them slowly. The integral oxygen cylinder shutoff valve and its wheel type handles can be damaged or loosened very easily. For this reason, cylinders must be stored properly and handled carefully. The cylinder should not be lifted or carried by the valve handle.

Never tamper with or attempt to repair oxygen cylinder valves.
Minimum service pressure for main cylinder is 1750 psi at 20.1°C (70°F).
Maximum service pressure for cylinder is 1850 psi at 21.1°C (70°F).
Minimum service pressure is 1710 psi, and maximum service pressure is 1760 psi at 10°C (50°F).
Minimum service pressure is 1925 psi, and maximum service pressure is 1950 psi at 37.8°C (100°F).

3. **Oxygen System — Operational Test**

(See Figure 201)

WARNING: OIL, GREASE OR ORGANIC MATTER NEAR COMPRESSED OXYGEN MAY RESULT IN SPONTANEOUS COMBUSTION AND/OR EXPLOSION. USE STANDARD MAINTENANCE PRACTICES FOR AVIATION OXYGEN SYSTEMS.

A. Ensure pilot and copilot oxygen regulator emergency valves are closed and passenger system shutoff valve is closed.

CAUTION: DO NOT LET THE OXYGEN CYLINDER PRESSURE DROP BELOW 50 PSI. IF IT DOES, DO NOT REPLENISH CYLINDER AS IT MUST BE REMOVED AND PURGED BY AN APPROVED SERVICING AGENCY BEFORE REPLENISHING.

NOTE: On Aircraft having ASC 80 and Aircraft 60 - 200, 322 and 323 excluding 114 and the passenger system shutoff valve has been removed.

B. Open oxygen supply shutoff valve slowly (pilot outboard skirt panel) to pressurize system; a fully charged cylinder will read 1750 to 1850 psi at 21.1°C (70°F).

C. Test regulator for leakage by obstructing outlet of oxygen breathing tube connected to pilots diluter-demand regulator (if flow indicator blinker closes in less than 30 seconds, there is excessive leakage); repeat procedure to detect leakage in copilots regulator.

NOTE: Thumb must be removed from disconnect after each continuous inhalation. If mask leaks or does not fit satisfactorily, tighten mask straps.

D. Check oxygen masks by placing thumb over connector at end of mask tube and inhaling very lightly (if there is no leakage, mask will adhere tightly to face during inhalation and definite resistance to inhalation will be encountered).

NOTE: Frequently check pressure gage to determine state of available oxygen supply.

E. With oxygen mask connected to regulator and system pressure on, recheck oxygen mask fit using oxygen; put on oxygen mask; inhale deeply and hold breath (blinker should open and close). Note position of flow indicator blinker. If flow indicator reopens, leak is indicated. Tighten oxygen mask straps until flow indicator closes.

F. Breathe normally, observing oxygen flow indicator for blink which will verify positive flow of oxygen.

G. Close oxygen system shutoff valve (pilot outboard skirt panel).

NOTE: Clean and sanitize the oxygen masks with a non-oil base cleaner approved for use on oxygen masks.

If oxygen cylinder needs replenishing after check, top off cylinder using normal replenishing procedure.

4. **Passenger Oxygen System — Operational Test**

CAUTION: USE ALL NORMAL PRECAUTIONS INVOLVED WITH OXYGEN DURING THIS TEST.

A. Activate source of oxygen.

B. Turn on passenger oxygen supply at regulator/shutoff valve.

C. Check for proper operation of warning devices if installed.

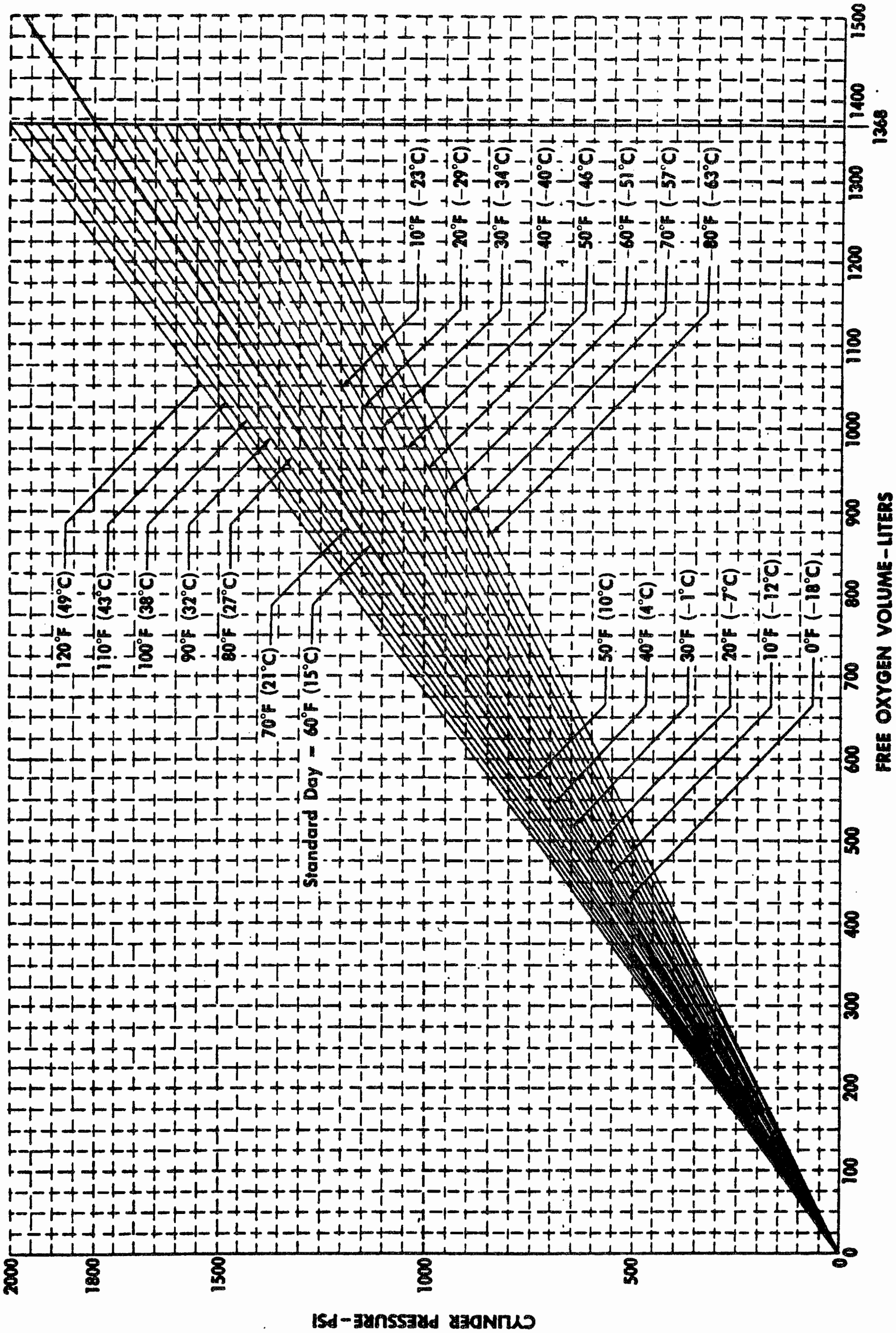
D. Ensure that oxygen is flowing at each passenger mask outlet including lavatory outlets and check masks for condition.

E. Inspect oxygen outlets for condition, security and proper operation.

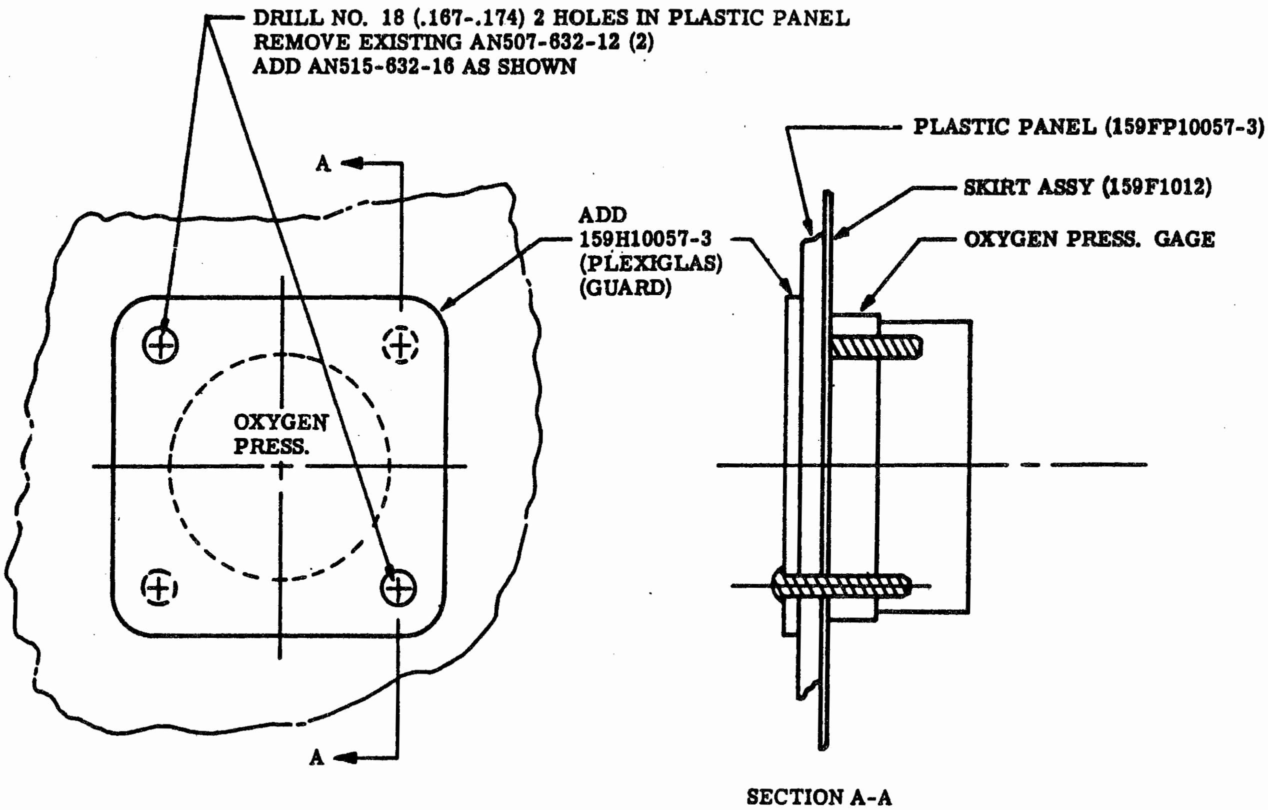
F. Turn off passenger oxygen supply and return system to normal configuration.

G. Clean and sanitize the oxygen masks with a non-oil base cleaner approved for use on oxygen masks.

H. Stow all masks properly.



Temperature, Pressure and Liters Relationship
Figure 201.



Oxygen Pressure Indication
Figure 202.

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OXYGEN CYLINDER — DESCRIPTION / OPERATION

1. Description

One 48.3 cubic foot capacity, high-pressure oxygen cylinder is installed in a mounting cradle just forward of the Fuselage Station 63 bulkhead on the pilots side. The cylinder rests horizontally on a cradle and is held firmly in place with band type clamps. It is accessible through an access hole in the left side of the nose skin. Cylinder should be removed only in event that it is damaged, below 50 psi, or at intervals specified in inspection schedule. Cylinder is equipped with a valve which is opened manually after system supply line is connected to it. Because the oxygen cylinder and associated shutoff valve is located out of the pressurized area, and is accessible only from the outside of the aircraft through an elliptical access plate, left side of the aircraft. Normal replenishing of the oxygen is accomplished through an external filler valve located in the nosewheel well. This valve enables maintenance personnel to fill cylinder without resorting to removing access plate and the cylinder from its rack. (See 35-0 Figure 1)

WARNING: USE STANDARD MAINTENANCE PRACTICES FOR AVIATION OXYGEN SYSTEM. PRECAUTIONS AGAINST OIL, GREASE AND ORGANIC MATTER SHOULD BE OBSERVED.

Cylinder recharging using external filler procedure is outlined in Chapter 12.

WARNING: MINIMUM OXYGEN ABOARD FOR TAKEOFF SHOULD BE 1000 PSI.

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OXYGEN CYLINDER — MAINTENANCE PRACTICES

1. **Oxygen Cylinder — Removal / Installation**

WARNING: OIL, GREASE OR ORGANIC MATTER NEAR COMPRESSED OXYGEN MAY RESULT IN SPONTANEOUS COMBUSTION AND/OR AN EXPLOSION. USE STANDARD MAINTENANCE PRACTICES FOR AVIATION OXYGEN.

RADAR MUST NOT BE IN OPERATION DURING REMOVAL/INSTALLATION.

A. Removal

(1) Ensure oxygen supply shutoff valve (pilots outboard skirt panel) is OFF.

CAUTION: IF THIS METHOD IS USED, TAKE SPECIAL PRECAUTION TO PREVENT DAMAGE TO RADAR COMPONENTS IN NOSE DURING REMOVAL AND REPLACEMENT OF CYLINDER.

NOTE: If radome is removed and radar antenna absorption blanket is unfastened from left side, oxygen cylinder may be removed and replaced without removing outside and nosewheel well opening access plates.

(2) Gain access to oxygen cylinder in nose by removing circular access plate on left side of aircraft forward of Fuselage Station 63. For remote cylinder installation, gain access as required.

WARNING: DO NOT DISCHARGE OXYGEN INTO THE COCKPIT COMPARTMENT, WHEEL WELL OR ANY CONFINED AREA.

(3) Before disconnecting tubing, bottle pressure should be reduced to 150 ± 50 psi.

(4) Fully close oxygen shutoff valve at cylinder if installed.

CAUTION: ALWAYS DISCONNECT TUBING AT CYLINDER END FIRST TO PREVENT COMPLETE DRAINING OF CYLINDER.

(5) Disconnect tubing at cylinder end.

(6) Remove tubing and cap both ends.

(7) Cap T-fitting

(8) Loosen wing nuts on cylinder hold down straps.

(9) In nosewheel well, remove elliptical plate (under oxygen cylinder shelf) to gain access to inboard strap wing nut.

CAUTION: DO NOT REMOVE BONDING JUMPER.

NOTE: To remove access cover, open nose gear clamshell doors. Its exact location is between nose gear uplock J hook structure and nosewheel well door cylinder timer valve. Access cover is secured by 11 screws and a bonding jumper.

(10) Remove oxygen cylinder.

B. Installation

(1) Ensure cylinder has current hydrostatic test and record date on attached card.

(2) Position cylinder so that valve is in same alignment as when removed.

(3) Fasten cylinder hold down straps, tighten wing nuts and safety.

WARNING: SOME OXYGEN WILL BE LOST WHEN TIGHTENING CYLINDER END OF TUBING, HANDS AND TOOLS MUST BE GREASE FREE.

(4) Apply thread sealer to fitting threads and connect coiled tubing to cylinder valve outlet and T-fitting.

(5) Tighten T-fitting end first.

(6) If installed slowly open cylinder shutoff valve.

(7) Check fitting at cylinder end and T-fitting for leaks by applying Leak-Tec Formula OX16 to fittings. Bubbles indicate leakage.

(8) Wipe off residue of Leak-Tec solution with clean cloth.

(9) Close all openings, access plates, install radar absorption blanket and close radome as required.

(10) Place cockpit oxygen supply shutoff valve to open.

(11) Observe oxygen pressure gage for full cylinder indication.

(12) Close cockpit oxygen supply shutoff valve.

2. Oxygen Cylinder — Hydrostatic Test

WARNING: THE HYDROSTATIC TEST MUST BE ACCOMPLISHED BY EXPERIENCED PERSONNEL IN A FACILITY EQUIPPED TO PREFORM THIS TEST.

A. Remove oxygen cylinder.

B. Perform hydrostatic test.

C. Install oxygen cylinder.

D. Place cockpit oxygen supply shutoff valve to open.

E. Observe oxygen pressure gage for full cylinder indication.

F. Close cockpit oxygen supply shutoff valve.

3. Oxygen Filler Valve — Removal / Installation

WARNING: OIL, GREASE OR ORGANIC MATTER NEAR COMPRESSED OXYGEN MAY RESULT IN SPONTANEOUS COMBUSTION AND/OR EXPLOSION. USE STANDARD MAINTENANCE PRACTICES FOR AVIATION OXYGEN SYSTEMS.

A. Removal

(1) Gain access to oxygen cylinder by removing circular access plate on left side of aircraft at Fuselage Station 63.

(2) Close shutoff valve on oxygen cylinder.

(3) Ensure oxygen supply shutoff valve (pilots outboard skirt panel in cockpit) is off.

NOTE: In the following step a small amount of oxygen will escape as the fitting is loosened, therefore loosen fitting slowly until all pressure in the line is dissipated.

On aircraft having ASC 200A the oxygen filler valve has been relocated to the left side of the aircraft between Fuselage Station 50.6 and Fuselage Station 56.8 and approximately WL 74.3. Access to the filler valve is provided through an outside door panel.

(4) Disconnect line from filler valve in nosewheel well. Install protective caps on fittings.

(5) Remove filler valve assembly, retain mounting hardware.

B. Installation

(1) Install filler valve using previously removed hardware.

(2) Remove protective caps and connect line.

(3) Open oxygen cylinder valve slowly to prevent surging.

NOTE: In the following step a slight allowable leakage will be obtained from the filler valve assembly, therefore, the check should be made with the filler valve safety cap on and tight. If leakage appears around cap, check cap for tightness. If tight, another filler valve assembly should be installed.

(4) Check filler valve connection for leakage using Leak-Tec Solution OX16. Absence of bubbles indicates that connection is satisfactory. Wipe off solution when leak check is completed.

(5) Ensure oxygen cylinder valve is fully open, then close oxygen cylinder access plate.

(6) In cockpit, turn on oxygen supply shutoff valve (pilots outboard skirt panel). Close cockpit valve when completed.

4. Cockpit Oxygen Shutoff Valve — Removal / Installation

A. Removal

(1) Gain access to oxygen cylinder by removing the circular access plate on left side of aircraft at Fuselage Station 63.

(2) Close shutoff valve on oxygen cylinder.

(3) Open oxygen supply shutoff valve (pilots outboard skirt panel) and bleed off any accumulated line pressure downstream of cylinder by momentarily placing emergency valve on either regulator to open position.

(4) Close regulator emergency valve.

(5) Disconnect oxygen lines from shutoff valve. Install protective caps on lines and fittings.

(6) Remove shutoff valve. Retain mounting hardware.

B. Installation

CAUTION: USE CARE WHEN INSTALLING FITTINGS ON OXYGEN REGULATOR AS TEFLON TAPE REDUCES NORMAL TORQUE FEEL DUE TO ITS LOW FRICTION CHARACTERISTICS. DO NOT OVERTORQUE.

NOTE: If a replacement shutoff valve is to be installed remove two fittings from old shutoff valve and transfer to same ports on replacement valve. Use Teflon tape on tapered pipe END of fitting only. Wrap tape two complete turns starting from tapered end. Wrap in direction fitting is to be tightened.

(1) Remove protective caps from valve and lines.

(2) Install valve in skirt panel using hardware removed previously. Ensure valve is closed.

(3) Connect lines.

(4) Open oxygen cylinder valve slowly to prevent surging.

(5) Test input fitting of valve for leakage using Leak-Tec Formula. If satisfactory, open valve and check output fitting for leakage. Absence of bubbles indicates satisfactory connections have been made. Wipe off solution after check.

(6) Close cockpit valve.

(7) Ensure oxygen cylinder valve is fully open; then close oxygen cylinder access plate.

5. Cockpit / Nose Wheel Well Oxygen Pressure Gage — Removal / Installation

A. Removal

(1) Close oxygen supply shutoff valve.

(2) Deplete oxygen pressure in line by momentarily opening emergency valve on regulator, disconnect line from gage.

(3) If gage face is covered with plastic shield remove shield by removing screws, retain screws.

(4) Remove screws securing gage and remove gage. Retain screws and install protective cap on line.

B. Installation

CAUTION: USE CARE WHEN INSTALLING FITTING ON OXYGEN GAGE AS TEFLON TAPE REDUCES NORMAL TORQUE FEEL DUE TO ITS LOW FRICTION CHARACTERISTICS. DO NOT OVERTORQUE.

NOTE: If a replacement gage is to be installed, transfer fitting from old gage to new gage. Use Teflon tape on tapered pipe end of fitting only. Wrap tape two complete turns starting from tapered end. Wrap in direction fitting is to be tightened

(1) Remove protective cap from line.

NOTE: AN6011-1B gage incorporated shatterproof lens therefore protective shield is not required over gage lens. If replacement gage does not have shatterproof lens and protective shield is not provided, it is recommended to fabricate a shield.

(2) Install gage and secure with screws, connect line.

(3) If required install protective shield and secure with screws.

(4) Open oxygen supply shutoff valve on pilots outboard skirt panel. Gage should indicate system pressure.

(5) Using Leak-Tec Formula check for leaks. Absence of bubbles indicates satisfactory connection.

(6) Wipe fitting and connection clean.

(7) Close oxygen supply shutoff valve on pilots outboard skirt panel.

6. Portable Oxygen Bottle — Removal / Installation

A. Removal

(1) Remove oxygen bottle.

B. Installation

(1) Install oxygen bottle.

(2) Record date of last hydrostatic test.

(3) Perform Portable Oxygen Bottle — Functional Test, see this Section.

7. Portable Oxygen Bottle — Hydrostatic Test

WARNING: THE HYDROSTATIC TEST MUST BE ACCOMPLISHED BY EXPERIENCED PERSONNEL IN A FACILITY EQUIPPED TO PERFORM THIS TEST.

A. Remove oxygen bottle.

B. Perform hydrostatic test.

C. Install oxygen bottle.

8. Portable Oxygen Bottle Regulator — Removal / Installation

A. Removal

(1) Remove regulator from bottle.

B. Installation

(1) Install regulator on bottle.

(2) Perform Portable Oxygen Bottle — Functional Test, see this Section.

9. Portable Oxygen Bottle — Functional Test

A. Perform functional test in accordance with manufacture's instructions.

DILUTER - DEMAND REGULATORS — DESCRIPTION / OPERATION

1. **Description**

The diluter-demand oxygen regulator provides correct mixture of oxygen and air to the user at all altitudes from sea level to 33,000 feet. On demand, it furnishes oxygen only. As the user inhales, a property mixed quantity of oxygen and air is delivered. The percentage of oxygen being delivered increases with increasing altitude, becoming 100% at an altitude of approximately 33,000 feet. The diluting action of air in proper proportion to oxygen is completely automatic.

Each crew member, pilot and copilot, have identical demand-type regulators. These are standard units with diluter and full EMERGENCY selections. The type of mask which is attached to the regulator is left to the discretion of the customers

2. **Operation**

The diluter control lever on the side of the regulator permits either oxygen diluted with air, or 100% oxygen to be supplied on demand. Positioning the diluter control lever to NORMAL OXYGEN permits the supply of oxygen to be diluted with ambient air. With the diluter lever in the 100% OXYGEN position, pure oxygen is delivered by the regulator at all altitudes.

The red emergency valve is provided to make possible the delivery of 100% oxygen flow under pressure directly from the oxygen cylinder. With the emergency valve in the open position, the diluter control lever is ineffective.

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DILUTER - DEMAND REGULATORS — MAINTENANCE PRACTICES

1. Diluter - Demand Regulator — Removal / Installation

WARNING: OIL, GREASE OR ORGANIC MATTER NEAR COMPRESSED OXYGEN MAY RESULT IN SPONTANEOUS COMBUSTION AND/OR EXPLOSION. USE STANDARD MAINTENANCE PRACTICES FOR AVIATION OXYGEN SYSTEMS.

A. Removal

(1) Close oxygen supply shutoff valve (pilots outboard skirt panel.)

(2) Deplete oxygen pressure between shutoff valve and regular by momentarily opening emergency valve on regulator.

(3) Gain access to regulator by opening housing.

(4) Disconnect supply and flow indicator line, cap fittings.

(5) Disconnect breather.

(6) Remove regulator and retain hardware.

(7) Install protective caps on ports.

B. Installation

CAUTION: USE CARE WHEN INSTALLING FITTINGS IN OXYGEN REGULATOR AS TEFLON TAPE REDUCES NORMAL TORQUE FEEL DUE TO ITS LOW FRICTION CHARACTERISTICS. DO NOT OVERTORQUE.

NOTE: If a replacement regulator is to be installed, transfer fittings from old regulator to same ports on replacement regulator. Use teflon tape on tapered pipe end of fitting only. Wrap tape two complete turns starting from tapered end. Wrap in direction fittings is to be tightened.

(1) Remove protective caps from fittings and ports.

(2) Install regulator and secure with hardware.

(3) Connect flow indicator line and supply line.

(4) Connect breathing hose.

(5) Ensure regulator emergency valve is closed.

(6) Slowly open oxygen supply shutoff valve at pilots outboard skirt panel. Observe for system pressure at oxygen pressure gage.

CAUTION: DO NOT LET OXYGEN PRESSURE FALL BELOW 50 PSI.

(7) Using Leak-Tec Formula, check regulator and connections for leaks. If no bubbles appear, connections are satisfactory.

(8) Clean fittings and connections after leak check.

(9) Test regulator for internal leakage. Obstruct outlet of oxygen breathing tube and observe flow indicator on skirt panel. If flow indicator blinker closes in less than 30 seconds, there is excessive leakage in regulator.

(10) Close oxygen supply shutoff valve at pilots outboard skirt panel.

(11) Close access previously opened.

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OXYGEN FLOW INDICATOR — DESCRIPTION / OPERATION

1. **Description**

An oxygen flow indicator functions in conjunction with each diluter-demand regulator to provide a visual indication of the proper operation of demand regulator by blinking at each inhalation of the person wearing an oxygen mask attached to the regulator. The oxygen flow indicator is a pressure-sensitive device that may be connected to the diluter-demand regulator only. The change of oxygen pressure resulting from inhalation and reaction of the spring within the instrument body, is source of operating power.

2. **Operation**

Oxygen under pressure is admitted into oxygen flow indicator from the diluter-demand regulator through the threaded connection in the rear of indicator. The pressure of the oxygen causes a flexible metal bellows within the case to move toward the front of flow indicator against the pressure of a spring. When spring pressure equals applied pressure of oxygen, the bellows comes to rest at some position between its limits of travel, position depending entirely upon the oxygen pressure.

At each inhalation of a person wearing a mask and drawing oxygen from the regulator to which the flow indicator is connected, pressure in the bellows chamber changes approximately 1/2 pound, causing the bellows to assume a new position. When the oxygen pressure returns to its original value (between inhalations), the bellows will assume the position it occupied previous to the 1/2 pound pressure change. The corresponding movements of the bellows cause a pushrod to move back and forth. this pushrod operates a set of shutter leaves by means of a friction clutch, thus opening and closing the leaves to give the effect of a blinker.

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OXYGEN FLOW INDICATOR — MAINTENANCE PRACTICES

1. Oxygen Flow Indicator — Removal / Installation

A. Removal

(1) Ensure oxygen supply (pilots outboard skirt panel) is closed.

(2) Gain access to flow indicator.

(3) Disconnect line from indicator.

(4) Remove indicator and retain hardware.

(5) Install protective cap on line and indicator.

B. Installation

CAUTION: USE CARE WHEN INSTALLING FITTING IN OXYGEN FLOW INDICATOR AS TEFLON TAPE REDUCES NORMAL TORQUE FEEL DUE TO ITS LOW FRICTION CHARACTERISTICS. DO NOT OVER TORQUE.

NOTE: If a replacement flow indicator is to be installed, transfer fitting from old indicator to port on replacement regulator. Use Teflon tape on tapered pipe end of fitting only. Wrap tape two complete turns starting from tapered end. Wrap in direction fitting is to be tightened.

(1) Remove protective caps from indicator and line.

(2) Install indicator and secure with hardware.

(3) Ensure associated regulator emergency valve is closed.

(4) Slowly open oxygen supply shutoff valve at pilots outboard skirt panel. Observe pressure gage is indicating system pressure.

(5) Using Leak-Tec Formula, check indicator and connection for leaks. If no bubbles appear, connections are satisfactory.

(6) Clean fitting and connection after leak check.

(7) Using appropriate regulator and face mask, check operation of flow indicator blinker.

(8) Close oxygen supply shutoff valve at pilots outboard skirt panel.

(9) Inspect area for presents of foreign objects, security of all attachments.

(10) Close access previously opened.

CHAPTER 38

WATER/WASTE

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Information for this chapter to be supplied when equipment is installed.

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AUXILIARY POWER UNIT

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** Pages added by Revision 46
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CHAPTER 49

AUXILIARY POWER UNIT

AUXILIARY POWER UNIT — DESCRIPTION / OPERATION

NOTE: Information contained in this chapter relating to the APU, APU electronic control unit, and any other components supplied by Garrett has been extracted from Garrett data available to Gulfstream Aerospace at the time of this writing, and is included for information purposes only. In the event of any conflict between this information and current or future Garrett information, the Garrett information will take precedence. Maintenance Manuals, Service Bulletins, and other publications to maintain your APU service information in current status are available from:

Garrett General Aviation Services Division

AiResearch Manufacturing Company of Arizona

P.O. Box 29003

Phoenix, AZ. 85038-9003

1. **Description**

The Gulfstream I Auxiliary Power Unit (APU), is used as a source of clean compressed air for the purpose of air-conditioning and pressurization. (See Figure 1) It also provides power to drive a 50A or 200A dc generator, which is used to charge batteries on the ground and supply emergency dc power in flight.

The APU is installed in the tail section, aft of the pressure dome, and is equipped with its own fire detection and extinguishing systems. (See Chapter 26) All operational controls for APU are located on the upper overhead panel.

For further information concerning major components located on the APU, See the Garrett Maintenance Manual for the GTC 85-37-2 APU. Other major components not located on the APU and their locations are listed below.

2. **Bleed Air Control System**

The bleed air control system is made up of the air load control valve, differential pressure regulator and load control thermostat. These units contain the necessary internal diaphragm assemblies, linkages, valves and air pressure and temperature sensing connections for control of the bleed air system.

3. **Lubrication System**

The gas turbine compressor has a complete self-contained lubrication system which provides pressurized and splash lubrication for all gears, shafts and bearings within the engine. The oil tank, mounted on one side of the compressor plenum, is equipped with an oil filler neck and screen. There is a drain port in the bottom of the tank. Oil tank capacity is 6.0 quarts; the oil type used is in accordance with lubricants approved by Garrett.

Oil temperature is regulated by the oil temperature regulator. The regulator incorporates an airflow, tube-type oil cooler, and a thermostatic temperature control valve. The valve keeps the oil recirculating within the gas turbine compressor until the oil temperature rises to 49°C (120°F) above inlet air temperature. As the temperature of the oil in the compressor rises above this level, the hot oil flows through the valve to the oil cooler, from the cooler, if flows to the oil tank after being cooled.

Lubrication oil pressure is utilized for operation of the fuel and ignition sequence switch. This switch is of the diaphragm actuated type, it is located on the upper side of the turbine plenum. The switch is calibrated to close the electric circuit to the fuel solenoid valve at a predetermined engine oil pressure of 2.5 to 3.5 psig (approximately 13% rpm), allowing fuel flow to the engine. The low oil pressure indicator switch is mounted adjacent to the fuel and ignition sequence switch and is also of the diaphragm actuated type. Indication of low oil pressure is provided when the switch functions and the low oil pressure warning light on the APU control section of the upper overhead panel comes on.

4. **Electrical System**

The APU electrical system, when supplied with a power source of 26 $\pm$2 volts dc, provides the means for starting, operating and stopping the compressor. (See Figure 2) The starter motor consists of a 24 to 28 V dc, series-wound motor and clutch assembly. The clutch assembly is comprised of three spring loaded pawls which engage, through inertia, with a jaw on the end of an accessory shaft and a friction clutch which is spring loaded to permit slippage at 100 to 110 inch pounds. The clutch mechanism provides automatic initial engagement of the starter motor; it automatically disengages when the speed of the accessory shaft exceeds that of the starter motor. A centrifugal switch assembly, mounted on the left side of the accessory section adjacent to the oil pump assembly, is driven by the accessory gear train. The centrifugal switch is of the self-actuating, self-resetting, flyweight type; it consists of three centrifugally actuated electrical switches. These switches are set to operate at 40 percent, 95 percent and 110 percent of governed speed, respectively. The 40 percent switch controls the operation of the ignition and starter circuit, and control circuits not required for starting the engine. These include opening of the circuit to the acceleration stabilizer and adjustable orifice assembly. The 95 percent switch energizes the load circuit to indicate that the engine is safe for loading above that rpm, and powers the starts counter and hour meter. The 110 percent switch functions to open the circuit to the fuel solenoid valve when an overspeed condition exists, and shuts off the fuel flow to the combustion chamber to stop the engine. The generator, driven by an auxiliary gear in the accessory gear train through a belt drive, is a continuous-duty unit rated at 50 amperes, 27V regulated dc on aircraft 4 - 60 and 114, and 200 amperes, 27V regulated dc on Aircraft 61 - 200, 322, and 323 (unless customer modified).

5. **APU Fire Detection System**

Three thermal switches are placed strategically in the APU container. Two are set for 450°F, the other for 600°F. Any one of these switches experiencing these temperatures closes a circuit and the APU FIRE light on the copilot side of the eyebrow panel will come on. This circuit is tested in conjunction with the FIRE DET TEST switch, on the pilots side of the eyebrow panel, along with the engine and nacelle systems. (See Chapter 26)

6. **APU Fire Extinguishing System**

Fire extinguishing within the APU container is accomplished by a single shot, electrically fired, 2 1/2 pound, CF_3BR fire-extinguishing bottle adjacent to the unit. The bottle is actuated by the APU FIRE EXT. switch on the copilot side of the eyebrow panel just outboard of the APU FIRE light. This is a guarded switch.

Firing this bottle injects the entire contents directly into the APU container through a short piece of tubing in the container.

WARNING: SHUT DOWN THE UNIT BY PLACING MASTER SWITCH OFF BEFORE FIRING THIS BOTTLE. OBSERVE EMERGENCY PROCEDURES FOR APU FIRE OUTLINED IN THE FLIGHT MANUAL.

7. **Operation**

A. General

Electrical power for APU operation is provided from the essential dc bus. This is accomplished by placing the master battery switch to the ON position. All APU controls will be located in the upper overhead panel in the cockpit. The start cycle is initiated by placing the APU master switch to ON and depressing the start button. Air is supplied to the APU through the inlet door on the aft upper fuselage, above the APU. The APU delivers power in the form of compressed clean air for air-conditioning and pressurization, and 28V dc electrical power from charging batteries on the ground, and essential dc power in flight. Pneumatic and electrical controls function automatically to control operation of the engine within specified limits.

B. Starting Sequence and Automatic Operation.

WARNING: AT INITIATION OF OPERATION AND DURING OPERATION OF THE ENGINE, PERSONNEL MUST STAND CLEAR OF THE COMPRESSOR AIR INLET AND THE HIGH TEMPERATURE EXHAUST. EXTREME CARE MUST BE EXERCISED TO PREVENT DEBRIS OR ANY OTHER FOREIGN MATERIAL FROM ENTERING THE AIR INLET. TURBINE OR COMPRESSOR FAILURE, WHICH MAY BE INDUCED BY FOREIGN MATERIAL ENTERING THE ENGINE, MAY BE SUFFICIENTLY VIOLENT TO DAMAGE THE EQUIPMENT AND ENDANGER NEARBY PERSONNEL.

When the APU master switch is placed in the ON position, electrical power is made available to the APU electrical system. As a result, the following operations are performed. (See Figure 3)

(1) The left normal booster pump comes on, regardless of booster pump switch position. This action delivers fuel through a continuous fuel line to an external fuel shutoff valve just outside of the unit container.

NOTE: On aircraft having ASC 146, which provides a second source of fuel for the APU, it is possible to supply fuel for the APU from the right hand tank by setting the R NORM boost pump switch to the ON position. This procedure would permit operation of the APU if the left normal boost pump had failed. The APU master switch has no effect on the operation of the right normal boost pump.

(2) The APU fuel shutoff valve opens. This allows the fuel to enter the unit up to its internal fuel sequencing valve. The internal fuel sequencing valve is closed, thus the fuel goes no further, but it is available for injection at the proper time in the automatic cycle.

(3) The APU air inlet door opens through an electrically actuated door motor. This makes air available to the unit when the starter is engaged. This door actuates a door interlock switch when the door is fully open. Unless the door is fully open, the unit cannot be started. Door opening takes about five seconds.

(4) The MAST switch also completes certain electrical feeds to the unit, making electrical power available for control and starting purposes.

(5) The low oil pressure (red) light for the APU comes on, indicating low oil pressure within the unit. (As the unit develops sufficient oil pressure this light goes out.)

CAUTION: DO NOT START THE UNIT WITH THE LOAD SWITCHES ON, FOR IT IS POSSIBLE TO FLAME OUT THE UNIT AT ALTITUDE IF THE AIR LOAD IS IMPOSED SIMULTANEOUSLY WITH THE LATTER PART OF THE ACCELERATION CYCLE. ALWAYS WAIT UNTIL THE GREEN LIGHT COMES ON THEN SELECT LOAD SWITCHES.

Pressing the start switch energizes the relays, actuates the start light, closes the oil drain solenoid and acceleration stabilizer solenoid, and initiates starter motor rotation. The starter motor pawls mesh with the accessory gear train and accelerate the engine. When a speed of approximately 5000 rpm (13 percent) is attained, rising oil pressure actuates the fuel and ignition sequence switch which closes circuits to the fuel solenoid valve and ignition system. The fuel solenoid valve opens to allow fuel to flow to the fuel atomizer, where it is sprayed into the combustion chamber and mixed with compressed air from the compressor section; the mixture is then ignited by the igniter plug.

Acceleration of the engine continues under the combined drive of the starter motor and combustion until a speed of approximately 16,750 rpm (40 percent) is reached. At this point the 40 percent switch in the centrifugal switch assembly is actuated, opening the circuits to the starter motor relay, ignition system, and acceleration stabilizer solenoid. The starter motor clutch assembly automatically disengages as the engine speed increases. As the engine speed reaches 95 percent of governed speed, the 95 percent switch in the centrifugal switch assembly is actuated, closing circuits to the APU loading relay, the OPER. RPM light, and powers the starts counter and hourmeter. The engine continues to accelerate under its own power to nominal governed speed of 36,000 RPM. An hourmeter and starts counter are mounted on the unit and actuate during the starting sequence.

CAUTION: ENSURE THAT THE RED LOW OIL PRESSURE LIGHT GOES OUT. UNDER NO CIRCUMSTANCES SHOULD THE GREEN OPER RPM LIGHT AND THE LOW OIL PRESSURE LIGHT BE ON TOGETHER. IF THIS OCCURS, SHUT THE UNIT DOWN IMMEDIATELY BY TURNING OFF THE APU MASTER SWITCH.

IF ENGINE FAILS TO LIGHT-OFF OR ACCELERATE ON INITIAL START ONLY, A RESTART IS PERMISSIBLE PROVIDING UNIT IS ALLOWED TO COME TO A COMPLETE STOP. POSITION MASTER SWITCH TO OFF, WAIT 30 SECONDS, THEN BEGIN NEW STARTING SEQUENCE.

DO NOT EXCEED STARTER MOTOR DUTY CYCLE OF ONE MINUTE ON AND FOUR MINUTES OFF.

C. Governed operation.

Engine speed is maintained within specified operating limits by the action of the fuel pump and control unit in supplying fuel under varying pressures to the combustion chamber.

Control of fuel pressure in relation to compressor discharge pressure is accomplished by the fuel pump acceleration limiter valve, which senses compressor pressure through a pneumatic line, and is actuated to bypass more or less fuel to the combustion chamber as compressor pressure varies. During the starting cycle, the acceleration stabilizer and orifice assembly functions to bleed off peaks of control air from the compressor to the limiter, thereby stabilizing the limiter bypassing action and consequently the actual fuel flow.

Control of fuel pressure in relation to turbine exhaust temperature is also accomplished by the fuel by-passing action of the fuel pump acceleration limiter valve. The acceleration and overtemperature control thermostat, connected to the limiter through a pneumatic line, is set to open at excessive turbine exhaust temperatures, and bleeds off control air from the limiter, which then bypasses more fuel to the combustion chamber.

Control of fuel pressure in relation to engine speed is accomplished by the fuel pump governor section, which senses engine speed by means of flyweights. As the engine speed exceeds predetermined limits, the flyweights move out and open a bypass valve to divert more or less fuel from the combustion chamber as speed increases or decreases. Further overspeed control of the engine is accomplished by the overspeed switch in the centrifugal switch assembly; when engine speed exceeds approximately 110 percent of governed speed, this switch, actuated by flyweights, de-energizes the fuel solenoid valve, thereby shutting off fuel flow to the combustion chamber and stopping the engine.

D. Air Load Operation

Loading the engine is accomplished by bleeding compressed air from the compressor through the air load control valve after engine speed reached 95 percent of governed speed. At this speed the 95 percent switch in the centrifugal switch assembly actuates to energize the OPER RPM light which glows to indicate the engine is ready for loading, plus closing the APU loading relay.

When bleed air is desired, the air switch is placed in the ON position, energizing the air load control valve solenoid. The solenoid valve opens a switcher valve and allows regulated control air from the compressor section to actuate the air load control valve actuator, which opens the butterfly valve and permits bleed air flow.

NOTE: Whenever practicable, operation of the unit should be planned a few minutes in advance to allow a three to five minute warm-up period at governed speed before air load is applied. This is not mandatory, but is recommended when the warmup delay will be of no serious consequence. Temperature gradients, and thermal stresses across the turbine wheel can be materially reduced resulting in increased unit life.

The air load control thermostat is connected to the air load control valve actuator through a pneumatic line. The thermostat senses increased turbine exhaust temperature as a result of excessive loading, and opens at a preset temperature to bleed off control air to the actuator. The actuator operates to decrease butterfly valve opening to limit air loading while allowing engine to maintain operating speed.

WARNING: THE APU AIR INCREASE/DECREASE VALVE IN THE BAGGAGE COMPARTMENT, MUST BE FULLY CLOSED (FULL INCREASE) BEFORE AND DURING FLIGHT TO ENSURE MAXIMUM AIRFLOW FROM THE APU SHOULD IT BE REQUIRED AS AN ALTERNATE AIR CONDITIONING AND PRESSURIZATION SOURCE. THE INCREASE/DECREASE VALVE IS INTENDED ONLY FOR USE DURING GROUND AIR CONDITIONING OPERATIONS.

An APU INCR/DECR valve is tee'd into the pneumatic control line to the air load control valve. It is manually controlled by the crew to modify airflows from the APU on the ground to prevent wind tunnel effect if the aircraft is being air conditioned with doors and/or windows open. This valve is located on the right side Fuselage Station 561 (pressure dome) just aft of the baggage door frame. It is accessible during flight.

CAUTION: THIS VALVE MUST BE FULLY CLOSED BEFORE FLIGHT.

When the bleed air load is removed from the engine by placing the APU AIR switch in the OFF position, air load control valve solenoid, de-energized, moves the switcher valve to divert the regulated control air to the other side of the shutoff valve actuator, causing the actuator to close the butterfly valve and shutoff bleed air.

Aircraft 1 - 200, 322 and 323 having Customer Bulletin 124 incorporated have a solenoid operated shutoff valve which is piped directly to the APU AIR increased/decreased needle valve as part of a remote control of APU airflow. Control of this valve is performed at a three-way APU AIR switch which is installed in place of the ON-OFF APU AIR switch normally supplied on the upper overhead panel. This optional modification enables the crew to attain full or partial APU AIRFLOW entirely from the cockpit, without having to go to the baggage compartment to operate the increase/decrease valve. It can be initially preset to a suitable airflow position for ground air conditioning. The adjustment of the APU airflow valve to give a suitable airflow will vary with the amount of heating or cooling required. From then on the crew, through the electrical circuitry, merely selects FLIGHT (max. airflow), GROUND (reduced airflow), or OFF (no airflow) by operating the APU AIR switch from the cockpit.

During flight in which the APU is operated for air supply the APU AIR switch on Aircraft 1 - 200, 322 and 323 having Customer Bulletin 124, is placed in the FLIGHT position to supply full APU air output.

Aircraft having ASC 81, the supercharger protection relay is installed to prevent APU air and blower air from being on simultaneously. This relay is energized when APU air switch is placed in the ON position. It dumps the blower on the right engine and through the APU loading relay powers the air loading valve solenoid to allow the APU to deliver air to the system.

E. Shutdown.

Normally, if using the unit for air supply, the AIR switch is placed in the OFF position for about one minute before shutting down the unit. This enables turbine wheel to cool down slower than a full and immediate shutdown.

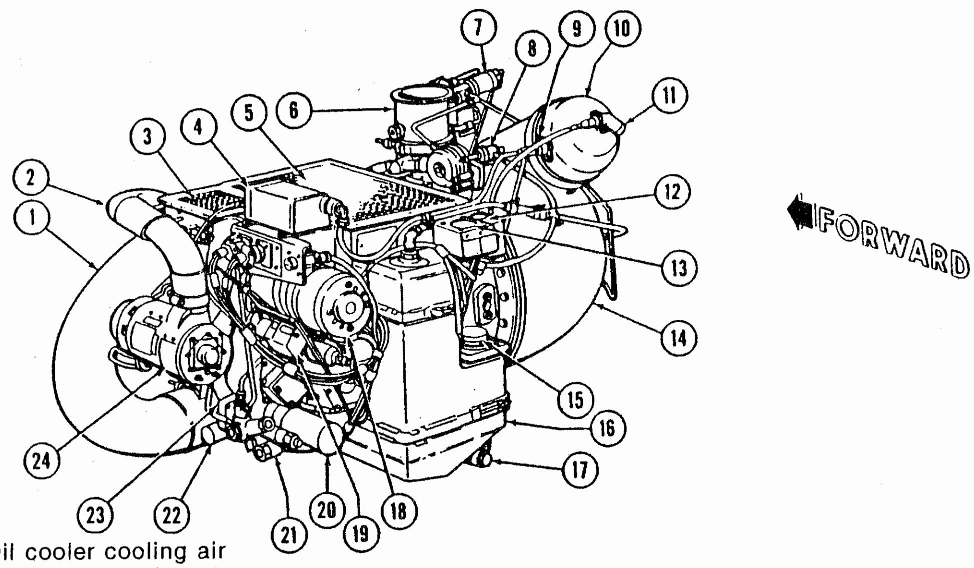
Shutting down the unit consists of placing the APU MAST switch in the OFF position. The fuel and electrical ceases, unit operation terminates and the air inlet door closes.

F. APU Air Ground Operating Practices

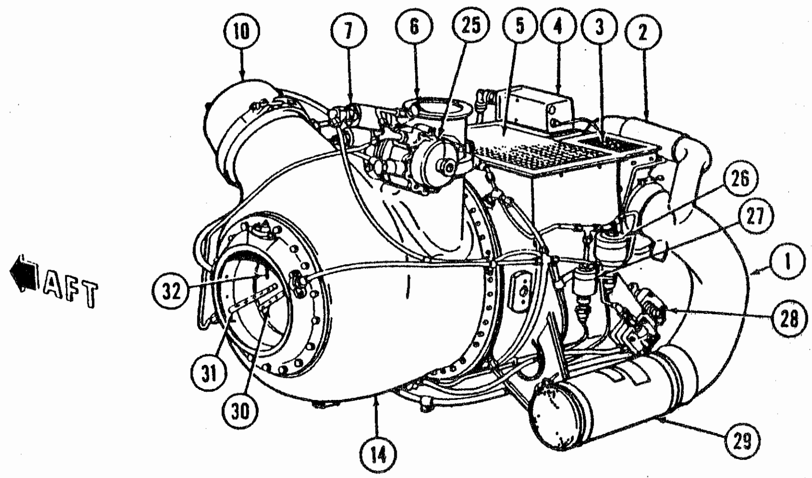
To provide a comfortable air conditioned aircraft the APU should be utilized prior to boarding passengers. After starting unit, place APU AIR switch to ON. In stabilizing the temperature quickly a higher airflow can be utilized before passengers enter by operating the APU INCREASE/DECREASE valve in the baggage compartment to INCREASE. On Aircraft having Customer Bulletin 124, place the APU switch in the cockpit in the GROUND position. The APU flow is reduced or shut down before boarding passengers to avoid the wind tunnel effect. With passengers aboard and the APU supplying air conditioning either the main door or a direct vision window is opened before air loading the APU.

CAUTION: DO NOT OPEN OR CLOSE THE MAIN DOOR WITH THE APU AIR ON, UNLESS A DV WINDOW IS OPEN. OPENING OR CLOSING THE DV WINDOW SHOULD BE DONE GENTLY TO PREVENT A PRESSURE SURGE. IF IT IS NECESSARY TO OPERATE THE APU FOR AIR CONDITIONING PURPOSES, DON'T HAVE APU AIR SWITCH ON DURING ENGINE STARTS OR ENGINE RPM'S BELOW 11,000 (NO PROP WASH).

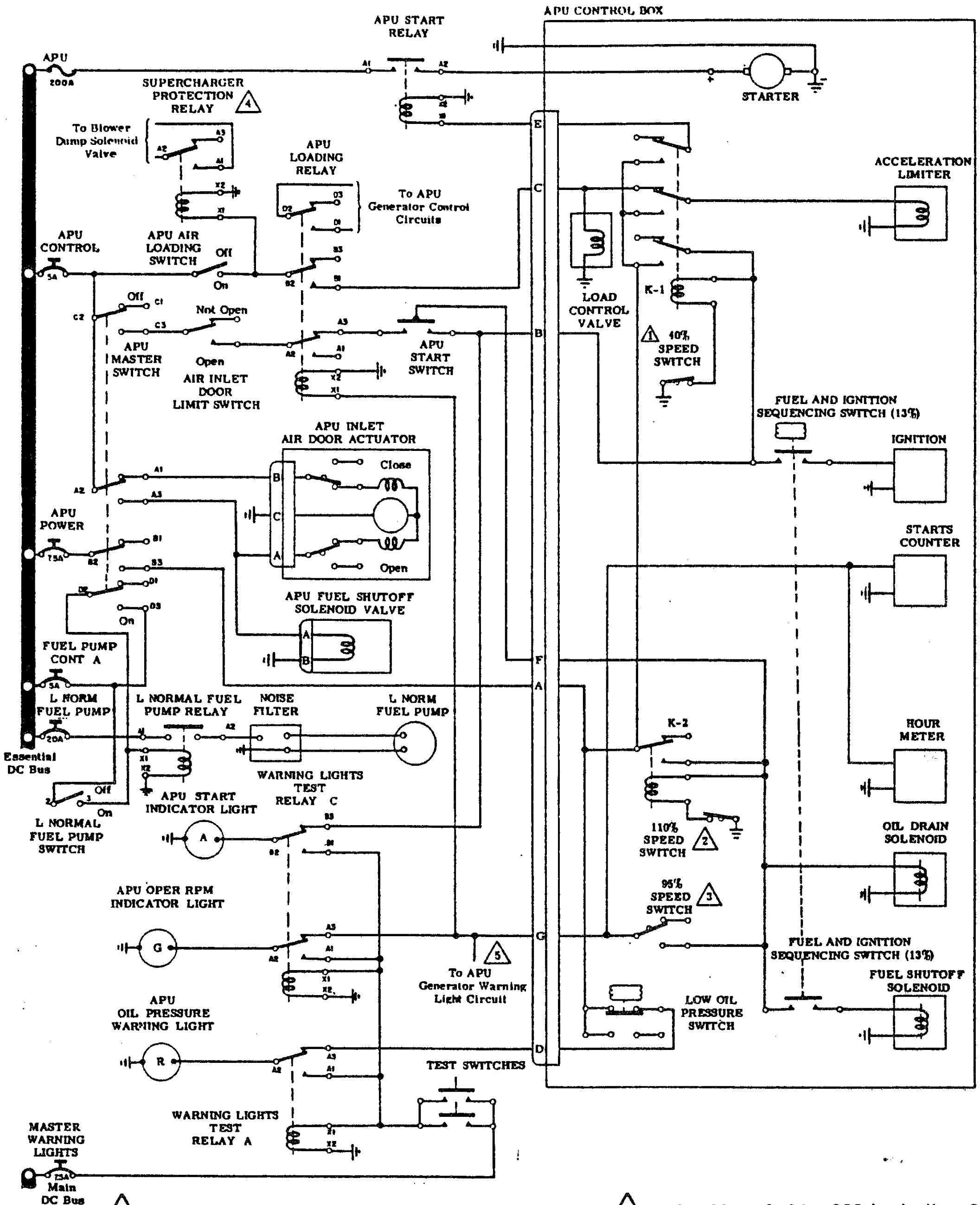
All of the above will prevent the APU from ingesting engine exhaust fumes, or prevent objectionable pressure surges for passenger comfort. To prevent ingestion of fumes by the APU while parked, taxiing, or prior to takeoff, avoid any significant crosswind with the APU AIR switch in the ON position. A crosswind from the left should particularly be avoided due to the relative location of the APU air inlet and the left engine exhaust.



1. Duct—Oil cooler cooling air
2. Duct—Generator cooling air
3. Inlet—Fan cooling air
4. Ignition Unit
5. Inlet—Compressor air
6. Valve—Air shutoff
7. Solenoid—Air shutoff valve
8. Regulator—Air pressure
9. Plug—Igniter
10. Chamber—Combustion
11. Atomizer Assy—Fuel
12. Counter—Start
13. Hourmeter
14. Plenum—Turbine
15. Filler Neck—Oil tank
16. Tank—Oil
17. Drain—Oil tank
18. Motor Assy—Starter
19. Switch Assy—Centrifugal
20. Filter—Oil pump
21. Pump Assy—Oil
22. Filter—Fuel pump
23. Pump and Control Unit—Fuel
24. Generator
25. Actuator—Air shutoff valve
26. Switch—Fuel and ignition sequence
27. Switch—Low oil pressure shutoff
28. Pump and Control Unit—Fuel
29. Cooler Assy—Oil
30. Thermostat—Acceleration and over-temperature control
31. Thermostat—Load control
32. Thermocouple

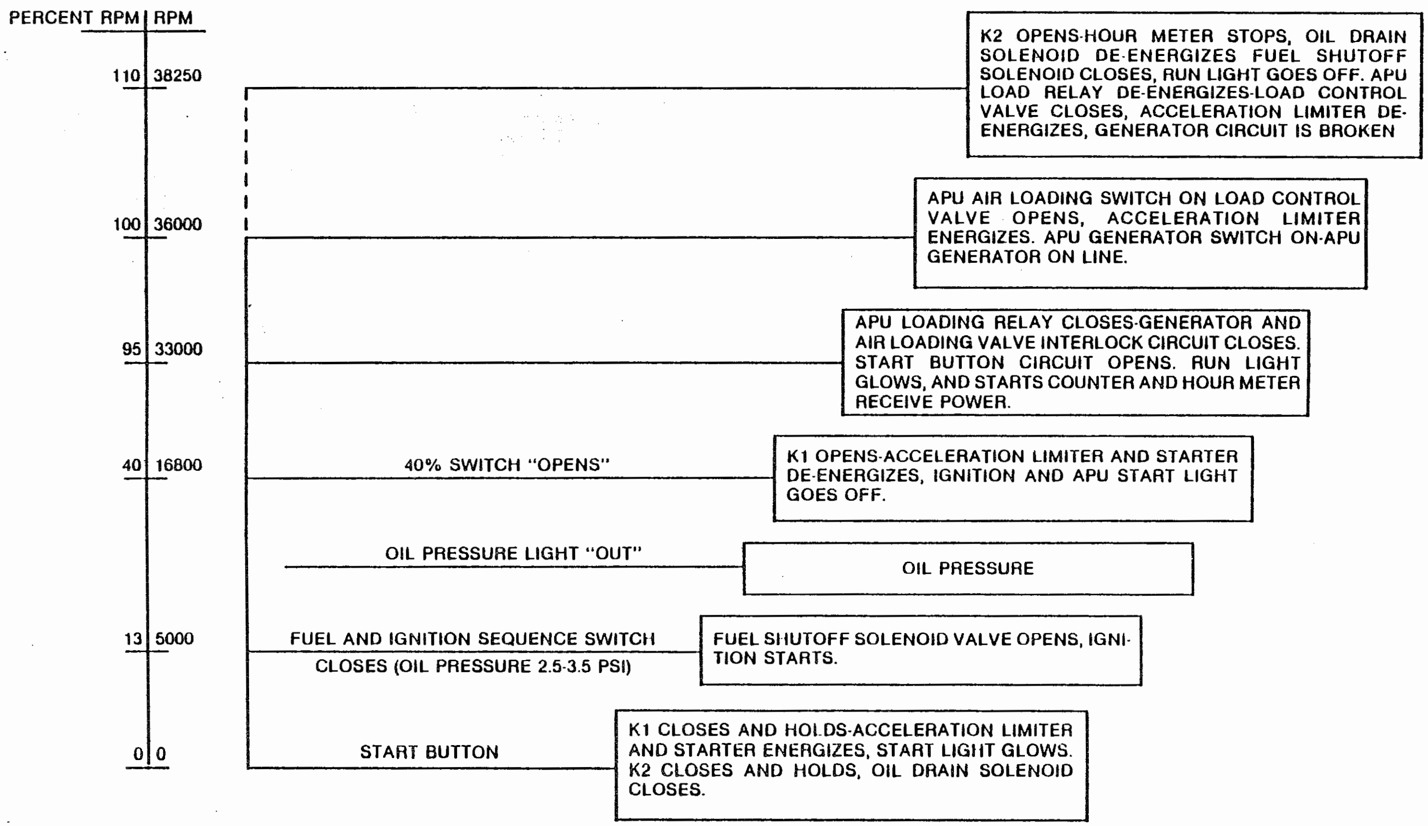


Auxilliary Power Unit
Figure 1.



1. 40% Speed SW Opens at 40% Speed.
2. 110% Speed SW Opens at 110% Speed.
3. 95% Speed SW Completes Circuit At 95% Speed.
4. On Aircraft 61 - 200 including 322, 323 and Aircraft having ASC 81.
5. On Aircraft 102 - 200 including 322, 323 and Aircraft having ASC 158A.

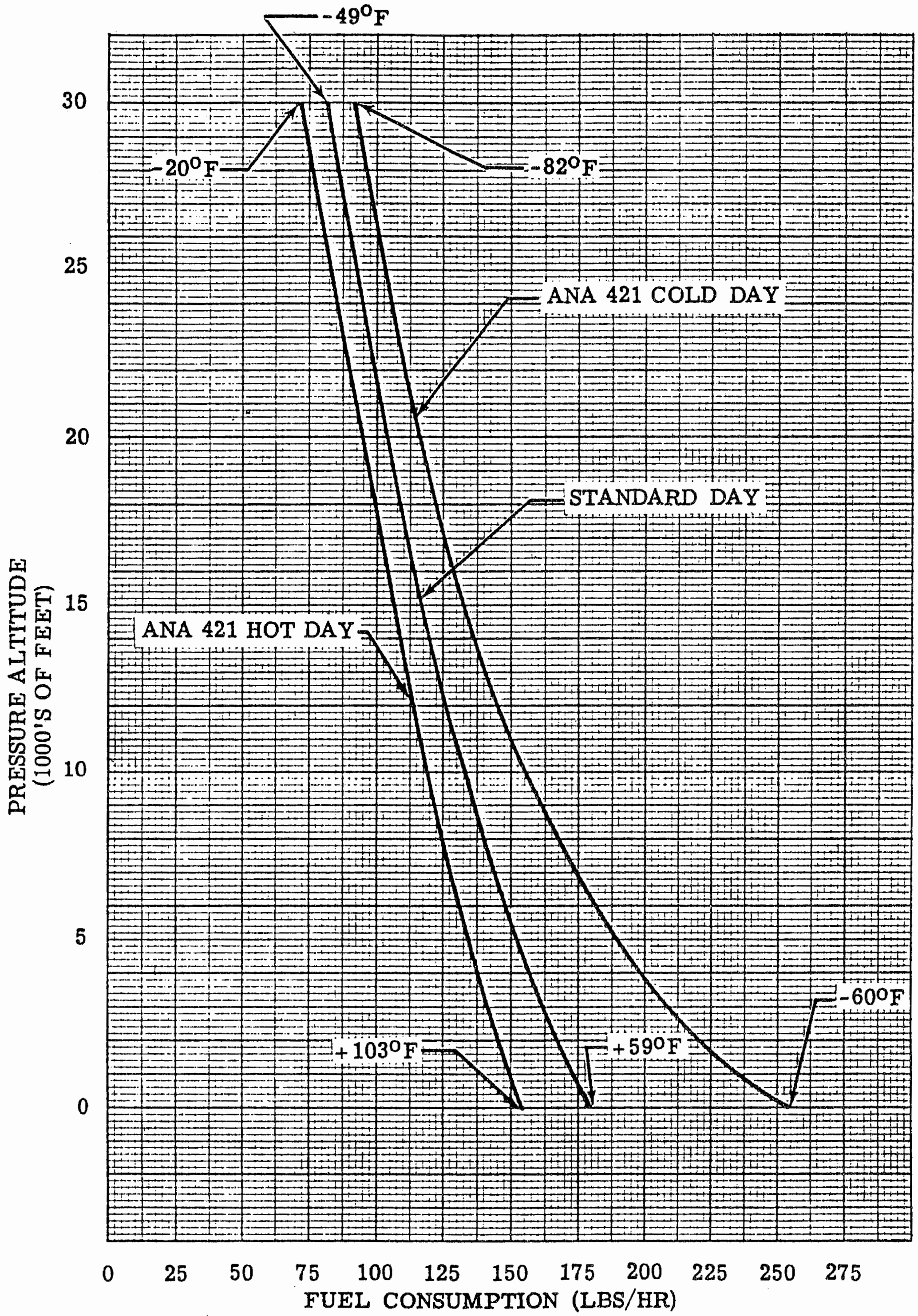
Auxilliary Power Unit Control Circuit — Schematic
Figure 2.



MASTER SWITCH ON

1. DOOR OPEN
2. L NORM FUEL PUMP ON
3. APU FUEL SHUTOFF VALVE OPEN.
4. LOW OIL PRESSURE LIGHT ON.

APU Operation Sequence
Figure 3.



Approximate APU Fuel Consumption (Maximum)
Figure 4.

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AUXILIARY POWER UNIT — FAULT ISOLATION

FAULT	POSSIBLE CAUSE	CORRECTION
Engine fails to operate when START switch is pressed.	Power supply defective.	Check circuit and power supply. Replace defective wiring or 200A fuse.
	Starter relay defective.	Replace starter relay.
	Starter switch defective.	Replace starter switch.
	Starter motor assembly defective.	Replace starter motor assembly.
	40% switch open.	Check centrifugal switch assembly and replace if necessary.
Starter operates, but does not motor engine.	Starter motor assembly defective.	Replace starter motor assembly.
	Starter clutch assembly defective.	Replace starter motor assembly.
	Accessory drive gears stripped.	Replace complete engine.
	Torsion shaft failure.	Check for positive coupling through engine by manually turning turbine wheel. Replace complete engine if torsion shaft is defective.
Starter motors engine, but combustion does not occur.	Fuel supply inadequate.	Disconnect fuel inlet line, check for fuel flow. Check fuel supply. Check or replace fuel filters.
	Pneumatic control line connection loose.	Tighten connection.
	Igniter plug fouled or defective.	Clean or replace igniter plug.
	Electrical power input low.	Check power supply voltage.
	Fuel solenoid valve circuit open, or fuel solenoid valve defective.	Check circuit. Replace fuel solenoid valve.
	Centrifugal switch circuit open or centrifugal switch defective.	Check circuit. Replace centrifugal switch assembly.
	Ignition unit defective.	Replace ignition unit.
	Fuel pump and control unit defective.	Replace fuel pump and control unit.
	Fuel atomizer orifice clogged.	Clean orifice or replace atomizer assembly.
	Oil pressure low.	Check and replace oil filter.
	Fuel and ignition sequence switch defective.	Replace switch.
	Low oil pressure shutoff switch stuck or defective.	Check circuit. Replace low oil pressure shutoff switch.

FAULT	POSSIBLE CAUSE	CORRECTION
Engine shuts down immediately after combustion occurs.	Holding relay defective.	Replace relay.
	Fuel and ignition sequence switch defective.	Replace switch.
	Ignition unit defective.	Replace ignition unit.
	Fuel pump and control unit defective.	Replace fuel pump and control unit.
Combustion occurs, but engine does not accelerate to governed speed.	Pneumatic control line connection loose.	Tighten connection.
	Acceleration and overtemperature control thermostat defective.	Replace thermostat.
	Fuel pump and control unit defective.	Replace fuel pump and control unit.
	Centrifugal switch defective.	Replace centrifugal switch assembly.
Rate of acceleration too slow.	Pneumatic control line connection loose.	Tighten connection.
	Acceleration and overtemperature control thermostat defective.	Replace thermostat.
	Fuel pump and control unit defective.	Replace fuel pump and control unit.
	Fuel atomizer orifice clogged.	Clean orifice or replace atomizer assembly.
	Air being bled from unit.	Check closed position of pneumatic shutoff valve. Close or replace if partially open. Check for leakage from ducts and turbine plenum.
	Acceleration stabilizer solenoid defective.	Replace acceleration stabilizer solenoid.
Rate of acceleration too fast.	Fuel pump and control unit defective.	Replace fuel pump and control unit.
	Acceleration and overtemperature control thermostat defective.	Replace thermostat.
	Acceleration stabilizer and adjust able orifice defective.	Check for proper actuation. Replace if necessary.
Engine accelerates at excessive turbine discharge temperature.	Acceleration and overtemperature control thermostat defective.	Replace thermostat.
	Fuel pump and control unit defective.	Replace fuel pump and control unit.
Smoke emitted for short time after start.	Excess of oil in system or oil tank.	Drain oil to proper level.
	Oil pump check valve leaking.	Check and clean check valve. Replace oil pump assembly if necessary.
	Scavenge oil pump defective.	Replace oil pump assembly.

FAULT	POSSIBLE CAUSE	CORRECTION
Engine exhaust smokes constantly.	Accessory housing vent plug loose.	Check visually. Tighten plug.
	Oil cooler leaking.	Check visually. Replace if necessary.
	Packing failure on oil jet and vent tubes, oil drain and vent tube, turbine end drain connection.	Replace parts and replace defective packing.
	Turbine drain check valve sticking closed.	Clean or replace check valve.
No initial oil pressure.	No oil prime.	Prime oil pump during starting. (After new installation, oil change, and after replacing oil filter only.)
	Oil supply low.	Replenish oil supply.
	Oil pump assembly defective.	Replace oil pump assembly.
	Oil filter dirty.	Replace oil filter.
Loss of oil pressure.	Oil pump check valve sticking open.	Push in primer cap slowly. Approximately halfway in, primer plunger should contact check valve. If contact is not felt, check valve is open. Replace oil pump assembly.
	Oil supply low.	Replenish oil supply.
	Oil filter dirty.	Replace oil filter.
Excessive oil pressure.	Oil pump check valve sticking closed, or valve spring cocked.	Push in primer cap slowly. Approximately halfway in, primer plunger should contact check valve. If contact is not felt, check valve is open. Replace oil pump assembly.
Excessive oil temperature.	Thermostatic temperature control valve defective.	Replace thermostatic temperature control valve.
	Oil supply low. Cooling air restricted.	Replenish oil supply. Check cooling air duct to oil temperature regulator assembly.
	Turbine seal failure.	Replace complete engine.
	Bearing failure.	Replace complete engine.
Engine exhaust flames when starting.	Turbine drain check valve sticking closed.	Clean or replace check valve.

FAULT	POSSIBLE CAUSE	CORRECTION
Fuel leaks from fuel and oil drain line.	Dirt under fuel pump drive shaft seal retainer, or packing failure.	Remove fuel pump and control unit. Remove drive shaft, seal retainer, and packings. Clean out foreign material and replace packings.
	Fuel pump drive shaft burred.	Remove fuel pump and control unit; check shaft. Replace fuel pump and control unit.
	Fuel pump and control unit misaligned on accessory housing pad.	Loosen attaching nuts and check alignment.
	Fuel pump drive shaft bottoming.	Loosen attaching nuts and check insulator and gaskets.
Air load control valve fails to open.	External wiring to APU AIR switch defective.	Check APU AIR switch and external circuit.
	Air load control valve solenoid defective.	Replace solenoid.
	Air pressure regulator defective. Pneumatic control line connection loose.	Replace air pressure regulator. Tighten connection.
	Load control thermostat sticking open.	Disconnect line from thermostat. No leakage should occur at less than 35 psig. If leakage occurs, replace thermostat.
	Air load control valve rate of change adjustment screw in too far.	If air leaks from rate control bleed port, turn rate adjustment screw out 1/4 turn and repeat test. If leakage persists, replace shutoff valve.
	Air load control valve poppet valve sticking open.	If air leaks from rate control bleed port, turn rate adjustment screw out 1/4 turn, and repeat test. If leakage persists, poppet valve is sticking. Replace valve.
	Air load control valve diaphragm ruptured.	Disconnect control air inlet line from regulator at valve. Remove fitting from actuator port on valve. Blow into port. Leakage from inlet connection indicates ruptured diaphragm. Replace valve.

FAULT	POSSIBLE CAUSE	CORRECTION
Engine speed drops more than 1000 rpm (maximum) as load is applied.	Load control thermostat set high.	Correct thermostat setting, or replace thermostat.
	Fuel pump governor setting improper.	If control air pressure at idle and load is normal, and peak fuel pressure during start is normal, check and adjust governor setting.
	Mechanical overload.	If engine takes extended time to start, and idle fuel pressure is high, check for mechanical malfunction. Replace complete engine, if necessary.
	Control air leakage.	If control air pressure is low at idle, and peak fuel pressure is low during start, check for loose control air connection and tighten. Check acceleration stabilizer and adjustable orifice for leakage.
	Acceleration and overtemperature control thermostat set low.	Check for control air pressure as maximum load is approached. If pressure drops below 14 psig, clean, adjust, or replace thermostat.
	Fuel supply to engine inadequate. Clogged screen in atomizer.	Check for fluctuating fuel pressure as load is applied. Check external fuel supply system for maintenance of pressure by connecting low pressure fuel gage to alternate bypass port of fuel pump and control unit.
	Fuel pump and control unit defective.	Replace fuel pump and control unit.
Engine fails to stop when APU MAST is pressed to OFF.	Holding relay defective (K2).	Replace holding relay (K2).
	Fuel solenoid valve defective.	Replace fuel solenoid valve.
	Switch defective.	Replace switch.
NOTE: For APU generator troubleshooting See Chapter 24.		

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AUXILIARY POWER UNIT — MAINTENANCE PRACTICES

1. **General Maintenance**

   A. Lubrication requirements are contained in Chapter 12 of Maintenance Manual or Garrett Maintenance Manual GTC85-37-2.

   B. Preflight Inspection should include the following:

   - Visually inspect all wiring, tubing, ducts and fittings for security of mounting and signs of leakage.
   - Inspect compressor air intake and turbine exhaust ducts for foreign material or other obstructions.
   - Check oil level in tank; add oil as required until level reaches full mark on dipstick.
   - Check for adequate fuel supply and security of connections.
   - Check for security of electrical connections, check for connection of external power source to engine electrical system.

   C. Spectrographic Oil Analysis

   (1) After APU run, open oil filter drain plug and sample jar 3/4 full for Spectrographic Oil Analysis.

   (2) Replace oil plug.

2. **APU — Operational Test**

WARNING: WHEN PERFORMING ANY OF THE FOLLOWING TESTS, ENSURE THAT ADEQUATE FIRE EXTINGUISHING EQUIPMENT IS AVAILABLE.

   A. In baggage compartment, turn APU increase-decrease valve to full decrease.

   B. Engage following circuit breakers; APU CONT, APU PWR, APU GEN, H_2O SEP, CBN/CKPT TEMP, L NORM PUMP and PUMP CONT A.

   C. Position APU master, air and generator switches to OFF.

   D. Energize main and essential dc buses from aircraft battery only (position BATT switch to NORM.)

   E. Hold both cabin and cockpit temperature control switches in manual decrease for at least 40 seconds.

   WARNING: AT INITIATION OF OPERATION AND DURING OPERATION OF ENGINE, PERSONNEL MUST STAND CLEAR OF COMPRESSOR AIR INLET AND HIGH TEMPERATURE EXHAUST. EXTREME CARE MUST BE EXERCISED TO PREVENT DEBRIS OR ANY OTHER FOREIGN MATERIAL FROM ENTERING AIR INLET. TURBINE OR COMPRESSOR FAILURE, WHICH MAY BE INDUCED BY FOREIGN MATERIAL ENTERING THE ENGINE, MAY BE SUFFICIENTLY VIOLENT TO DAMAGE THE EQUIPMENT AND ENDANGER NEARBY PERSONNEL.

   F. Set APU master switch ON. Allow 10 seconds for APU inlet door to open. Observe oil pressure light flashes on and left normal fuel boost pump is running.

   CAUTION: IF APU IS EQUIPPED WITH TGT INDICATOR, MAXIMUM TGT DURING START IS 650°C, AND DURING RUN WITH AIR LOAD 590°C.

   TIME BETWEEN ACTUATING START BUTTON AND FLASHING ON OF OPER RPM LIGHT SHOULD BE 10 TO 20 SECONDS. OBSERVE THAT OIL PRESSURE AND START LIGHTS GO OUT DURING START CYCLE. ON INITIAL START OF NEW OR OVERHAULED ENGINE, OR ON ENGINE THAT HAS NOT BEEN OPERATED FOR SEVERAL DAYS, DEPRESS OIL PRIMER PLUNGER ON OIL PUMP.

   G. Push APU start button and observe that APU start light flashes on.

   H. Place APU AIR switch ON (GROUND if equipped with FLIGHT-GROUND switch).

   I. Turn APU increase-decrease valve toward increase (clockwise) and observe increase of air into cockpit and cabin. Incoming air should be cold. Return switch to original position.

   J. Hold both cabin and cockpit temperature control switches in manual increase position for 10 seconds. Observe that incoming air temperature increases.

3. APU External Fuel Shutoff Valve and APU Fuel Pressure Regulator — Removal / Installation

A. Removal

NOTE: APU external fuel shutoff valve and APU fuel pressure regulator are mounted together and are removed as an assembly.

(1) Remove electrical power from aircraft.

(2) Gain access to assembly in tail compartment just aft and below pressure dome, forward of APU enclosure.

(3) Disconnect electrical plug from shutoff valve.

(4) Disconnect fuel lines upstream of shutoff valve and downstream of pressure regulator.

(5) Disconnect pressure regulator overboard vent line.

(6) Remove assembly.

(7) If either shutoff valve or regulator is to be replaced, separate units on bench and replace unit involved; make up complete assembly as removed.

B. Installation

CAUTION: ENSURE FLOW ARROW ON APU FUEL SHUTOFF VALVE POINTS OUTBOARD (TOWARDS PRESSURE REGULATOR).

(1) Mount assembly on brackets.

(2) Connect following lines:

- Fuel inlet line to shutoff valve.
- Pressure regulator overboard vent line.
- Rigid tubing from regulator outlet to 900 fitting on APU enclosure.

(3) Connect electrical plug to shutoff valve.

(4) Perform APU Fuel Shutoff Valve and Pressure Regulator mdash. Functional Test, this Section.

(5) Start and run APU; check for leaks at connections.

(6) Shutoff APU.

(7) Inspect area for presents of foreign objects, security of all attachments.

(8) Install access panel.

4. APU Fuel Pressure Regulator / Shutoff Valve — Functional Test

NOTE: This procedure incorporates check for both APU external fuel shutoff valve and APU fuel pressure regulator. Both have identical requirements and may be checked at the same time.

A. Gain access to assembly through access panel on left side of aircraft.

B. Disconnect and remove rigid tubing from 90° angle fitting on APU enclosure and outlet on APU fuel pressure regulator.

C. Connect short flex line to 90° angle fitting on APU enclosure and other short flex line to outlet port of APU fuel pressure regulator. Join short flex lines to T-fitting. Connect long flex line to remaining port of T-fitting. Long flex line should extend outside of aircraft. (See Figure 201)

D. Provide a suitable container for catching fuel from open end of flex line.

E. Check APU off (APU master switch, air switch and generator switch).

F. Energize main and essential dc buses (battery switch NORMAL).

G. Place left main boost pump switch to ON.

H. No fuel should flow from open end of flex line (check LOW FUEL PRESSURE light on annunciator panel is out).

I. Place left main boost pump switch OFF (check LOW FUEL PRESSURE light is on).

J. Place APU master switch ON. Fuel should flow from open end of flex line.

K. Place APU master switch OFF. Fuel flow should stop.

L. Connect 0 - 30 psi gage to flex hose. (See Figure 201)

M. Place APU master switch ON. Start and run APU.

N. Pressure gage should indicate 14 ± 2 psi.

O. Place APU master switch OFF.

P. Remove test gage, flex lines and T-fitting and install rigid tubing between APU fuel pressure regulator outlet and 90° fitting on APU enclosure.

Q. Place APU master switch ON and check for leaks at connections.

R. Place APU master switch OFF.

S. Remove electrical power from aircraft.

T. Inspect area for presents of foreign objects, security of all attachments.

U. Install access panel.

5. **APU Enclosure — Inspection**

A. Inspect interior of APU enclosure through access covers 106-APU-4, 106-APU-5, 106-APU-6 and 106-APU-12 for:

(1) Condition of fuel system Installation.

(2) Condition of electrical system Installation.

(3) Condition of bleed air system Installation.

(4) Condition of inlet and exhaust Installation.

(5) Condition of access covers and attaching hardware.

(6) Presence of foreign objects and/or fluid accumulations.

(7) Inspect APU exhaust pipe clamp for damage and security.

(8) Condition and security of container.

(9) Condition of APU mounts and attaching hardware.

(10) Record APU Hours __________ and Starts _________.

B. Air Outlet duct, located upper forward left side of APU box.

(1) Remove housing assembly to gain access to flexible duct.

(2) Inspect for presence of foreign objects then close access covers.

(3) Remove flexible duct (if required)

(4) Inspect duct for cracks, frayed areas, cuts, holes and evidence of leakage.

(5) Install flexible duct And housing assembly.

C. Inspect area for presents of foreign objects, security of all attachments, then close access covers.

6. **APU Tailpipe — Removal / Installation**

(Aircraft having GTC-37-2 APU)

A. Removal

(1) Open APU PWR and APU CONT circuit breakers located on pilots circuit breaker panel.

(2) Remove APU access panel 107-RH APU 1

(3) Gain access to tailpipe clamp by removing APU enclosure access panels 106-APU-1, -2, -3, -11 or -12. (This step for Aircraft not having ASC 228).

CAUTION: ALLOW APU TO COOL DOWN BEFORE PERFORMING NEXT STEP.

(4) Use masking tape to show tailpipe position before clamp removal.

(5) Remove tailpipe clamp and tailpipe.

B. Installation

(1) Install clamp loosely on tailpipe and exhaust flange.

(2) Position tailpipe using tape marks installed in A.(4) above.

(3) Slowly tighten clamp until seated.

NOTE: Use rawhide mallet to seat clamp on APU exhaust flange and tailpipe.

(4) Torque NUCO clamp P/N N4208-812M 50 - 60 inch pounds and Aeroquip clamp P/N 4353C-812M 40 inch-pounds.

(5) Inspect area for presents of foreign objects, security of all attachments.

(6) Install APU access and enclosure panels.

7. APU — Removal / Installation

CAUTION: BEFORE PERFORMING ANY OF THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Open tail access door aft of APU access plate.

NOTE: It is necessary for one man to enter fuselage through tail access door to gain access to APU access splice plates, box assembly and ducting.

(2) Remove large APU access plate on right side of fuselage, disconnect lines and fittings at bottom of box assembly and air inlet line at top forward corner.

CAUTION: TO PREVENT DAMAGE AND TO FACILITATE REMOVAL OF BOX ASSEMBLY, REMOVE FITTINGS FROM BOTTOM OF BOX.

(3) Open APU two section access door on left side of fuselage; disconnect fuel lines, drain lines and fittings at bottom of box assembly.

NOTE: All directions are given as if facing inboard.

(4) Remove screws securing front cover of APU relay box and remove cover.

(5) Ensure electrical power is off.

(6) Disconnect large No. 00 cable from inside box which is connected to APU generator cutout relay.

CAUTION: CABLE IS CONNECTED TO ESSENTIAL DC BUS AND MUST BE INSULATED TO PREVENT DAMAGE IN EVENT DC POWER IS APPLIED WITH APU REMOVED.

(7) Insulate cable terminal to ensure contact cannot be made with structure.

(8) Disconnect No. 10 ground wire (50 amp generator), or No. 0 ground wire (200 amp generator) from box.

(9) Disconnect large electrical connector on left side of APU relay box.

(10) Working in fuselage remove screws that secure lower inlet duct to upper duct.

(11) Unfasten lock fasteners that secure duct to APU box and remove duct.

(12) Remove couplings from upper section of bleed duct assembly and disconnect deicer line.

(13) Remove bleed duct assembly from bleed duct housing and remove housing from box.

(14) Remove V-band clamp and flex bleed duct.

(15) Remove mounting bold access doors from APU box assembly and remove bolts securing box assembly.

(16) Remove APU and box assembly.

NOTE: Use adjustable stand to support and lower unit after removal from aircraft.

(17) Remove old unit from box assembly and install new unit into box assembly. Transfer all electrical harness connections and other hardware as required to make up QEC configuration.

NOTE: When APU is removed, inspect the elevator and rudder bellcranks in the pressure box under the APU for condition, security and operation, as this is the only time that these components are accessible.

If APU is to be removed and sent to repair facility, consult Garrett GTC 85-37-2 APU Illustrated Parts Catalog for attaching mounts peculiar to the Gulfstream installation and remove and retain these parts for reinstallation.

B. Installation

NOTE: Unit is considered assembled, completely configured as QEC and in box assembly.

(1) With box assembly on adjustable stand, raise to align with tracks in fuselage.

(2) Slide unit into position, install and tighten mount bolts; install bolt access covers.

(3) Install flex bleed duct with bleed duct housing positioned over it and secure.

(4) Install upper bleed duct on housing. Ensure flange of flex bleed duct is between housing and upper duct tubing. Install screws and secure.

(5) To install two couplings on bleed duct, work O-ring so it stretches before inserting into female end of tubing and insert O-ring in female end. Lubricate exposed portion with compound. Join male and female ends of bleed duct and secure. After tightening screws on coupling; ensure coupling can rotate and does not bind.

(6) Connect de-icer line to fitting on top of bleed duct tubing. Position lower inlet duct over opening in box and secure with lock type fasteners.

(7) Fasten top part of lower inlet duct to upper inlet duct.

(8) Install fuel and drain fittings in position of bottom of box.

(9) Connect fuel and drain lines to fittings on left and right side of box.

(10) Ensure electrical power is off and connect electrical connections.

(11) Connect air line to right forward corner of box and close all hand hole covers in box.

(12) Install large APU access plate on right side of fuselage. Ensure that splice plates are installed inside of door assembly.

(13) After unit is shut down check for leaks.

(14) Inspect area for presents of foreign objects, security of all attachments.

(15) Close any access previously removed.

8. APU Air Inlet Door Actuator — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Gain access to actuator in tail compartment a top of APU air inlet duct.

(3) Remove electrical connector from actuator.

(4) Remove attaching hardware from door linkage; remove actuator.

B. Installation

(1) If new or replacement actuator is to be installed, before connecting actuator to door linkage, connect electrical Connector and energize essential dc bus with APU master switch off; actuator will run to closed limit if not already there.

(2) Mount actuator to mounting bracket with attaching hardware and connect to door linkage with door closed (flush with skin).

(3) Perform APU Air Inlet Door Actuator — Operational Test, see this Section.

9. **APU Air Inlet Door Actuator — Operational Test**

A. Energize essential dc bus with APU master switch in off.

B. Set APU master switch on, observe that actuator moves door to 0pen position and inlet door actuates door interlock switch when fully open. Door opening takes approximately five seconds.

C. Set APU master switch off, door should go to full closed (flush with skin).

D. De-energize essential dc bus.

10. **APU Increase/Decrease Valve — Removal / Installation**

A. Removal

(1) Gain access to valve (just aft of baggage door frame in baggage compartment).

(2) Disconnect tubing lines from rear of valve.

(3) Remove attaching hardware on bracket, remove valve.

B. Installation

(1) Install valve on mounting bracket with attaching hardware, connect tubing lines.

(2) Start and air load APU.

(3) Check valve operation in both increase and decrease directions, check for appropriate airflow response.

NOTE: If aircraft is equipped with APU AIR FLIGHT - GROUND switch, use GROUND position of switch to check valve operation.

(4) Shut down air and APU.

11. **APU Air Check Valve — Removal / Installation**

A. Removal

(1) Gain access to APU air check valve in tall compartment forward of APU.

(2) Disconnect lower duct clamp and seal.

(3) Remove mount screws at flange.

(4) Remove check valve.

B. Installation

(1) Inspect seal and replace if necessary.

(2) Install check valve on mount using retained hardware.

(3) Inspect duct coupling seal and replace if necessary.

(4) Install duct coupling and seal; tighten coupling.

(5) Start and air load APU and check for leaks.

(6) Inspect area for presents of foreign objects, security of all attachments.

(7) Shut off APU and close access.

12. **APU K1 and K2 Relay — Removal / Installation**

A. Removal

(1) Ensure electrical power is off before removing or installing components.

(2) gain access to relays, under ignition unit on left side of APU.

(3) Remove screws securing relay receptacle plate to mounting bracket.

(4) Withdraw relay assembly from mounting bracket and remove relay involved.

NOTE: Both relays are plug-in type. K1 has four wires and K2 has nine wires to their receptacle.

B. Installation

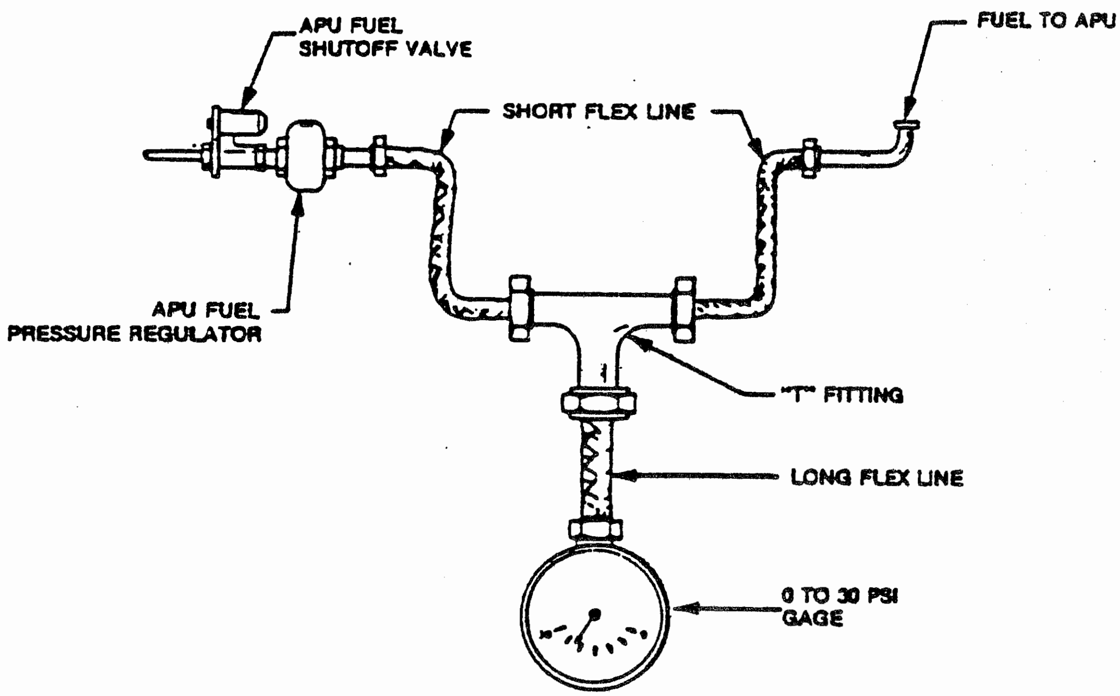
(1) Insert relay and connect Plug.

(2) Install relay assembly into mounting bracket.

(3) Inspect area for presents of foreign objects, security of all attachments.

(4) Close any access previously removed.

(5) Perform APU — Operational Test, see this Section.



APU External Fuel Shutoff Valve and APU Pressure Regulator — Test Gage Installation
Figure 201.

LIST OF EFFECTIVE PAGES

CHAPTER 51

STRUCTURES GENERAL

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** Pages added by Revision 48
*** Pages deleted by Revision 48

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CHAPTER 51

STRUCTURES GENERAL

STRUCTURES GENERAL – DESCRIPTION / OPERATION

1. Description

The Gulfstream airframe is engineered for failsafe structural integrity. The failsafe multi-element design feature will allow the structure surrounding a damaged member to bear the added loads, thus preventing total failure. These alternate load paths and low operating stresses reduce structural fatigue, consequently, less maintenance is required.

A. Doors (Refer to Chapter 52).

The aircraft is equipped with a main entrance (airstair) door, emergency escape hatch, and a baggage compartment door. These doors may be opened from inside or outside the aircraft. The airstair door and the baggage compartment door are provided with inflatable rubber seals to retain fuselage pressurization. The airstair door, located on the left side of the fuselage, features an internally housed folding stairway. Numerous access doors are installed in the exterior surfaces of the fuselage, wings, nacelles, and empennage for ease of service and maintenance.

B. Fuselage (Refer to Chapter 53).

The fuselage is semimonocoque in construction and has an outside diameter of 94 inches and an inside diameter of 88 inches. The basic structure consists of bulkheads, frames, and stringers, with a stressed skin covering.

C. Nacelles (Refer to Chapter 54).

The wing nacelles are permanently attached to the wing structure and provide support for the power plants. The left and right main wheels and gear are housed in the nacelle lower structures. Removable access panels provide for easy access to all components of the power plant.

D. Stabilizers (Refer to Chapter 55).

The stabilizers consist of the fin and left and right horizontal stabilizers. The fin and horizontal stabilizers are equipped with flush type de-icer boots.

E. Windows (Refer to Chapter 56).

The V-shaped windshield, direct vision windows, and fixed side windows comprise the cockpit windows. Horizontal elliptical windows are provided in the passenger cabin. Cabin windows No. 2 and 3, on both sides of the fuselage, are used as emergency escape exits.

F. Wing (Refer to Chapter 57).

The full cantilever wing has a span of 78.5 feet and consists of the center section and inner and outer panel box beam structures. The stub center section is continuous through the lower portion of the fuselage midsection with attachments made at the sides of the fuselage. The upper and lower covers of the box beams consist of integrally-stiffened aluminum alloy planks. The inner panel box beam structure forms the integral fuel tank and the outer panel box beam structure carries the bladder type water/methanol tanks.

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52-7-0	2	July 31/87

* Pages revised by Revision 40.
** Pages added by Revision 40.
*** Pages deleted by Revision 40.

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DOORS

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DOORS

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DOORS — DESCRIPTION / OPERATION

1. General

The airstair door provides a means of access for passengers and crew. This door incorporates a folding stairway and handrails which are housed within the fuselage when door is closed. Airstair permits passage in and out of aircraft without use of ground support equipment.

The baggage compartment door can easily be reached from the passenger compartment, and can be used as an emergency escape exit.

On Aircraft not having ASC 197 an overhead escape hatch is located forward of the passenger cabin. Numerous access doors are provided in the exterior surface of the aircraft to facilitate servicing and maintenance. Airstair and baggage compartment doors and overhead escape hatch are designed to be opened from either side.

A door warning system is provided for airstair and baggage compartment doors. If either or both doors are not properly closed and locked, the door warning light on the master caution display panel will come on.

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MAIN ENTRANCE DOOR — DESCRIPTION / OPERATION

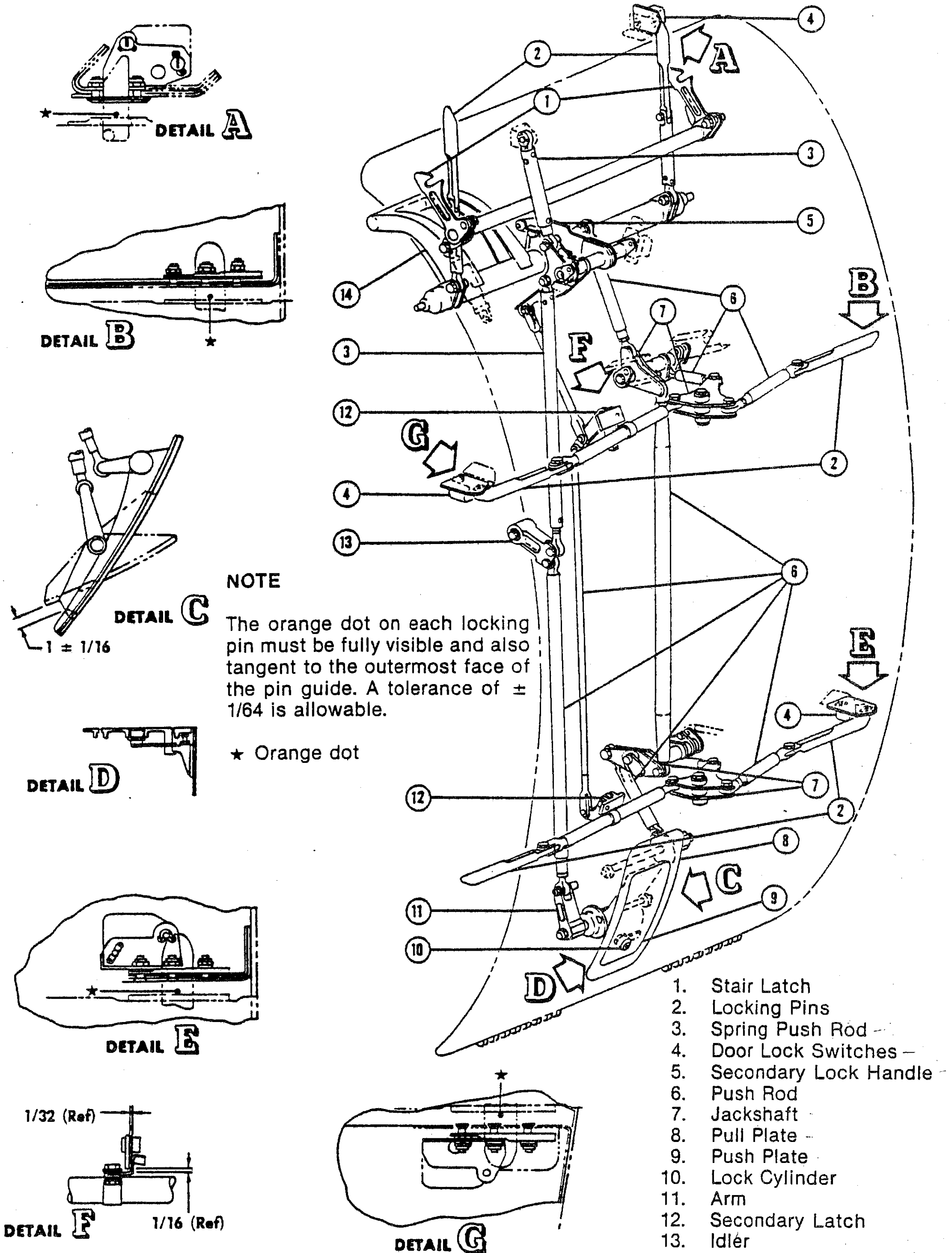
1. **Description**

The airstair door consists of a door structure and attached folding stairway. Door and stairway are extended and retracted by means of an electrically controlled, hydraulically actuated mechanism. The door structure, hinged to the fuselage structure, is of rib and beam construction with a stressed skin covering. Hydraulic pressure to operate the door is supplied by the auxiliary hydraulic system. The auxiliary pump is electrically driven by a motor powered from the main dc bus. Use of batteries as a power source for this bus allows door to be operated independently of ground handling equipment. The door is locked in closed position by six sliding bayonets, which are mechanically linked to manually operated external and internal primary locking handles. (See Figure 1) The lower forward bayonet actuates a valve which controls pressurization and depressurization of the inflatable door seal. Upper forward, lower aft and top aft bayonets actuate microswitches which open and close circuits to the auxiliary hydraulic pump and door warning light in the master caution system. A push bar, attached to the primary mechanism, locks or unlocks the door from inside the aircraft. A flush push-pull handle, located on the outside lower edge of the door, is used for operation of locking mechanism from outside the aircraft. An inner push handle actuates a secondary locking mechanism and outer pull handle actuates the primary locking mechanism.

Two spring loaded latches mechanically lock upper and lower sections of the stairway to each other during the opening cycle. During the powered close cycle these latches are unlocked hydraulically. See Maintenance Practices.

Maximum rotation of door from its fully closed position to point at which it reaches the handrail stops is 137°. Normal rotation of door from its fully closed position to static ground position is 134.5°. Door and stair operation accommodates any combination of strut and/or tire inflation or deflation. The external lock handle latch incorporates a key operated barrel lock for locking of the aircraft. The lock handle latch can be actuated from aircraft interior even though locked from exterior because of a bungee in the linkage. Any one of the bayonets can fail and the door can still withstand cabin pressure. Loss of any one bolt in the lock pin linkage does not affect operation of other pins. If the outside battery switch is left in the ON position, the light marked OUTSIDE BATT SW in the master caution light display panel will come on so condition may be corrected before flight.

CAUTION: UNDER NO CONDITION SHOULD THIS SWITCH REMAIN ON DURING FLIGHT OR ANY TIME THAT THE DOOR SYSTEM IS NOT BEING ACTUATED FROM NOSE WHEEL WELL DOOR SELECTOR SWITCH.



NOTE

The orange dot on each locking pin must be fully visible and also tangent to the outermost face of the pin guide. A tolerance of ± 1/64 is allowable.

★ Orange dot

1. Stair Latch
2. Locking Pins
3. Spring Push Rod
4. Door Lock Switches
5. Secondary Lock Handle
6. Push Rod
7. Jackshaft
8. Pull Plate
9. Push Plate
10. Lock Cylinder
11. Arm
12. Secondary Latch
13. Idler
14. Push Bar

Airstair Door components
Figure 1.

AIRSTAIR DOOR — DESCRIPTION

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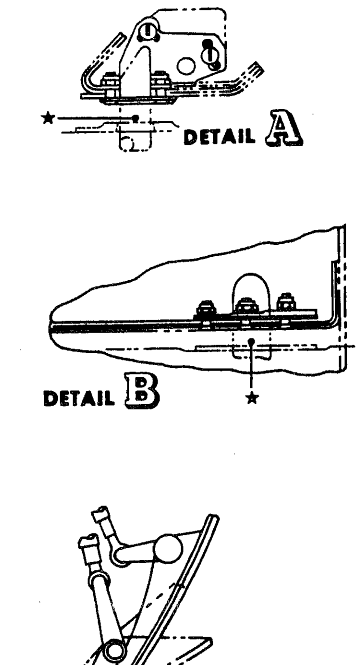

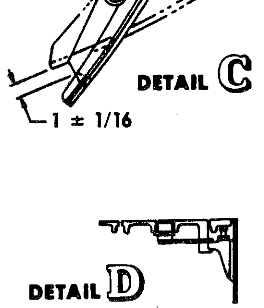

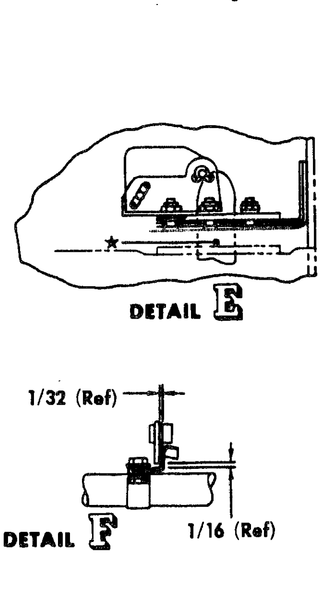

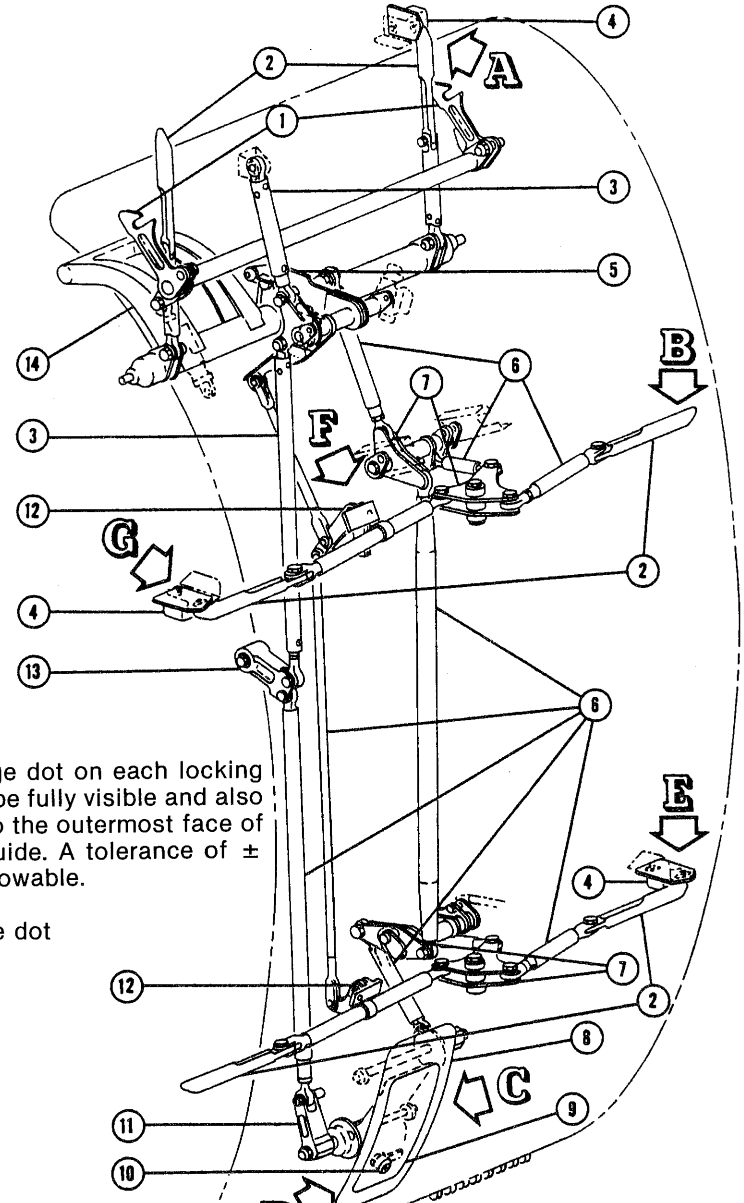

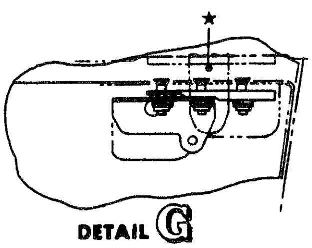

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Airstair Door Components
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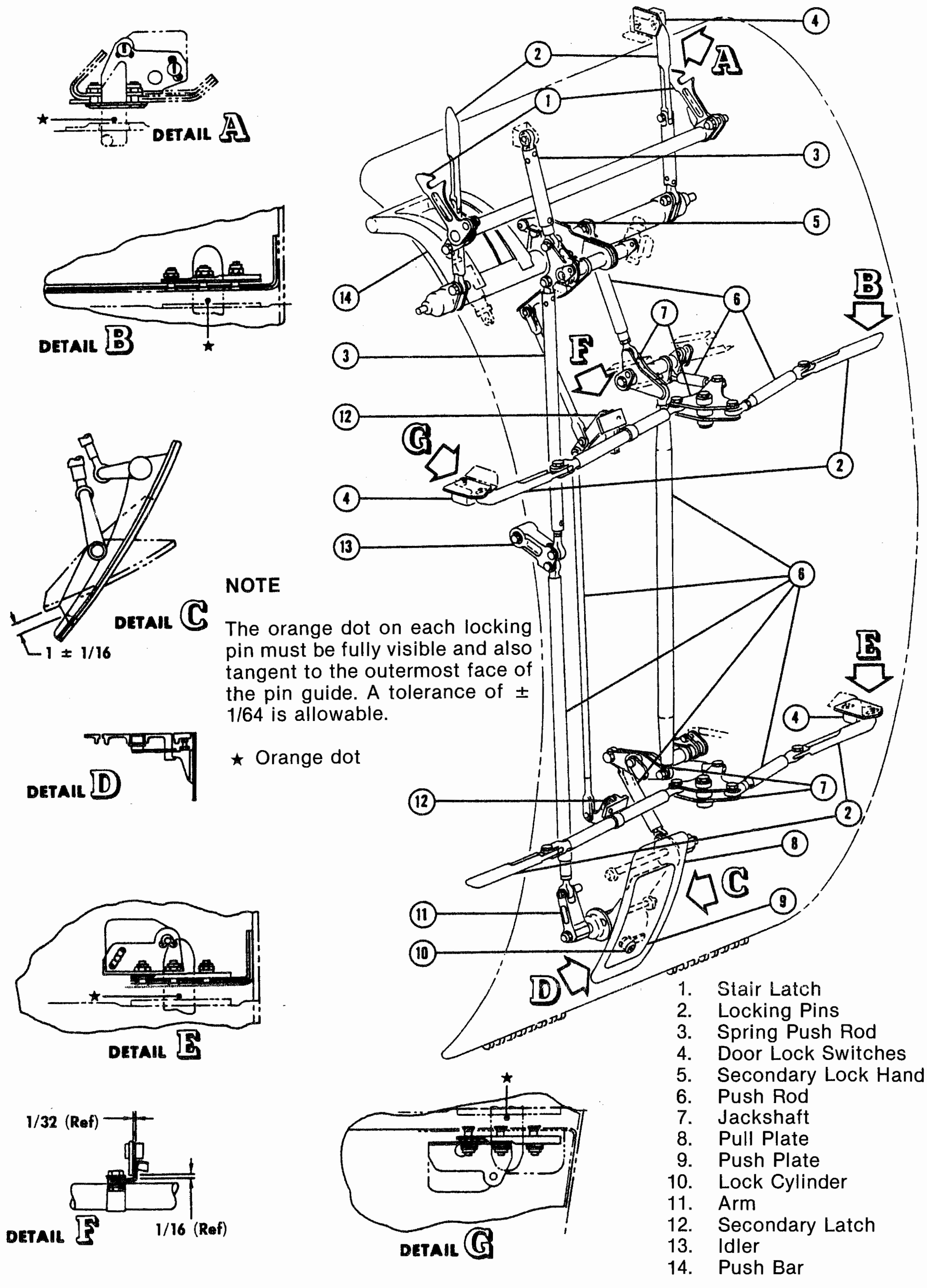

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Airstair Door Components
Figure 1

MAIN ENTRANCE DOOR — MAINTENANCE PRACTICES

1. **Main Entrance Door to Aircraft — Rigging**

NOTE: A clearance of 4 1/2 feet below fuselage center is required to fully extend stair.

A. Place main entrance door in the extended position.

B. Remove main door cable access covers on forward side of door.

C. Install a 5/16 inch rigging pin into forward crank of drive shaft assembly. (See Figure 201).

NOTE: The 5/16 inch bolt that attaches drive shaft crank to rim of large quadrant must be removed to use rig pin hole.

CAUTION: DO NOT OPERATE DOOR WHILE RIG PIN IS INSTALLED.

D. Lock lower sets of steps in unfolded position, and install cable assembly.

E. Tension cables to 100 $\pm$ 5 pounds.

F. Position door so that bottom of stair is 38 1/2 inches below center of fuselage.

G. Adjust pushrod so that jackshaft crank is against stop.

H. Remove rigging pin and install 5/16 inch bolt through drive shaft crank and large quadrant. Install and tighten bolt nut.

CAUTION: DO NOT PERMIT ACTUATING CYLINDER TO BOTTOM.

I. Operate door slowly and check clearances in folded position.

NOTE: Ensure clip portion of 159AM10036-1, -3 clamp assemblies has a 1/8 inch clearance from pushrod cranks when secondary handle is in locked position.

J. To increase step rotation relative to door rotation, increase length of pushrod and screw stop further into position.

2. **Main Entrance Door from the Cockpit — Operation**

CAUTION: NEVER ALLOW THE DOOR TO FREE FALL OPEN IF AIR HAS NOT BEEN BLED FROM HYDRAULIC SYSTEM AFTER SERVICING, OR IF LAST CLOSING OF THE DOOR HAS BEEN MADE MANUALLY. TAG THE OUTSIDE LATCH AFTER CLOSING DOOR MANUALLY TO WARN OTHERS THAT DOOR MUST BE OPENED WITH EXTREME CARE, UTILIZING SUFFICIENT PERSONNEL TO EASE THE DOOR TO THE OPEN POSITION SINCE THE DASHPOT ACTION IS NOT EFFECTIVE AS THE CYLINDER IS WITHOUT FLUID IN THIS SITUATION.

NEVER PLACE ANY WEIGHT ON THE STAIRS UNLESS THE STAIRS ARE SUPPORTED AT THE LOWER END.

A. Energize the main dc bus by setting the battery master switch to NORM.

B. Set door safety switch to UNSAFE.

C. Place main entrance door switch to CLOSE.

D. When door is fully closed, lift secondary latch and pull primary locking handle down. Then release secondary latch.

NOTE: When the door is locked, the solenoid held toggle switch will automatically return to the OFF position.

E. Place door safety switch to SAFE.

F. To open, lift secondary latch.

G. Push primary locking handle until roller falls into second detent.

H. Push door outward. Door will free fall to open position.

3. **Open Door — From Outside**

CAUTION: THE CAUTIONS IN ABOVE PARAGRAPH APPLY WHEN OPENING THE DOOR FROM OUTSIDE.

A. Depress external secondary latch.

B. Pull out primary latch until roller falls into unlock detent.

C. Pull door outward, door will free fall to open position.

4. **Closing the Door — Manually**

CAUTION: THE FOLLOWING PROCEDURE IS TO BE USED ONLY WHEN ABSOLUTELY NECESSARY AND IS NOT A NORMAL OPERATING MODE. EXTREME CARE SHOULD BE EXERCISED WHEN CLOSING THE DOOR BY THIS METHOD.

DO NOT RAISE THE LOWER FOUR STEPS MORE THAN TEN INCHES ABOVE THE NORMAL STATIC GROUND POSITION WITHOUT FIRST RELEASING THE HOOKS AT THE LOWER STAIR. OTHERWISE, THE SELECTOR CABLES MAY BE OVERLOADED AND FAIL.

TAG THE OUTSIDE LATCH BEFORE CLOSING THE DOOR MANUALLY TO WARN OTHERS THAT THE DOOR MUST BE OPENED WITH EXTREME CARE. UTILIZE SUFFICIENT PERSONNEL TO EASE THE DOOR TO THE OPEN POSITION. THE DASHPOT ACTION IS NOT EFFECTIVE AS THE CYLINDER IS WITHOUT FLUID IN THIS SITUATION.

A. Release the hooks at the lower fold joint of the lower stair.

B. Using sufficient personnel, raise main entrance door to closed.

NOTE: As the door is being raised, the lower set of stairs and the handrails will fold automatically.

5. **Main Entrance Door Cables — Tension Check**

NOTE: A clearance of 4 1/2 feet below fuselage center is required to fully extend stair.

A. Place main entrance door in the extended position.

B. Remove main door cable access covers on forward side of door.

C. Remove long pushrod on forward side of door from pushrod bellcrank and step and handrail folding mechanism bellcrank.

D. Remove 5/16 inch bolt that attaches the forward crank of drive shaft assembly to rim of large quadrant.

CAUTION: DO NOT OPERATE DOOR WHILE RIG PIN IS INSTALLED.

E. Install a 5/16 inch rig pin into forward crank of drive shaft assembly.

F. Tension cables to 100 $\pm$ 5 pounds.

G. Remove rig pin and install 5/16 inch bolt through drive shaft crank and large quadrant. Install and tighten nut.

H. Connect pushrod to pushrod bellcrank and step and handrail folding mechanism bellcrank, ensure jack shaft crank is against stop.

I. Operate door slowly and check clearances in folded position.

J. Install covers.

6. **Main Entrance Door — Removal / Installation**

CAUTION: IF AIRCRAFT IS ON JACKS, USE EXTRA CARE WHEN REMOVING DOOR. IF IT IS ALLOWED TO SWING FREE, IT WILL STRIKE AIRCRAFT.

A. Removal

(1) Remove bolts from handrail pivot bearings at bulkheads.

(2) Remove upper attach bolt from fwd pushrod and bellcrank.

(3) Remove upper attach bolt from aft pushrod and door actuator bellcrank.

(4) Remove fwd swivel cover and disconnect air and hydraulic lines. Cap open lines.

(5) Remove hinge pin between upper step and fuselage, fold step to gain access to door hinge pins.

CAUTION: LOWER STEPS ARE NOT TO BE UNLATCHED UNTIL THE DOOR IS FULLY SUPPORTED ON A SUITABLE SUPPORT. FAILURE TO PROPERLY SUPPORT THE DOOR WILL RESULT IN EXTENSIVE DAMAGE TO THE FUSELAGE WHEN THE STEPS UNLOCK.

(6) Unlock lower stair half and fold it back upon the door.

(7) Lift door to an approximately horizontal level, position a suitable cradle underneath and let door recline into it.

(8) Remove door hinge pins. Inspect pins for indications of wear. If wear is evident, both pins should be replaced.

B. Installation

(1) Position door so hinge segments of door align with fuselage hinge segments. Install hinge pins.

(2) Unfold and lock lower step section. Cradle can now be removed.

(3) Install hinge pin between upper step and fuselage.

(4) Connect air and hydraulic lines.

(5) Attach aft pushrod to door actuator bellcrank.

(6) Attach fwd pushrod to bellcrank assembly.

(7) Attach handrails to pivot bearings on bulkheads.

(8) Operate door slowly through one cycle and check for proper clearance and for hydraulic leak.

(9) Install swivel cover.

7. Main Entrance Door — Inspection

A. Procedure

(1) Remove main door fairings (fwd and aft) cable / sector covers (fwd and aft) and insulation packages.

B. Inspect Main Entrance Door for:

(1) Condition of structural members.

(2) Condition of upper and lower swivels, downlatch actuator, associated hydraulic lines and connections.

(3) Condition of electrical wire runs and connections.

(4) Condition of inflatable seal.

(5) Condition and proper operation of door, door handle, crank and bayonets.

(6) Condition of main door bellcrank.

(7) Inspect cables for worn or broken strands, especially in area where cable passes over edges of flat spot on crank sector. Replace cables with worn or broken strands. Inspect airstair torque tube.

(8) Visually inspect insulation packages for water intrusion.

(9) Visually inspect for corrosion. Treat for any corrosion found.

NOTE: Engineering evaluation is required whenever corrosion has removed more than 20% of the original material thickness over a distance equal to the flange width if it occurs on flange or one inch if it occurs on a web.

(10) Install (dry) insulation packages.

C. Perform Door Cable Tension Check Using The Following Procedures:

(1) Remove pushrod (long rod on forward side of door).

(2) Unlock lower set of steps from the down position.

(3) Remove 5/16 inch bolt that attaches drive shaft crank to rim of large quadrant to use as rig pin hole.

(4) Raise the lower set of steps until rig pin holes line up.

(5) Install 5/16 inch rig pin into crank of drive shaft assembly until rig pin protrudes into rig pin hole of upper set of steps.

(6) Lock lower set of steps in the unfolded position.

(7) Tension cables to 100 $\pm$ 5 pounds.

(8) Unlock lower set of steps from the down position.

(9) Remove rig pin and install 5/16 inch bolt through drive shaft and large quadrant.

(10) Install and tighten nut on bolt.

(11) Install pushrod, adjustment of this rod and screw stop may be necessary so lower steps will lock in down position.

(12) Operate door slowly and check clearances in the folded position. A clearance of 4 1/2 feet below fuselage center is required to fully extend stair.

(13) Inspect for presence of foreign objects then install door fairings and cable / sector covers (fwd and aft).

D. Inspect Main Entrance Door For Proper Fit As Follows:

(1) From inside aircraft with door latched, grasp handle and alternately apply body weight in inboard and outboard direction at top center of door. If door does not exhibit movement or audible sounds which can be attributed to freedom of movement between latching pins (bayonets) and their retainers, the door is acceptable.

(2) If door does exhibit movement, inspect per (3) and (4) below.

(3) With door in latched position with cabin pressure differential at zero, measure and record step between outer skin of door and adjacent fuselage while applying outboard load as in (1) above. Measure step at door top centerline.

(4) Apply an inboard load as noted in (1) above. Measure this step at the same location as in (3) above and compare readings. If difference in measurements exceeds 0.040 inch, door has abnormal displacement and bayonets and striker plates should be inspected to determine if replacement is necessary.

8. Main Entrance Door Hinge Pins — Inspection

NOTE: It is recommended that operators stock a set of replacement pins before starting this procedure.

CAUTION: REMOVE HINGE PINS ONE AT A TIME. KEEP WEIGHT OFF OF STEPS WHILE PINS ARE REMOVED.

A. Lower entrance door to fully down and locked position.

B. Pull hinge pin out of lower hinge on top step and fold step and riser up.

C. Remove metal safety locks that secure the hinge pins, retain hardware.

D. Remove air stair door hinge pins one at a time.

E. Inspect for visible indications of wear. If there are visible indications of wear (i.e. "Crankshafting") replace both hinge pins.

NOTE: Standard pin diameter is 0.242 - 0.246 inch with hinge hole diameter 0.250 - 0.254 inch.

Oversize pins (diameter 0.2515 - 0.2520 inch) are available for holes worn to at least 0.254 inch but less than 0.264 inch. Door hinge halves may be reamed to 0.254 inch to accept oversize pins P/N 1159RDB398-1.

F. Install hinge pin.

G. Repeat Step E. above for remaining hinge pin.

H. Install hinge pin safety locks with hardware that was retained from Step C. above.

I. Lower step and riser and install hinge pin that was removed in Step B. above.

9. Main Entrance / Baggage Door — Operational Test

WARNING: NEVER ALLOW THE DOOR TO FREE FALL OPEN IF AIR HAS NOT BEEN BLED FROM HYDRAULIC SYSTEM AFTER SERVICING, OR IF LAST CLOSING OF DOOR WAS MADE MANUALLY. TAG PULL PLATE AFTER CLOSING D OR MANUALLY, TO WARN OTHERS THAT DOOR MUST BE OPENED WITH EXTREME CARE UTILIZING SUFFICIENT PERSONNEL TO EASE THE DOOR TO THE OPEN POSITION SIN E DASH POT ACTION IS NOT EFFECTIVE AS THE CYLINDER IS WITHOUT FLUID IN THIS CASE.

CAUTION: WHEN OPENING THE ENTRANCE DOOR, THE ROLLER ON THE PUSH BAR ASSEMBLY MUST ALWAYS BE ROTATED INTO THE SECOND DETENT OF THE SECONDARY LOCK BEFORE THE DOOR IS PERMITTED TO SWING DOWN.

DO NOT PLACE ANY WEIGHT ON THE STAIRS UNLESS THE STAIRS ARE SUPPORTED AT THE LOWER END. NEVER LEAVE THE OUTSIDE BATTERY SWITCH IN THE ON POSITION UNLESS ACTUATING THE DOOR SYSTEM USING THE NOSE WHEEL WELL OUTSIDE DOOR SWITCH.

NOTE: If aircraft has DOOR ISOLATION switches installed, place them in NORM (guard down-toggle down position) before starting this test. This test starts with entrance and baggage doors open.

A. Energize main and essential dc buses. CABIN OPEN lights in master panel should be on.

B. Supply a pressure of 15 psi or greater in de-icing system pressure manifold (as shown on pressure gage in cockpit) by connecting air pressure to pneumatic de-icing system upstream of regulator in either nacelle or operating APU with full air output to cabin and cockpit.

C. Close and lock baggage door. Do not insert pip pin in secondary lock handle. Observe that door seal inflates when bayonets extend. CABIN OPEN light (master warning lights panel) should remain on.

D. Check that bayonets are fully extended by observing that orange dot on each pin is fully visible tangent to outermost face of pin guide plate. A tolerance of $\pm$ 1/64 inch is allowable.

E. Close entrance door using AIRSTAIR switch (pilots circuit breaker panel). Lock door. CABIN OPEN light will still remain on. Place AIRSTAIR switch OFF. Observe that entrance door seal inflates when bayonets are extended. Check that all bayonets are fully extended by observing that orange dot on each bayonet is visible in same manner as those on baggage door.

F. Place DOOR SAFETY switch (lower overhead panel) to SAFE. CABIN OPEN lights should go out.

G. Slightly lift entrance door secondary lock handle. CABIN OPEN light should come on. **Do not open door.**.

H. Place entrance door secondary lock handle to lock position. CABIN OPEN light should go out.

I. Place pip pin in baggage door secondary lock handle. CABIN OPEN light should come on.

J. Remove and stow baggage door secondary lock handle pip pin. CABIN OPEN light should go out.

K. Open baggage door. Observe that seal deflates. CABIN OPEN light should come on during first movement of secondary lock handle and remain on.

NOTE: Steps L - Q apply only to Aircraft 81-200, 322 and 323 and those Aircraft having ASC 110 (Door Isolation Switches). Aircraft 1-80 and 114 not having ASC 110 continue with Step R. to end of test.

L. In cockpit, place BAGGAGE DOOR ISOLATION switch (outboard side of Pilots Station 133 relay panel) to guard-up, toggle-up position. CABIN OPEN lights should go out.

M. Return BAGGAGE DOOR ISOLATION switch to NORM, guard-down, toggle-down position. CABIN OPEN light should come on.

N. Close and lock baggage door. **Do not insert pip pin into secondary lock handle**. CABIN OPEN light should go out.

O. Place DOOR SAFETY switch to UNSAFE. Open entrance door using AIRSTAIR switch. Observe that seal deflates when bayonets are retracted. CABIN OPEN light should come on when secondary lock handle is moved and should remain on.

P. Place DOOR SAFETY switch to SAFE. Place MAIN DOOR ISOLATION switch to guard up, toggle-up position. CABIN OPEN light should go out.

Q. Return MAIN DOOR ISOLATION switch to guard-down, toggle-down position. CABIN LIGHT should come on.

R. Place DOOR SAFETY switch to UNSAFE. Turn off APU or external air switch (if used), BATT NORM / EMER switch and EXT PWR switch (if used).

S. Open outside battery switch cover, left nacelle and turn on outside battery switch.

NOTE: Left battery must be connected.

T. Close entrance door using OUTSIDE DOOR switch in nosewheel well. Lock entrance door using outside lock handle and push-plate. When primary lock handle inserts bayonets, the auxiliary pump will stop. Return OUTSIDE DOOR switch to OFF.

U. Open entrance door by depressing push-plate and pulling lock handle. Allow door to free fall. Ensure that action is normal on door extension and that stair down latch is properly engaged before stair finally rests on ground (note audible engagement.) Observe dampened action of door and stair opening toward latter part of extension.

V. Open baggage door using outside handle. Ensure that handle moves freely with no evidence of binding.

W. Close baggage door using outside handle and ensure proper operation.

X. Turn off outside battery switch and close cover plate.

Y. Return system to normal configuration and position doors as desired.

10. Main Entrance Door Locking Mechanism - Rigging

(See Figure 201)

A. Install shaft (13) and primary handle (14) into door.

NOTE: Lock cylinder (17) must be in the locked position and the stop (18) must be engaged.

B. Position handle flush to contour.

C. Engage inner handle secondary lock (6) on handle assembly (6).

NOTE: This is the door locked position.

D. Place 1/4 inch rigging pin through upper jackshaft assembly (7) and rigging hole.

E. Install push rod (12).

F. Place 1/4 inch rigging pin through jackshaft assembly (8) and rigging hole.

G. Install push rod (19).

H. Adjust push rod (9) to a length of 6 15/16 inches, center-to-center.

I. Install and fasten crank (16) onto upper shaft (13) with handle (15) flush to contour.

J. Install crank (11) onto lower shaft (13) with spacer tube (10) against frame flange and hold 1 inch $\pm$ 1/16 inch dimension between handles.

K. Aircraft having ASC 112, adjust push rod (20) to provide positive detenting of secondary latch from inside and outside. When adjustment is complete safety-wire adjusting nut to push rod.

L. Remove rigging pins installed in Steps D. and F. above.

M. Visually check bayonets in extended position for orange dot, which must be completely visible.

11. Main Entrance Door from Nose Wheel Well — Operation

CAUTION: AFTER MAIN ENTRANCE DOOR HAS BEEN CLOSED AND LOCKED, THE OUTSIDE BATTERY SWITCH SHOULD BE PLACED IN OFF POSITION TO PREVENT A CONTINUAL DRAIN ON THE BATTERY.

A. Energize main dc bus by placing outside battery switch to ON position.

B. The door safety switch must be in UNSAFE position.

C. Place main entrance door switch, (located on the nose wheel well J-box to CLOSE position.

D. When door is closed, push primary locking handle down. This engages secondary latch. (Secondary latch bar should be flush with door skin.)

NOTE: When the door is locked, the solenoid held toggle switch will automatically return to the OFF position.

E. Set outside battery switch to OFF and close switch access cover.

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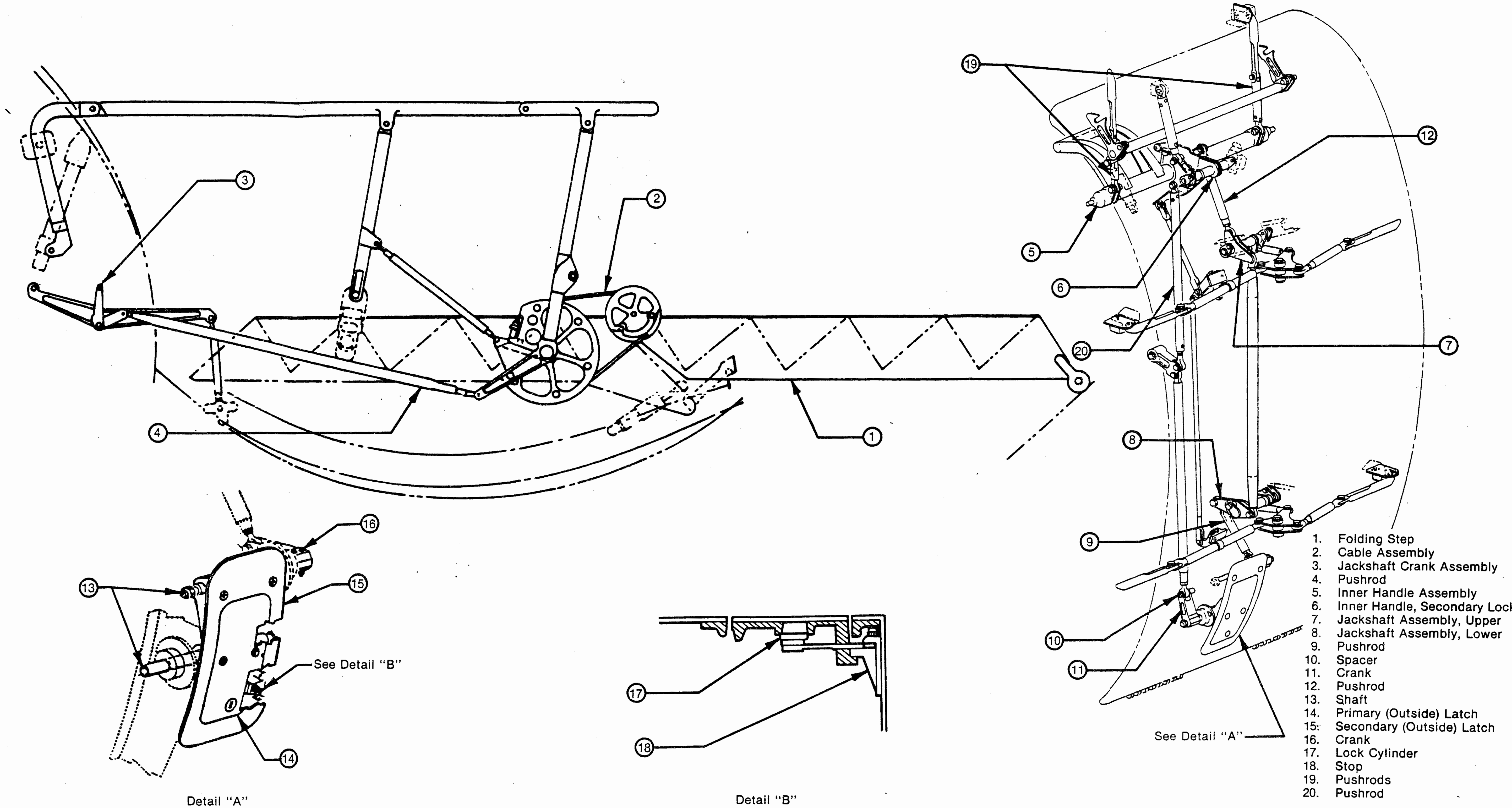


Main Entrance Door Installation Diagram
Figure 201.

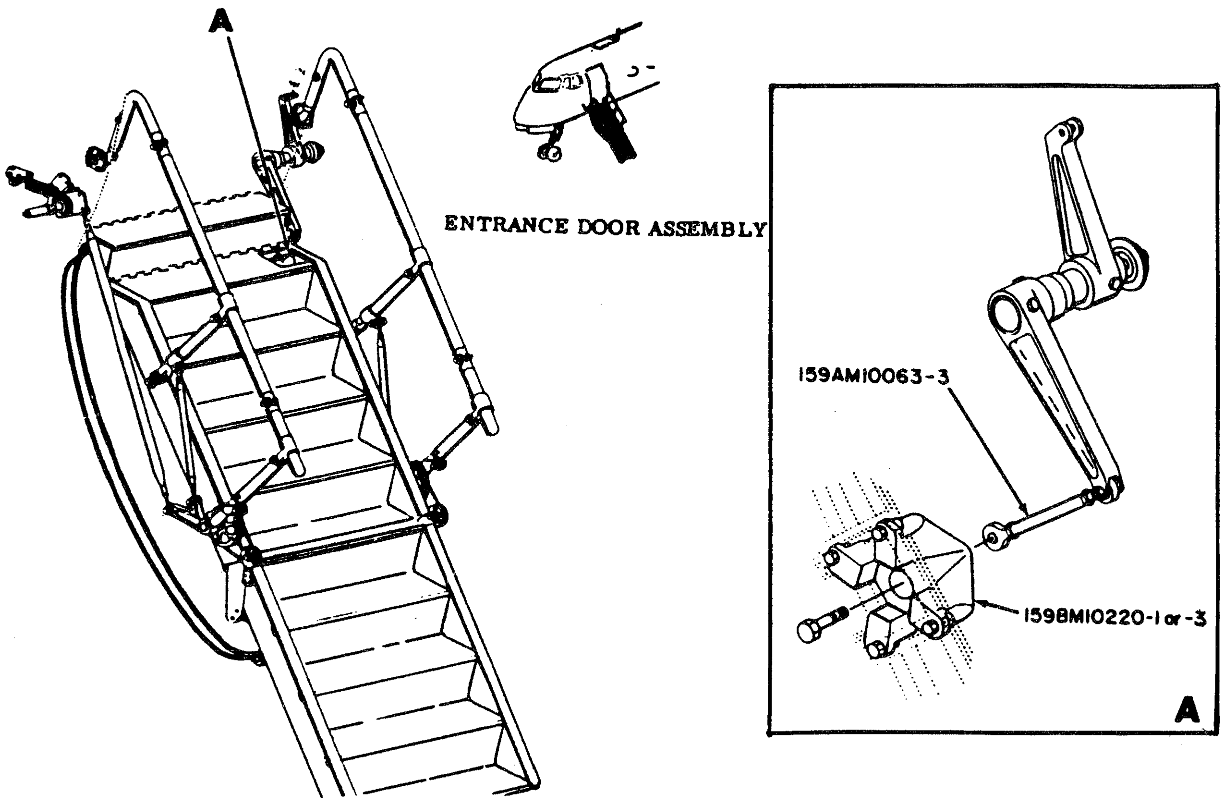
AIRSTAIR DOOR — MAINTENANCE PRACTICES

1. **Inspection**
   A. Inspect stairway, door, handrails and all attachment fittings for cleanliness, security of attachment and general condition.
      (1) Ensure that two spring loaded latches are locking upper and lower halves of stairway in extended position.
      (2) Inspect all hydraulic lines and units for signs of leakage or obvious damage.
      (3) Check six locking pins for burrs, contortions and cleanliness.
      (4) Check lower end of stairway for contact with ground.

2. **Main Entrance Door Fitting — Inspection**

   NOTE: This inspection not required if part number of fitting is 159BM10220-5.

   A. Inspect main entrance door fitting as follows: (see Figure 201). This inspection complies with Customer Bulletin 298.
      (1) Visually inspect door fitting (P/N 159BM10220-1 or -3) for cracking. Replace cracked fittings.
      (2) Visually inspect push rod (P/N 159AM10063-3) for movement or play in rivets attaching end fittings to tube. Replace push rod if movement is noticeable.



Main Entrance Door Fitting — Inspection
Figure 201

AIRSTAIR DOOR HYDRAULIC SYSTEM — DESCRIPTION / OPERATION

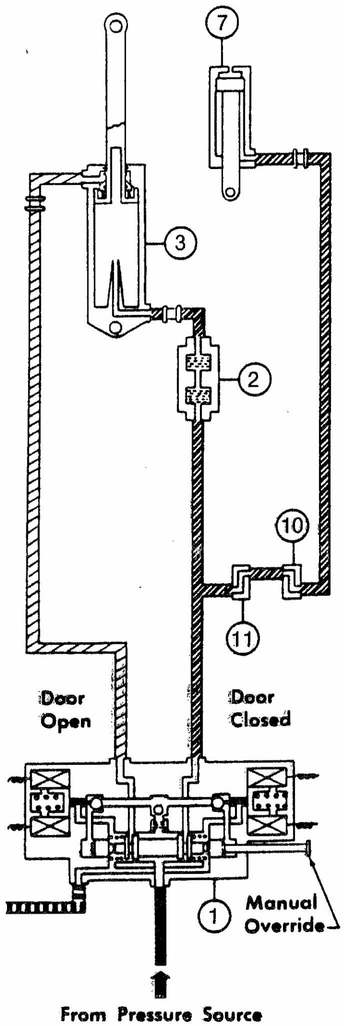
1. **Description** (See Figures 1 and 2)

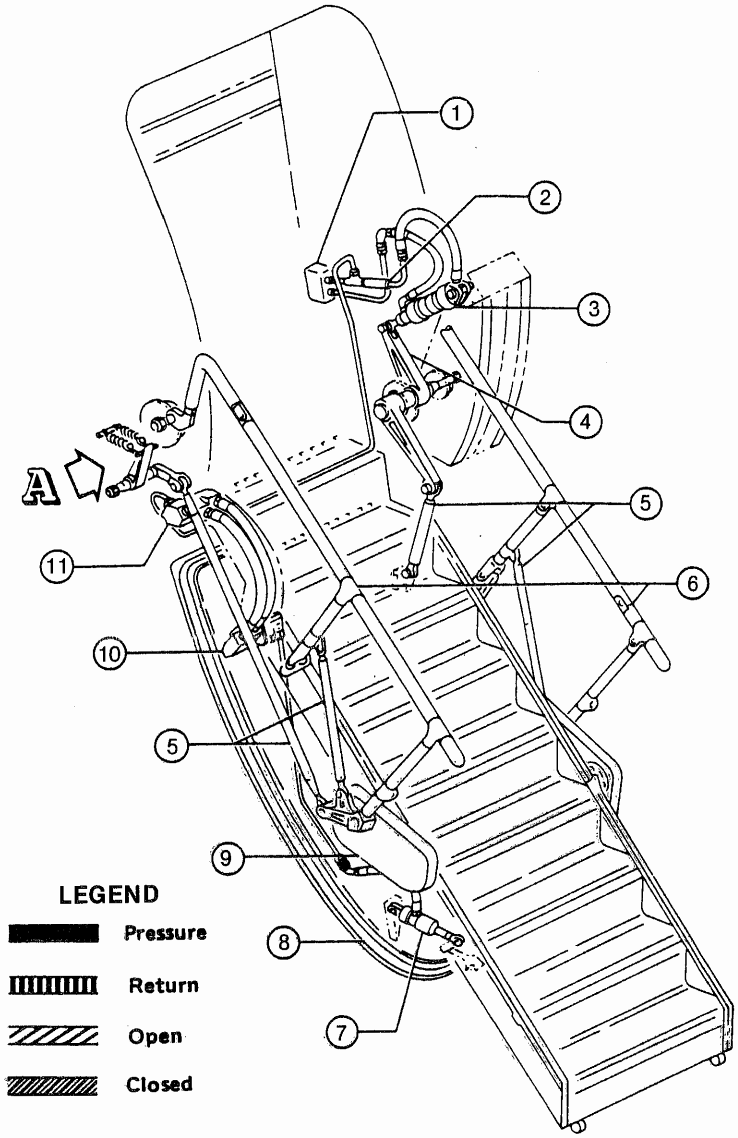
The airstair door hydraulic system consists of a door and stairs selector valve, floor swivel, door swivel, one restrictor, door and stair cylinder, and down latch release cylinder. The system is supplied operating pressure by the auxiliary hydraulic system. The door and stairs selector valve, located aft of the bulkhead at Fuselage Station 169, directs hydraulic pressure to the rod end of the door and stair cylinder when the door is to be opened, and to the head end when the door is to be closed. Floor and door swivels, located at the door hinge, allow hydraulic pressure to be directed to down latch release cylinder and air pressure to door seal.

2. **Operation**

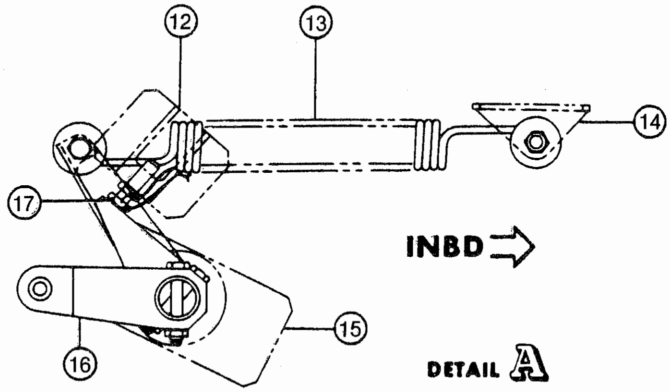
When tracing flow paths on the schematic flow diagram, auxiliary hydraulic system pressure (when made available to pressure port of door and stair selector valve) flows through the valve, past the selector spool, through the filter, past the ball check and to the door open or closed pressure seats. Setting either door control switch to OPEN energizes the solenoid on the right side of the valve. When solenoid plunger is retracted, the poppet leaves the pressure seat and closes off the return. Hydraulic fluid then is permitted to enter right of the spool, shifting the spool left. Hydraulic fluid then flows out of the door OPEN port and into the rod end of the door and stairs cylinder. The cylinder then retracts opening the door and unfolding the stairs. Return pressure flows from the head end of the cylinder, through a restrictor and to the door CLOSED port of the selector valve. Return fluid then flows past the selector spool to the return port of the valve and back to the reservoir.

Setting either door control switch to CLOSE energizes the solenoid on the left side. When solenoid retracts, the poppet will leave the pressure seat and close off the return. Hydraulic pressure then is permitted to enter left of the spool, thus shifting the spool right. Hydraulic pressure then flows out of the door close port into the head end of the door and stairs cylinder and to the down latch release cylinder through the door and stair swivels. The down latch release cylinder then retracts, removing locking hooks, allowing door and stairs cylinder to fold stairs and close the door.

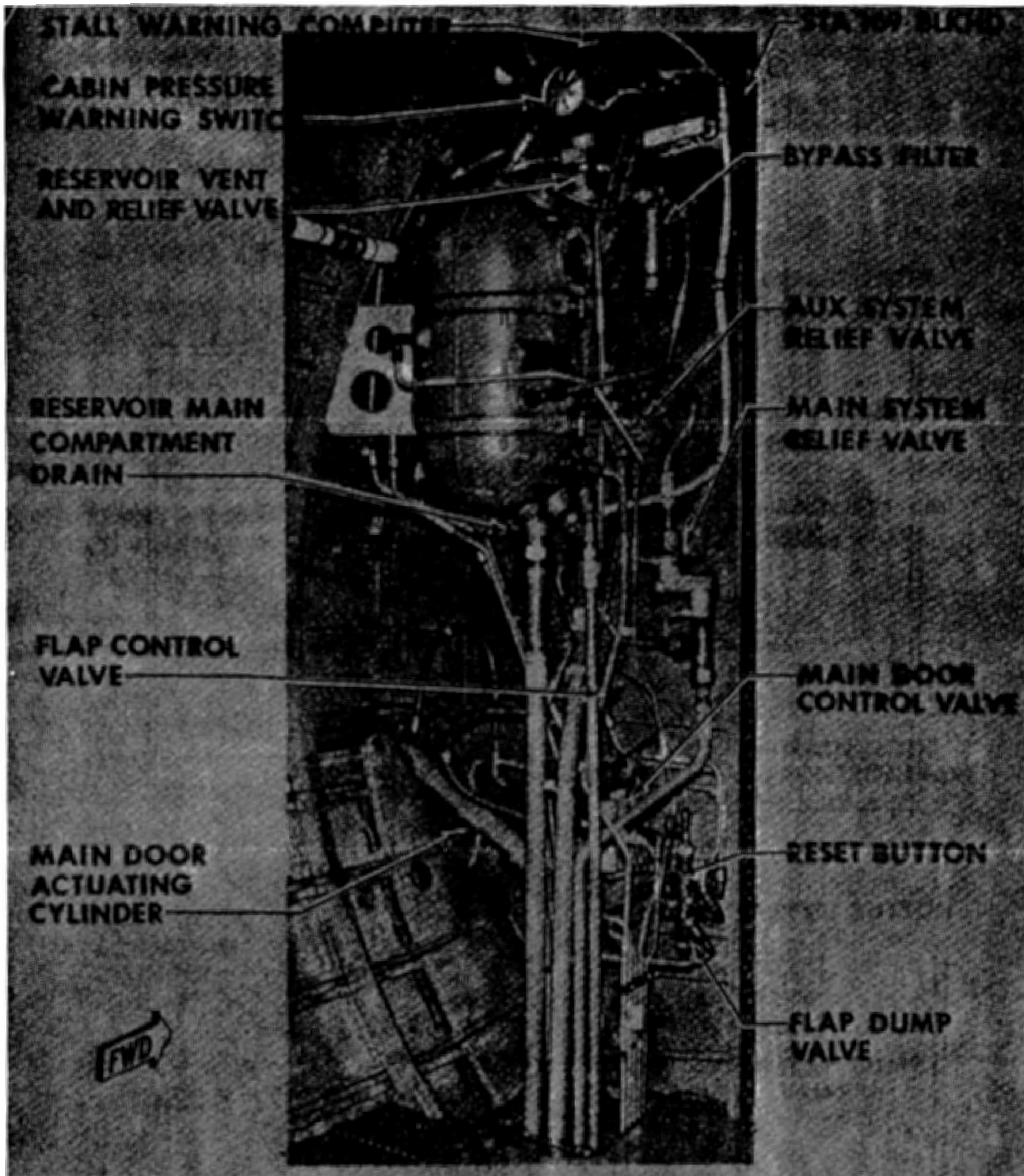




1. Door and Stair Selector Valve
2. Restrictor
3. Door and Stair Cylinder
4. Jackshaft
5. Push Rods
6. Handrails
7. Down Latch Cylinder
8. Inflatable Door Seal
9. Sector Cable Housing
10. Door Swivel
11. Floor Swivel
12. Push Rod Stop Fitting
13. Overtravel Spring
14. Overtravel Spring Fitting
15. Jackshaft Bearing
16. Jackshaft Assembly
17. Adjusting Bolt



Airstair Door Hydraulic System
Figure 1



Hydraulics Compartment
Figure 2

AIRSTAIR DOOR HYDRAULIC SYSTEM — MAINTENANCE PRACTICES

1. **Airstair Door to Aircraft — Rigging** (See Figure 201)
   A. Install a 5/16-inch rigging pin into forward crank of drive shaft assembly (159AM10052-1).
   B. Lock folding Step (1) in unfolded position and install cable assembly (2).
   C. Tension cables 100 ± 5 pounds.
   D. Position door so that bottom of stair is 38-1/2 inches below center of fuselage (WL 14-21/32).
   E. Adjust push rod (4) so that jack shaft crank (3) is against stop.
   F. Operate door slowly and check clearances in folded position.
   G. To increase step rotation relative to door, increase length of push rod and screw stop further into fitting.

   CAUTION: DO NOT PERMIT ACTUATING CYLINDER TO BOTTOM.

   NOTE: A clearance of 4-1/2 feet below fuselage center is required to fully extend the stair.

2. **Airstair Door Locking Mechanism — Rigging** (See Figure 201)
   A. Install shaft (13) and primary handle (14) into door.

   NOTE: Lock cylinder (17) must be in the locked position and the stop (18) must be engaged.

   B. Position handle flush to contour.
   C. Engage inner handle secondary lock (6) on handle assembly (6).

   NOTE: This is the door locked position.

   D. Place 1/4-inch rigging pin through upper jackshaft assembly (7) and rigging hole.
   E. Install push rod (12).
   F. Place 1/4-inch rigging pin through jackshaft assembly (8) and rigging hole.
   G. Install push rod (19).
   H. Adjust push rod (9) to a length of 6-15/16 inches, center-to-center.
   I. Install and fasten crank (16) onto upper shaft (13) with handle (15) flush to contour.
   J. Install crank (11) onto lower shaft (13) with spacer tube (10) against frame flange and hold 1 inch ± 1/16-inch dimension between handles.
   K. Aircraft having ASC 112, adjust push rod (20) to provide positive detenting of secondary latch from inside and outside. When adjustment is complete safety-wire adjusting nut to push rod.
   L. Remove rigging pins installed in Steps D. and F. above.
   M. Visually check bayonets in extended position for orange dot, which must be completely visible.

3. **Airstair Door Actuating Cylinder — Adjustment**

   NOTE: To ensure that door operating lever is not overloaded in closed position through the action of manually engaging locking bayonets, adjust length of door actuating cylinder as follows:

   A. Open airstair door and disconnect terminal end of actuating cylinder from jackshaft arm.
   B. Manually place the door in the closed and locked position.
   C. Hydraulically extend actuating cylinder to its full length. Adjust rod end terminal by screwing it in or out so that length of the cylinder is 1/16-inch longer than required to align terminal with corresponding hole in the jackshaft arm when door is closed and locked. After adjustment, tighten terminal locking nut and key. Secure with safety-wire.

   NOTE: Door actuator may require additional adjustment to obtain proper snubbing action. If door slams upon initial retraction, shorten door actuator rod end as required to obtain proper snubbing.

   D. Reconnect cylinder terminal to jackshaft arm. Connection is facilitated if door is open. Observe all precautions when opening door.

4. **Entrance Door Actuating Cylinder — Removal / Installation**
   A. Removal (See Figure 202)
      (1) Gain access to door actuating cylinder in hydraulics compartment, aft of bulkhead at door entrance.
      (2) With door open, disconnect two hydraulic lines to actuating cylinder. Cap lines and fittings.
      (3) Disconnect terminal end of cylinder and retain hardware.
      (4) Using sufficient manpower, manually close and lock entrance door.

WARNING: WITH CYLINDER DISCONNECTED, DOOR HAS NO RESTRAINT AND NO ATTEMPT SHOULD BE MADE TO OPEN OR CLOSE THE DOOR UNLESS SUFFICIENT PERSONNEL ARE AVAILABLE.

(5) Disconnect head end of cylinder and retain hardware.
(6) Remove cylinder.

B. Installation

NOTE: If cylinder is to be replaced, adjustment of the terminal end must be made as specified in the following procedure. If same cylinder is to be installed and no change or adjustment of the terminal end has been made, disregard terminal end adjustment, (Step (3) below).

(1) Position actuating cylinder with piston retracted and secure head end to structure with hardware previously removed. Do not connect terminal end at this time.

WARNING: WITH CYLINDER DISCONNECTED THE DOOR HAS NO RESTRAINT. NO ATTEMPT SHOULD BE MADE TO OPEN OR CLOSE THE DOOR UNLESS SUFFICIENT PERSONNEL ARE AVAILABLE.

(2) Remove protective caps and connect hydraulic lines to cylinder.

CAUTION: WHILE EXTENDING CYLINDER PISTON ENSURE THAT IT CLEARS ALL TUBING, ETC; IN HYDRAULIC COMPARTMENT. APPLY HYDRAULIC PRESSURE AS SLOWLY AS POSSIBLE.

(3) Adjust rod end terminal by screwing it in or out so that length of cylinder is 1/16-inch longer than required to align terminal end with corresponding hole in the bellcrank when the door is closed and locked. After proper adjustment is made, tighten and safety locking nut on terminal.
(4) Manually open the entrance door using sufficient manpower to restrict door movement until fully down.
(5) Connect terminal end by retracting cylinder hydraulically until terminal end aligns with bellcrank. Install hardware previously removed.
(6) Using either inside AIRSTAIR or OUTSIDE DOOR switch and auxiliary pump, raise and lower door several times to bleed any entrapped air from the system, using personnel for door restraint especially on opening.
(7) Check that when door is fully extended, approximately 1/4-inch of piston travel remains at cylinder when handrail stops reach stop fitting. Piston must not bottom.
(8) Leave door in position desired, and close any access previously opened.

5. **Entrance Door Selector Valve — Removal / Installation**

A. Removal (See Figure 202)

(1) Gain access to selector valve in hydraulic compartment aft of entrance door.
(2) Disconnect electrical connector from selector valve.
(3) Disconnect hydraulic lines from selector valve.
(4) Install protective caps on electrical and hydraulic connector, lines and fittings.
(5) Remove selector valve and retain hardware.

B. Installation

(1) Install selector valve with hardware previously removed.
(2) Remove all protective caps from hydraulic lines and ports, and electrical connector.
(3) Connect hydraulic lines and electrical connector to valve.
(4) Energize main and essential dc buses.
(5) Raise and lock entrance door using inside or outside door switch. Check for proper operation.
(6) Unlock and lower entrance door using inside or outside door switch.
(7) Operate several times to eliminate any trapped air. Leave door open. Shut off electrical power.
(8) Replace access panels.

6. **Entrance Door Restrictor — Removal / Installation**
   A. Removal (See Figure 202)
      (1) Gain access to restrictor in hydraulic compartment aft of entrance door.
      (2) Disconnect hydraulic lines and remove restrictor.
      (3) Install protective caps in hydraulic lines.
   B. Installation
      (1) Remove protective caps from hydraulic lines.
      (2) Install restrictor between hydraulic lines.
      (3) Energize main and essential dc buses.
      (4) Place DOOR SAFETY switch in UNSAFE position.
      (5) Raise and lock entrance door using inside airstair switch. Check for leakage and proper door operation.
      (6) Operate door several times to eliminate trapped air.
      (7) Return door to position desired and turn off electrical power.
      (8) Close access.

7. **Entrance Door Downlatch Cylinder — Removal / Installation**

WARNING: AFTER CHANGING ANY HYDRAULIC COMPONENT IN THE ENTRANCE DOOR SYSTEM ENSURE THAT HYDRAULIC SYSTEM IS COMPLETELY BLED OF AIR BEFORE ALLOWING DOOR TO FREE FALL OPEN.

   A. Removal (See Figure 202)
      (1) With entrance door in down position, use a hammer handle or other firm piece of wood to unlatch two latches at lower stair.
      (2) Fold lower stair over upper stair.
      (3) Place a padded stand under door to support entire assembly.
      (4) Disconnect hydraulic line at downlatch cylinder. Install protective caps on open port and line.
      (5) Remove cylinder. Retain hardware.

      NOTE: Measure extended length of cylinder after removal if being replaced.

   B. Installation
      (1) Adjust replacement downlatch cylinder to exact extended length measured on removed cylinder.
      (2) Install cylinder. Use hardware previously removed.
      (3) Remove protective caps and install hydraulic line.
      (4) Remove padded stand supporting door.
      (5) Carefully unfold lower stair and engage latches.
      (6) Energize main and essential dc buses.
      (7) Using OUTSIDE DOOR switch begin to close door. When lower stair is folded over upper stair, release switch. Check downlatch cylinder and hydraulic line for fluid leakage.
      (8) After leakage check, operate OUTSIDE DOOR switch to unfold and latch lower stair and fully extend door. Check for audible latch engagement.
      (9) Remove electrical power.

8. **Entrance Door and Floor Swivels — Removal / Installation**

WARNING: AFTER CHANGING ANY HYDRAULIC COMPONENT IN THE ENTRANCE DOOR SYSTEM ENSURE HYDRAULIC SYSTEM IS COMPLETELY BLED OF AIR BEFORE ALLOWING DOOR TO FREE FALL OPEN.

   A. Removal (See Figure 202)
      (1) Unlatch and lower entrance door.
      (2) Remove fairing cover over door swivels at top forward end of entrance door.
      (3) Disconnect hydraulic and pneumatic lines from swivel.

      NOTE: The upper swivel is the floor swivel while the lower swivel is the door swivel.

      (4) Install protective caps on lines and ports.

(5) Remove swivel. Retain hardware.

B. Installation

(1) Install swivel using hardware previously removed.
(2) Remove protective caps from lines and fittings.
(3) Install hydraulic and pneumatic lines on swivel.
(4) Energize main and essential dc buses.
(5) Provide at least 15 psi of pressure in the de-icing system pressure manifold (as read on the cockpit gage) by either connecting an air supply to pressure manifold in nacelle or operating the APU with air on and set cabin and cockpit temperature controls to full cooling.
(6) Begin to raise entrance door with inside AIRSTAIR switch. Ensure there is no fluid leakage from swivel or hydraulic connection during door operation.
(7) Fully raise and lock door.
(8) Observe that door seal inflates properly when bayonets are extended. Ensure there is no air leakage at swivel or pneumatic line connection.
(9) Open and fully extend door. Check for fluid leakage.
(10) Remove air supply and electrical power if no longer needed.
(11) Install fairing cover.

9. **Entrance Door Hinge Pins — Inspection**

CAUTION: REMOVE HINGE PINS ONE AT A TIME, KEEP WEIGHT OFF STEPS WHILE PINS ARE REMOVED.

NOTE: This inspection complies with CB 291.

A. Lower entrance door to fully down and locked position.
B. Pull hinge pins (piano wire) out of lower hinge on top step and fold step and riser up.
C. Remove metal safety locks that secures the hinge pin, retaining hardware.
D. Remove air stair door hinge pins (piano wire) one at a time.
E. Inspect for visible indications of wear.
F. Install hinge pin.
G. Repeat Step E for remaining hinge pin.
H. Install hinge pin safety locks with hardware retained from Step C. above.
I. Lower step and riser and install hinge pins removed in Step B. above.

10. **Entrance and Baggage Door — Operational Test**

WARNING: NEVER ALLOW THE DOOR TO FREE FALL OPEN IF AIR HAS NOT BEEN BLED FROM HYDRAULIC SYSTEM AFTER SERVICING, OR IF LAST CLOSING OF DOOR WAS MADE MANUALLY. TAG PULL PLATE AFTER CLOSING DOOR MANUALLY, TO WARN OTHERS THAT DOOR MUST BE OPENED WITH EXTREME CARE, UTILIZING SUFFICIENT PERSONNEL TO EASE THE DOOR TO THE OPEN POSITION SINCE THE DASH POT ACTION IS NOT EFFECTIVE AS THE CYLINDER IS WITHOUT FLUID IN THIS CASE.

CAUTION: WHEN OPENING THE ENTRANCE DOOR, THE ROLLER ON THE PUSH BAR ASSEMBLY MUST ALWAYS BE ROTATED INTO THE SECOND DETENT OF THE SECONDARY LOCK BEFORE THE DOOR IS PERMITTED TO SWING DOWN.

DO NOT PLACE ANY WEIGHT ON THE STAIRS UNLESS THE STAIRS ARE SUPPORTED AT THE LOWER END. NEVER LEAVE THE OUTSIDE BATTERY SWITCH IN THE ON POSITION UNLESS ACTUATING THE DOOR SYSTEM USING THE NOSE WHEEL WELL OUTSIDE DOOR SWITCH.

NOTE: If aircraft has DOOR ISOLATION switches installed, place them in NORM (guard down-toggle down position) before starting this test.

This test starts with entrance and baggage doors open.

A. Energize main and essential dc buses. CABIN OPEN lights in master panel should be on.
B. Supply a pressure of 15 psi or greater in de-icing system pressure manifold (as shown on pressure gage in cockpit) by connecting air pressure to pneumatic de-icing system upstream of regulator in either nacelle or operating APU with full air output to cabin and cockpit.

C. Close and lock baggage door. **Do not insert pip pin in secondary lock handle.** Observe that door seal inflates when bayonets extend. CABIN OPEN light (master warning lights panel) should remain on.
D. Check that bayonets are fully extended by observing that orange dot on each pin is fully visible tangent to outermost face of pin guide plate. A tolerance of ± 1/64-inch is allowable.
E. Close entrance door using AIRSTAIR switch (pilots circuit breaker panel). Lock door. CABIN OPEN light will still remain on. Place AIRSTAIR switch OFF. Observe that entrance door seal inflates when bayonets are extended. Check that all bayonets are fully extended by observing that orange dot on each bayonet is visible in same manner as those on baggage door.
F. Place DOOR SAFETY switch (lower overhead panel) to SAFE. CABIN OPEN lights should go out.
G. Slightly lift entrance door secondary lock handle. CABIN OPEN light should come on. **Do not open door.**
H. Place entrance door secondary lock handle to lock position. CABIN OPEN light should go out.
I. Place pip pin in baggage door secondary lock handle. CABIN OPEN light should come on.
J. Remove and stow baggage door secondary lock handle pip pin. CABIN OPEN light should go out.
K. Open baggage door. Observe that seal deflates. CABIN OPEN light should come on during first movement of secondary lock handle and remain on.

NOTE: Steps L - Q apply only to Aircraft 81 - 200 including 322, 323 and those Aircraft having ASC 110 (Door Isolation Switches). Aircraft 1 — 80 including 114 not having ASC 110 continue with Step R. to end of test.

L. In cockpit, place BAGGAGE DOOR ISOLATION switch (outboard side of Pilots Station 133 relay panel) to guard-up, toggle-up position. CABIN OPEN lights should go out.
M. Return BAGGAGE DOOR ISOLATION switch to NORM, guard-down, toggle-down position. CABIN OPEN light should come on.
N. Close and lock baggage door. **Do not insert pip pin into secondary lock handle.** CABIN OPEN light should go out.
O. Place DOOR SAFETY switch to UNSAFE. Open entrance door using AIRSTAIR switch. Observe that seal deflates when bayonets are retracted. CABIN OPEN light should come on when secondary lock handle is moved and should remain on.
P. Place DOOR SAFETY switch to SAFE. Place MAIN DOOR ISOLATION switch to guard up, toggle-up position. CABIN OPEN light should go out.
Q. Return MAIN DOOR ISOLATION switch to guard-down, toggle-down position. CABIN LIGHT should come on.
R. Place DOOR SAFETY switch to UNSAFE. Turn off APU or external air switch (if used), BAT NORM / EMERG switch and EX PWR switch (if used).
S. Open outside battery switch cover, left nacelle and turn on outside battery switch.

NOTE: Left battery must be connected.

T. Close entrance door using OUTSIDE DOOR switch in nosewheel well. Lock entrance door using outside lock handle and push-plate. When primary lock handle inserts bayonets, the auxiliary pump will stop. Return OUTSIDE DOOR switch to OFF.
U. Open entrance door by depressing push-plate and pulling lock handle. Allow door to free fall. Ensure that action is normal on door extension and that stair down latch is properly engaged before stair finally rests on ground (note audible engagement.) Observe dampened action of door and stair opening toward latter part of extension.
V. Open baggage door using outside handle. Ensure that handle moves freely with no evidence of binding.
W. Close baggage door using outside handle and ensure proper operation.
X. Turn off outside battery switch and close cover plate.
Y. Return system to normal configuration and position doors as desired.

11. **Entrance Door Structure — Proper Fit / Cables — Inspection**
A. Inspect main entrance door for proper fit as follows:
(1) Close entrance door.
(2) From inside of aircraft with door latched, grasp door handle and alternately apply body weight in the inboard and outboard direction at the top center of the door. If the door does not exhibit movement or sounds which can be attributed to freedom of movement between bayonets and retainers, the door is acceptable.
(3) If door does exhibit movement or sounds, proceed to Steps (4) and (5) below.

(4) With door in latched position, measure and record gap between outer skin of door and adjacent fuselage while applying an outboard load. Measure at door top centerline.
(5) Apply an inboard load and take measurement at same point. Compare readings, if difference exceeds 0.040 inch, door has abnormal displacement. Bayonets and striker plates should be inspected to determine if replacement is required.

B. Extend entrance door to fully down and locked position.
C. Remove forward and aft airstair door sector cover assembly. Retain hardware.
D. Inspect cables for wear, broken strands and flat spots.

NOTE: If a cable does not pass inspection, it is recommended that cables be replaced as a set.

E. Remove fairings between airstair door and insulation packages.
F. Inspect door structure for corrosion. Treat any corrosion found. (This inspection complies with part #1 of CB 307, corrosion in the main entry door.)

NOTE: Engineering evaluation is required whenever corrosion has removed more than 20% of the original material thickness over a distance equal to the flange width if it occurs on a flange, or 1 inch if it occurs on a web.

G. Install insulation packages and fairings between the airstair and door.
H. Install forward and aft airstair door sector cover assembly using retained hardware from Step C. above
I. If cables have been replaced, perform Entrance and Baggage Door — Operational Test.

12. **Entrance Door Cable Tension — Check**

A. Extend entrance door to fully down and locked position.
B. Remove forward and aft airstair door sector cover. Retain hardware.
C. Inspect cables, if cables show signs of wear (broken strands, flat spots, etc.) replace cables as a set.
D. Check tension on cables as follows:
(1) Unlock lower set of steps from the down position. Raise entrance door by hand to relieve load on long pushrod on forward side of door. Remove pushrod.
(2) Raise lower set of steps until 5/16 inch bolt that attaches driveshaft crank to outer rim of large quadrant can be removed. Remove bolt.

CAUTION: DO NOT OPERATE DOOR WITH RIG PIN INSTALLED.

(3) Install 5/16 inch rig pin into crank of drive shaft assembly until rig pin protrudes into rig pin hole of upper set of steps.
(4) Lock lower set of steps in the unfolded position.
(5) Check cable tension. Should be 100 $\pm$5 pounds.

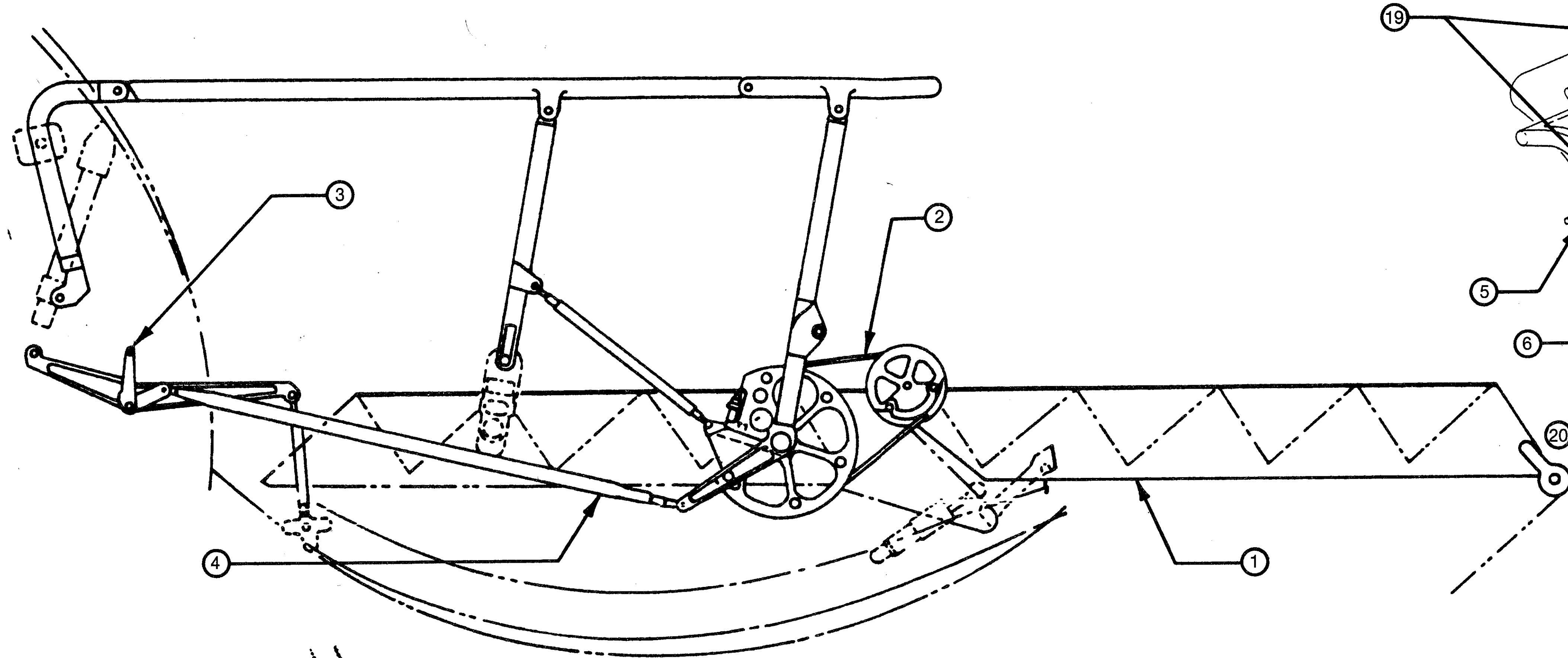
E. Unlock lower set of steps from the down positon.
F. Remove rig pin and install 5/16 inch bolt through drive shaft and large quadrant.
G. Install and tighten nut on bolt.
H. Install pushrod.

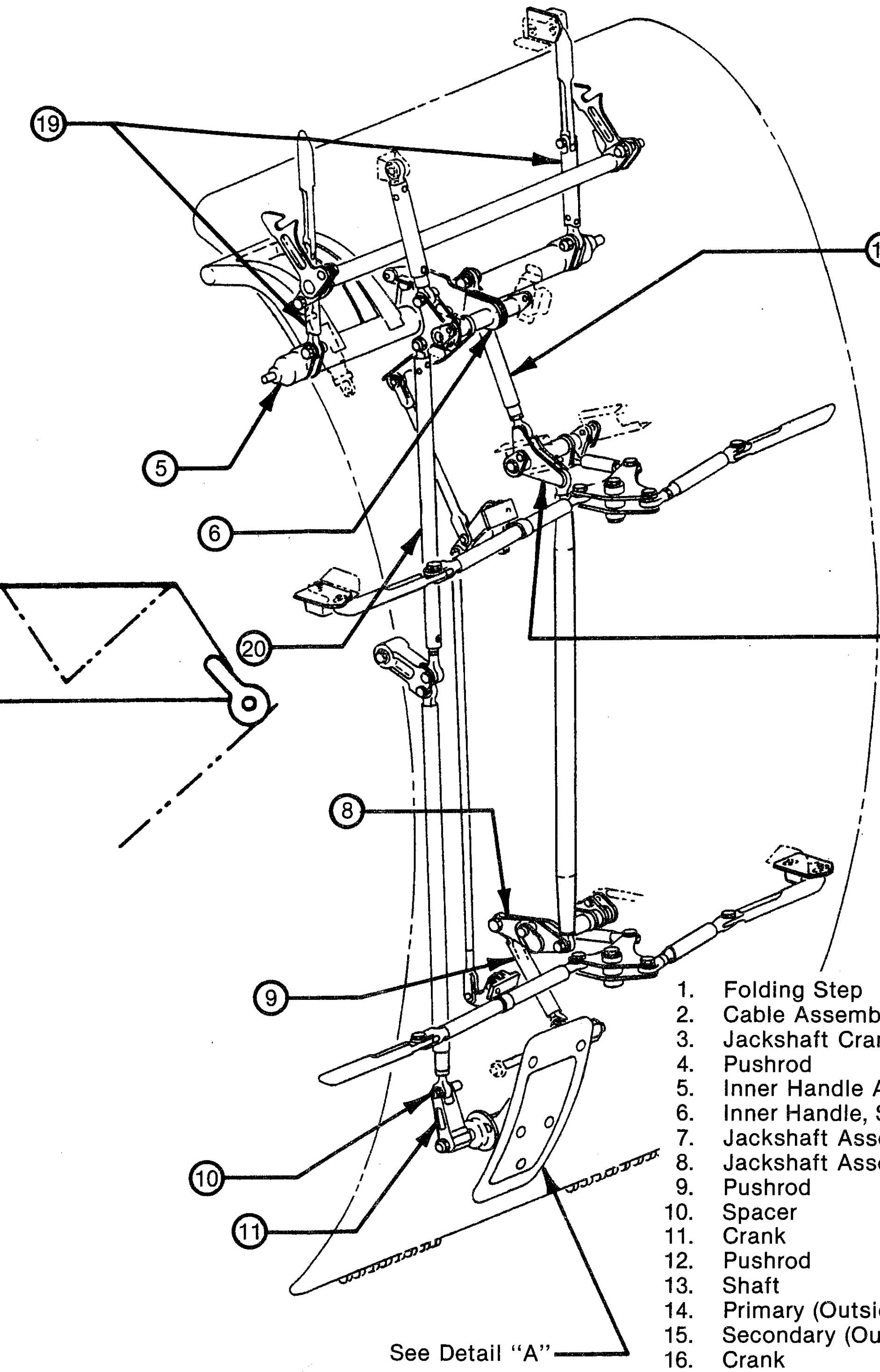
NOTE: Adjustment of this rod and screw stop may be necessary so lower steps will lock in down position.

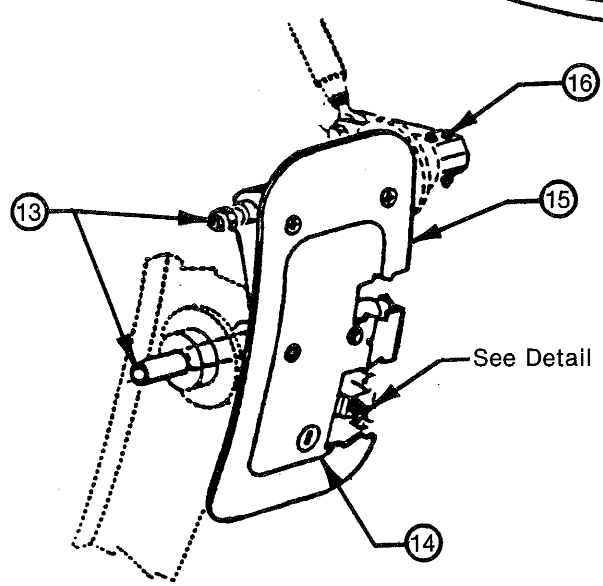
I. Operate door slowly and check clearances in the folded position.

NOTE: A clearance of 4-1/2 feet below fuselage center is required to fully extend stair.

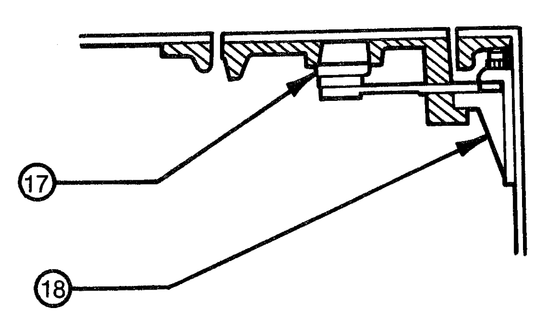
J. Inspect for presence of foreign objects then install fairings.







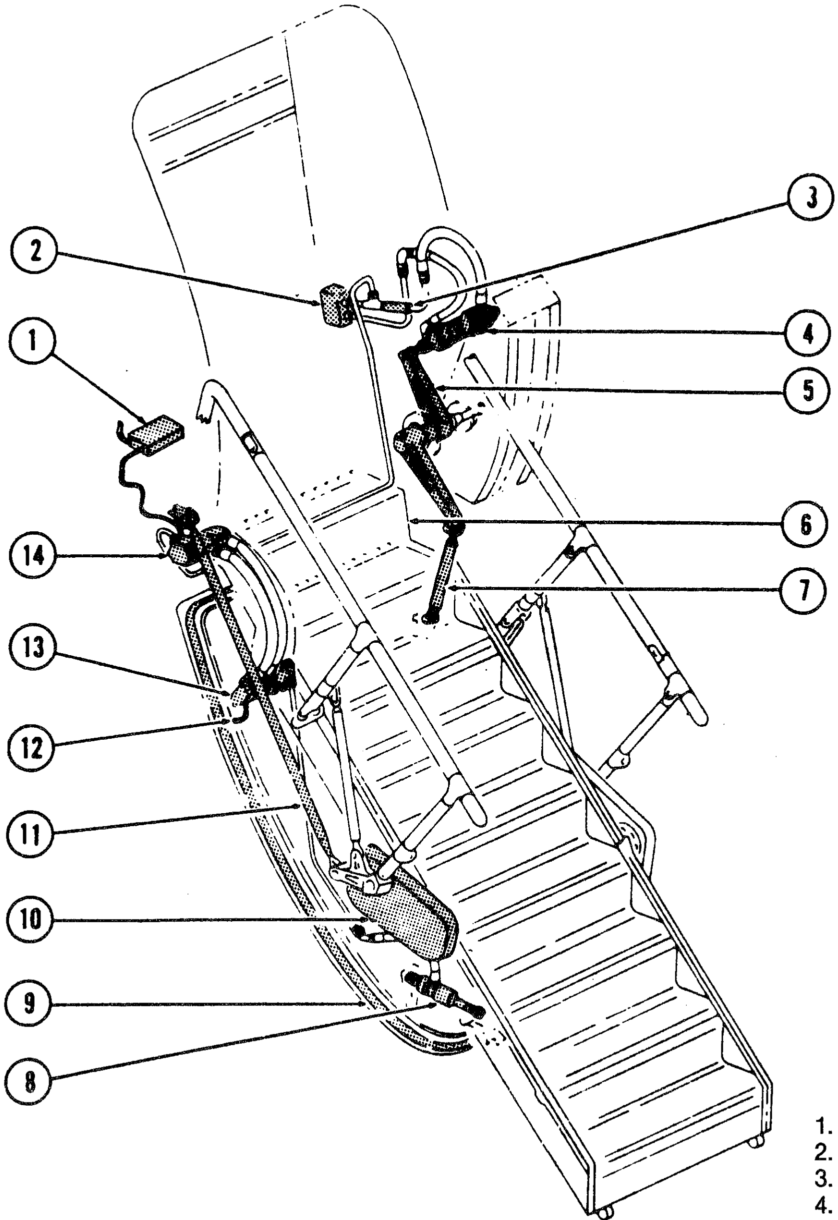
Detail "A"



Detail "B"

1. Folding Step
2. Cable Assembly
3. Jackshaft Crank Assembly
4. Pushrod
5. Inner Handle Assembly
6. Inner Handle, Secondary Lock
7. Jackshaft Assembly, Upper
8. Jackshaft Assembly, Lower
9. Pushrod
10. Spacer
11. Crank
12. Pushrod
13. Shaft
14. Primary (Outside) Latch
15. Secondary (Outside) Latch
16. Crank
17. Lock Cylinder
18. Stop
19. Pushrods
20. Pushrod

Airstair Door Installation Diagram
Figure 201



1. Entrance Door Seal Control Valve
2. Entrance Door Selector Valve
3. Restrictor
4. Actuating Cylinder
5. Jackshaft Assembly
6. Upper Step
7. Aft Pushrod
8. Downlatch Cylinder
9. Door Seal
10. Sector Cable Housing
11. Forward Pushrod
12. Door Seal Inflation Tube
13. Door Swivel
14. Floor Swivel

Airstair Door
Figure 202

C. Close and lock baggage door. **Do not insert pip pin in secondary lock handle.** Observe that door seal inflates when bayonets extend. CABIN OPEN light (master warning lights panel) should remain on.
D. Check that bayonets are fully extended by observing that orange dot on each pin is fully visible tangent to outermost face of pin guide plate. A tolerance of ± 1/64-inch is allowable.
E. Close entrance door using AIRSTAIR switch (pilots circuit breaker panel). Lock door. CABIN OPEN light will still remain on. Place AIRSTAIR switch OFF. Observe that entrance door seal inflates when bayonets are extended. Check that all bayonets are fully extended by observing that orange dot on each bayonet is visible in same manner as those on baggage door.
F. Place DOOR SAFETY switch (lower overhead panel) to SAFE. CABIN OPEN lights should go out.
G. Slightly lift entrance door secondary lock handle. CABIN OPEN light should come on. **Do not open door.**
H. Place entrance door secondary lock handle to lock position. CABIN OPEN light should go out.
I. Place pip pin in baggage door secondary lock handle. CABIN OPEN light should come on.
J. Remove and stow baggage door secondary lock handle pip pin. CABIN OPEN light should go out.
K. Open baggage door. Observe that seal deflates. CABIN OPEN light should come on during first movement of secondary lock handle and remain on.

NOTE: Steps L - Q apply only to Aircraft 81 - 200 including 322, 323 and those Aircraft having ASC 110 (Door Isolation Switches). Aircraft 1 — 80 including 114 not having ASC 110 continue with Step R. to end of test.

L. In cockpit, place BAGGAGE DOOR ISOLATION switch (outboard side of Pilots Station 133 relay panel) to guard-up, toggle-up position. CABIN OPEN lights should go out.
M. Return BAGGAGE DOOR ISOLATION switch to NORM, guard-down, toggle-down position. CABIN OPEN light should come on.
N. Close and lock baggage door. **Do not insert pip pin into secondary lock handle.** CABIN OPEN light should go out.
O. Place DOOR SAFETY switch to UNSAFE. Open entrance door using AIRSTAIR switch. Observe that seal deflates when bayonets are retracted. CABIN OPEN light should come on when secondary lock handle is moved and should remain on.
P. Place DOOR SAFETY switch to SAFE. Place MAIN DOOR ISOLATION switch to guard up, toggle-up position. CABIN OPEN light should go out.
Q. Return MAIN DOOR ISOLATION switch to guard-down, toggle-down position. CABIN LIGHT should come on.
R. Place DOOR SAFETY switch to UNSAFE. Turn off APU or external air switch (if used), BAT NORM / EMERG switch and EX PWR switch (if used).
S. Open outside battery switch cover, left nacelle and turn on outside battery switch.

NOTE: Left battery must be connected.

T. Close entrance door using OUTSIDE DOOR switch in nosewheel well. Lock entrance door using outside lock handle and push-plate. When primary lock handle inserts bayonets, the auxiliary pump will stop. Return OUTSIDE DOOR switch to OFF.
U. Open entrance door by depressing push-plate and pulling lock handle. Allow door to free fall. Ensure that action is normal on door extension and that stair down latch is properly engaged before stair finally rests on ground (note audible engagement.) Observe dampened action of door and stair opening toward latter part of extension.
V. Open baggage door using outside handle. Ensure that handle moves freely with no evidence of binding.
W. Close baggage door using outside handle and ensure proper operation.
X. Turn off outside battery switch and close cover plate.
Y. Return system to normal configuration and position doors as desired.

11. **Entrance Door Structure — Proper Fit / Cables — Inspection**
A. Inspect main entrance door for proper fit as follows:
(1) Close entrance door.
(2) From inside of aircraft with door latched, grasp door handle and alternately apply body weight in the inboard and outboard direction at the top center of the door. If the door does not exhibit movement or sounds which can be attributed to freedom of movement between bayonets and retainers, the door is acceptable.
(3) If door does exhibit movement or sounds, proceed to Steps (4) and (5) below.

(4) With door in latched position, measure and record gap between outer skin of door and adjacent fuselage while applying an outboard load. Measure at door top centerline.
(5) Apply an inboard load and take measurement at same point. Compare readings, if difference exceeds 0.040 inch, door has abnormal displacement. Bayonets and striker plates should be inspected to determine if replacement is required.

B. Extend entrance door to fully down and locked position.
C. Remove forward and aft airstair docr sector cover assembly. Retain hardware.
D. Inspect cables for wear, broken strands and flat spots.

NOTE: If a cable does not pass inspection, it is recommended that cables be replaced as a set.

E. Remove fairings between airstair and door.
F. Inspect door structure for corrosion. Treat any corrosion found. (This inspection complies with part #1 of CB 307, corrosion in the main entry door.)

NOTE: Engineering evaluation is required whenever corrosion has removed more than 20% of the original material thickness over a distance equal to the flange width if it occurs on a flange, or one inch if it occurs on a web.

F. Replace fairings between the airstair and the door.
G. Install forward and aft airstair door sector cover assembly using retained hardware from Step C. above
H. If cables have been replaced, perform Entrance and Baggage Door — Operational Test.

12. **Entrance Door Cable Tension — Check**

A. Extend entrance door to fully down and locked position.
B. Remove forward and aft airstair door sector cover. Retain hardware.
C. Inspect cables, if cables show signs of wear (broken strands, flat spots, etc.) replace cables as a set.
D. Check tension on cables as follows:
(1) Unlock lower set of steps from the down position. Raise entrance door by hand to relieve load on long push-rod on forward side of door. Remove push-rod.
(2) Raise lower set of steps until 5/16 inch bolt that attaches driveshaft crank to outer rim of large quadrant can be removed. Remove bolt.

CAUTION: DO NOT OPERATE DOOR WITH RIG PIN INSTALLED.

(3) Install 5/16 inch rig pin into crank of drive shaft assembly until rig pin protrudes into rig pin hole of upper set of steps.
(4) Lock lower set of steps in the unfolded position.
(5) Check cable tension. Should be 100 $\pm$5 pounds.

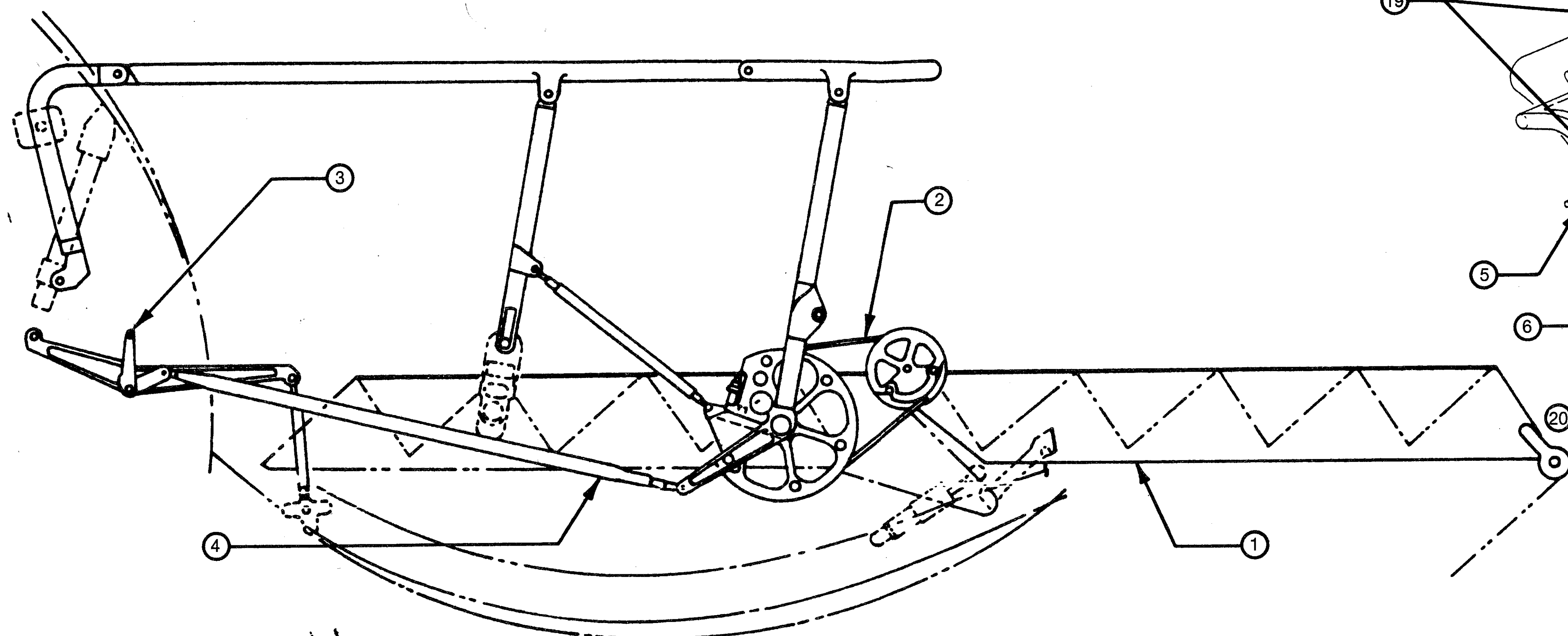
E. Unlock lower set of steps from the down positon.
F. Remove rig pin and install 5/16 inch bolt through drive shaft and large quadrant.
G. Install and tighten nut on bolt.
H. Install pushrod.

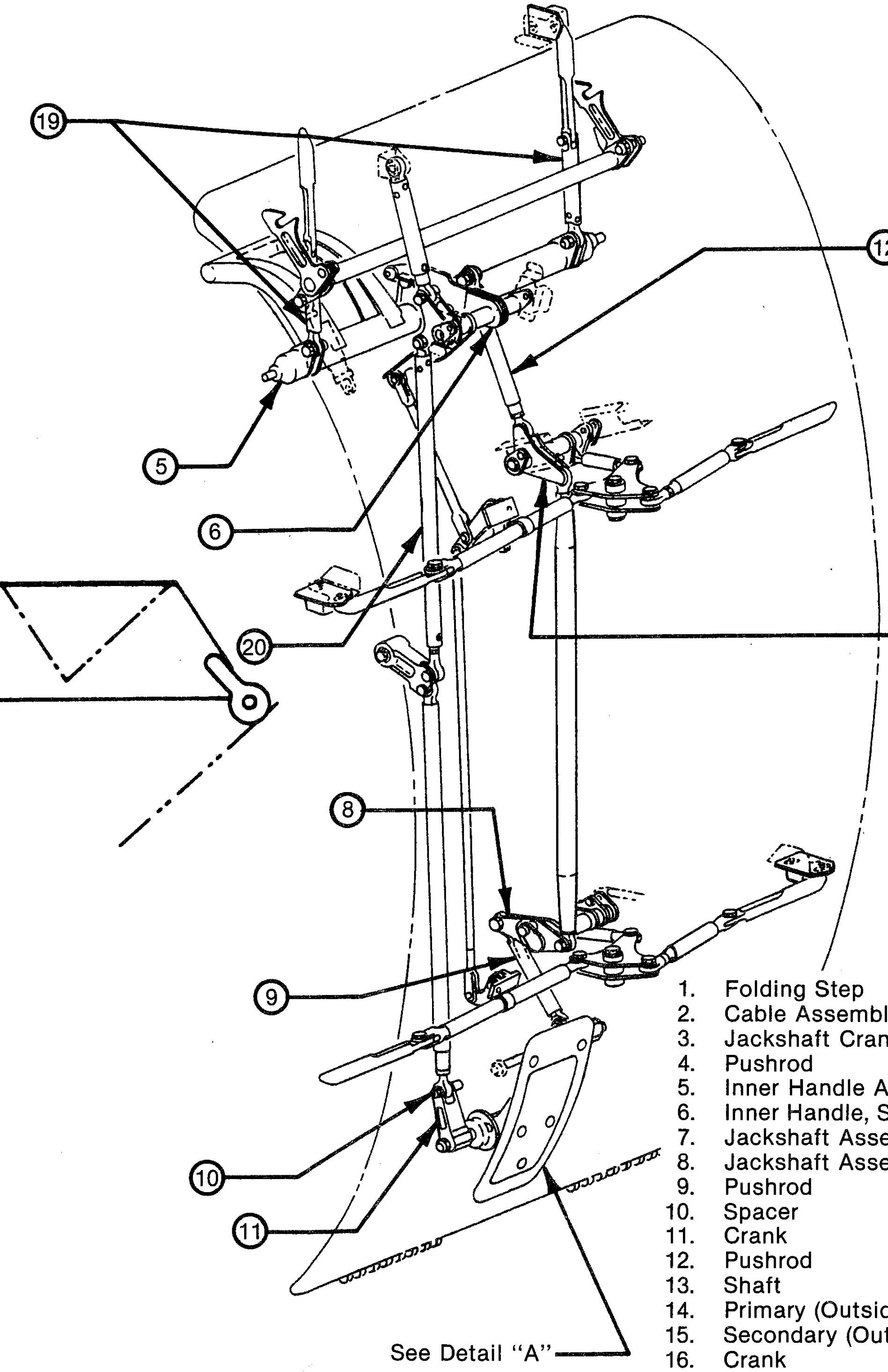
NOTE: Adjustment of this rod and screw stop may be necessary so lower steps will lock in down position.

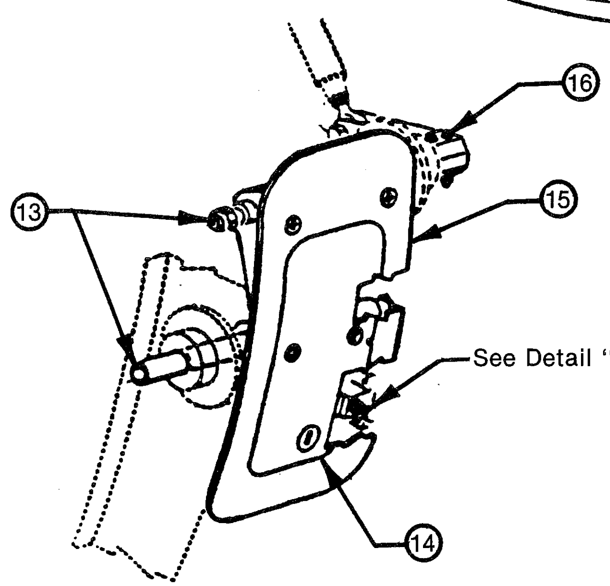
I. Operate door slowly and check clearances in the folded position.

NOTE: A clearance of 4-1/2 feet below fuselage center is required to fully extend stair.

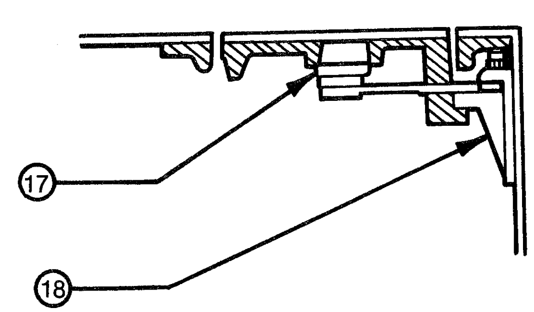
J. Inspect for presence of foreign objects then install fairings.







Detail "A"



Detail "B"

1. Folding Step
2. Cable Assembly
3. Jackshaft Crank Assembly
4. Pushrod
5. Inner Handle Assembly
6. Inner Handle, Secondary Lock
7. Jackshaft Assembly, Upper
8. Jackshaft Assembly, Lower
9. Pushrod
10. Spacer
11. Crank
12. Pushrod
13. Shaft
14. Primary (Outside) Latch
15. Secondary (Outside) Latch
16. Crank
17. Lock Cylinder
18. Stop
19. Pushrods
20. Pushrod

Airstair Door Installation Diagram
Figure 201

MAIN ENTRANCE DOOR HYDRAULIC SYSTEM — DESCRIPTION / OPERATION

1. Description

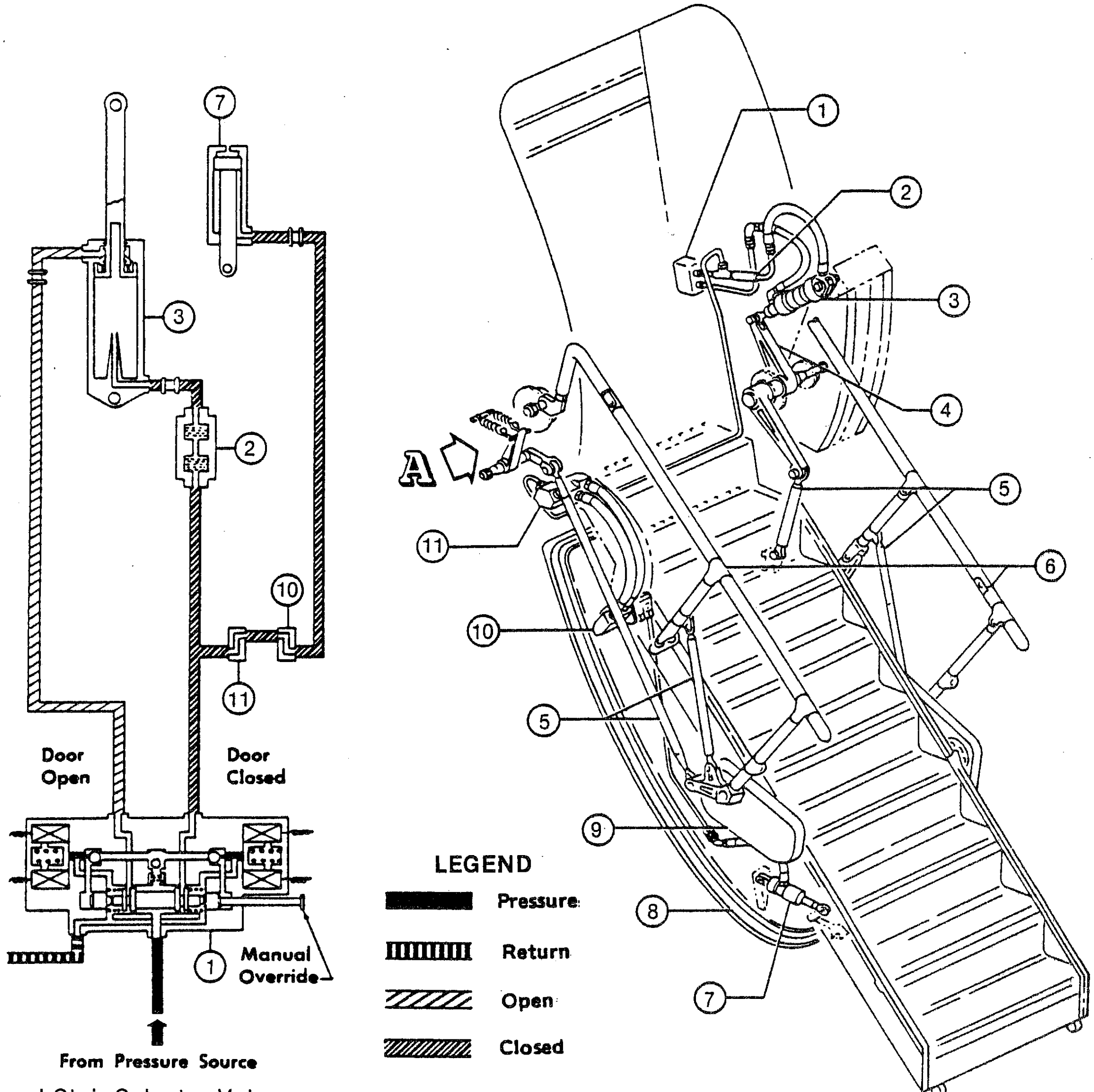
(See Figure 1 and Figure 2)

The airstair door hydraulic system consists of a door and stairs selector valve, floor swivel, door swivel, one restrictor, door and stair cylinder, and down latch release cylinder. The system is supplied operating pressure by the auxiliary hydraulic system. The door and stairs selector valve, located aft of the bulkhead at Fuselage Station 169, directs hydraulic pressure to the rod end of the door and stair cylinder when the door is to be opened, and to the head end when the door is to be closed. Floor and door swivels, located at the door hinge, allow hydraulic pressure to be directed to down latch release cylinder and air pressure to door seal.

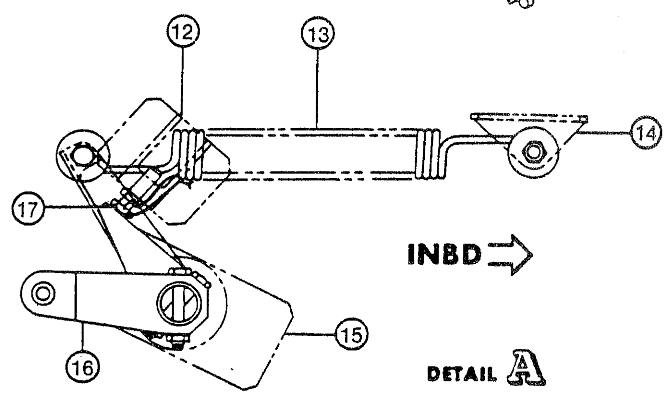
2. Operation

When tracing flow paths on the schematic flow diagram, auxiliary hydraulic system pressure (when made available to pressure port of door and stair selector valve) flows through the valve, past the selector spool, through the filter, past the ball check and to the door open or closed pressure seats. Setting either door control switch to OPEN energizes the solenoid on the right side of the valve. When solenoid plunger is retracted, the poppet leaves the pressure seat and closes off the return. Hydraulic fluid then is permitted to enter right of the spool, shifting the spool left. Hydraulic fluid then flows out of the door OPEN port and into the rod end of the door and stairs cylinder. The cylinder then retracts opening the door and unfolding the stairs. Return pressure flows from the head end of the cylinder, through a restrictor and to the door CLOSED port of the selector valve. Return fluid then flows past the selector spool to the return port of the valve and back to the reservoir.

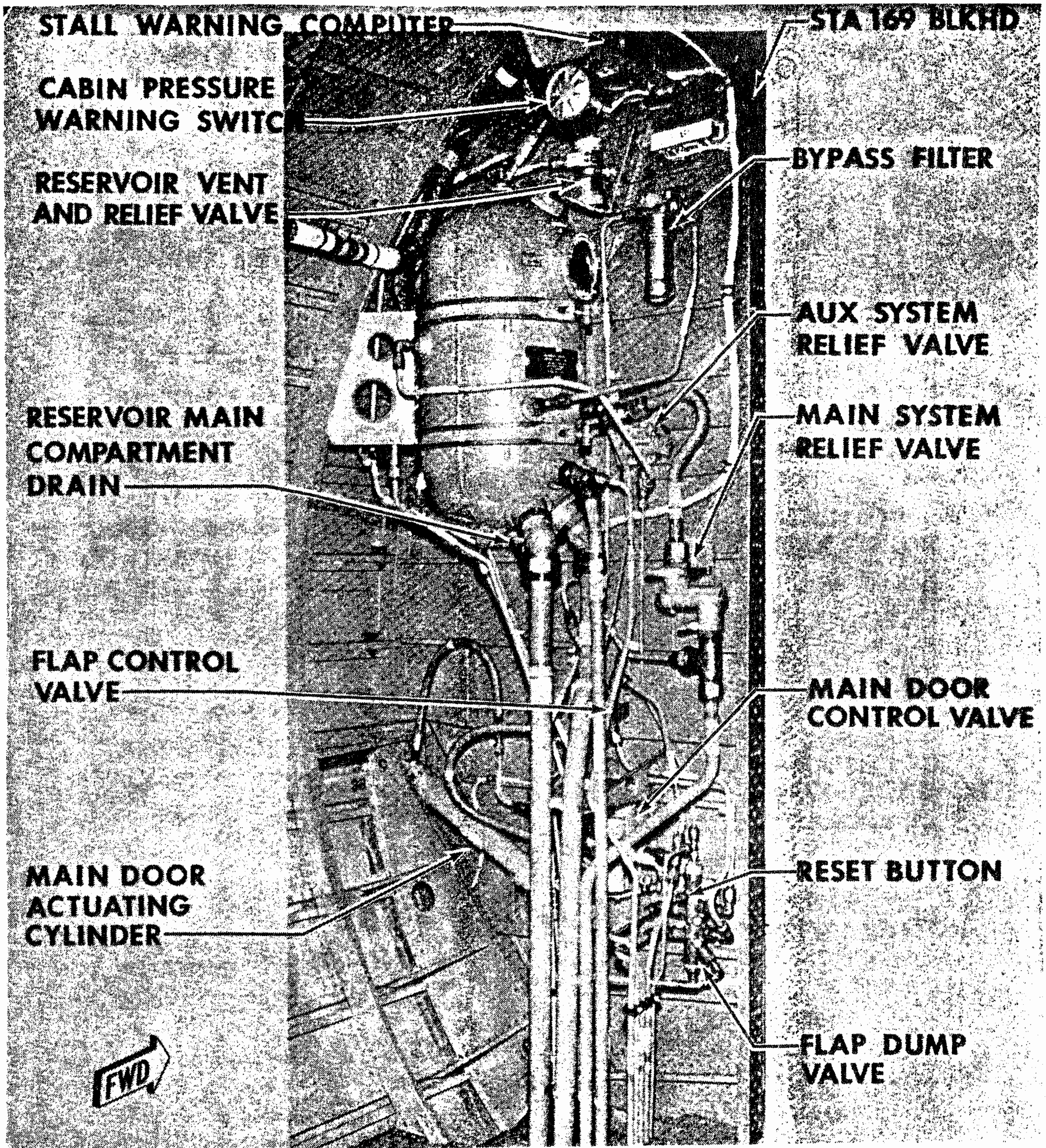
Setting either door control switch to CLOSE energizes the solenoid on the left side. When solenoid retracts, the poppet will leave the pressure seat and close off the return. Hydraulic pressure then is permitted to enter left of the spool, thus shifting the spool right. Hydraulic pressure then flows out of the door close port into the head end of the door and stairs cylinder and to the down latch release cylinder through the door and stair swivels. The down latch release cylinder then retracts, removing locking hooks, allowing door and stairs cylinder to fold stairs and close the door.



1. Door and Stair Selector Valve
2. Restrictor
3. Door and Stair Cylinder
4. Jackshaft
5. Push Rods
6. Handrails
7. Down Latch Cylinder
8. Inflatable Door Seal
9. Sector Cable Housing
10. Door Swivel
11. Floor Swivel
12. Push Rod Stop Fitting
13. Overtravel Spring
14. Overtravel Spring Fitting
15. Jackshaft Bearing
16. Jackshaft Assembly
17. Adjusting Bolt



Main Entrance Door Hydraulic System
Figure 1.



Hydraulic Compartment
Figure 2.

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AIRSTAIR DOOR ELECTRICAL CONTROL SYSTEM — DESCRIPTION / OPERATION

1. **General** (See Figure 1 and 2)

The airstair door electrical control system consists of a pilots door safety switch, main door lock switches, inside and outside door switches, inside switch control relay and auxiliary hydraulic pump switches and control relay. The system is powered from the main dc bus through a 5A DOOR CONT circuit breaker, pilots circuit breaker panel. Moving DOOR SAFETY switch, located on the lower overhead panel, to SAFE causes the circuit between the power source and door control switches to open. When the DOOR SAFETY switch is set to UNSAFE, a door warning light on the master panel will come on. Door lock switches, located within the structure surrounding the door, are actuated by bayonets in the door lock mechanism. When the door is unlocked, the released lock switches will complete the power circuit between the door safety switch and the inside and outside door control switches. The outside door switch, mounted on the nose wheel well junction box, allows the door to be operated only when the door safety switch is set to UNSAFE and bayonets retracted.

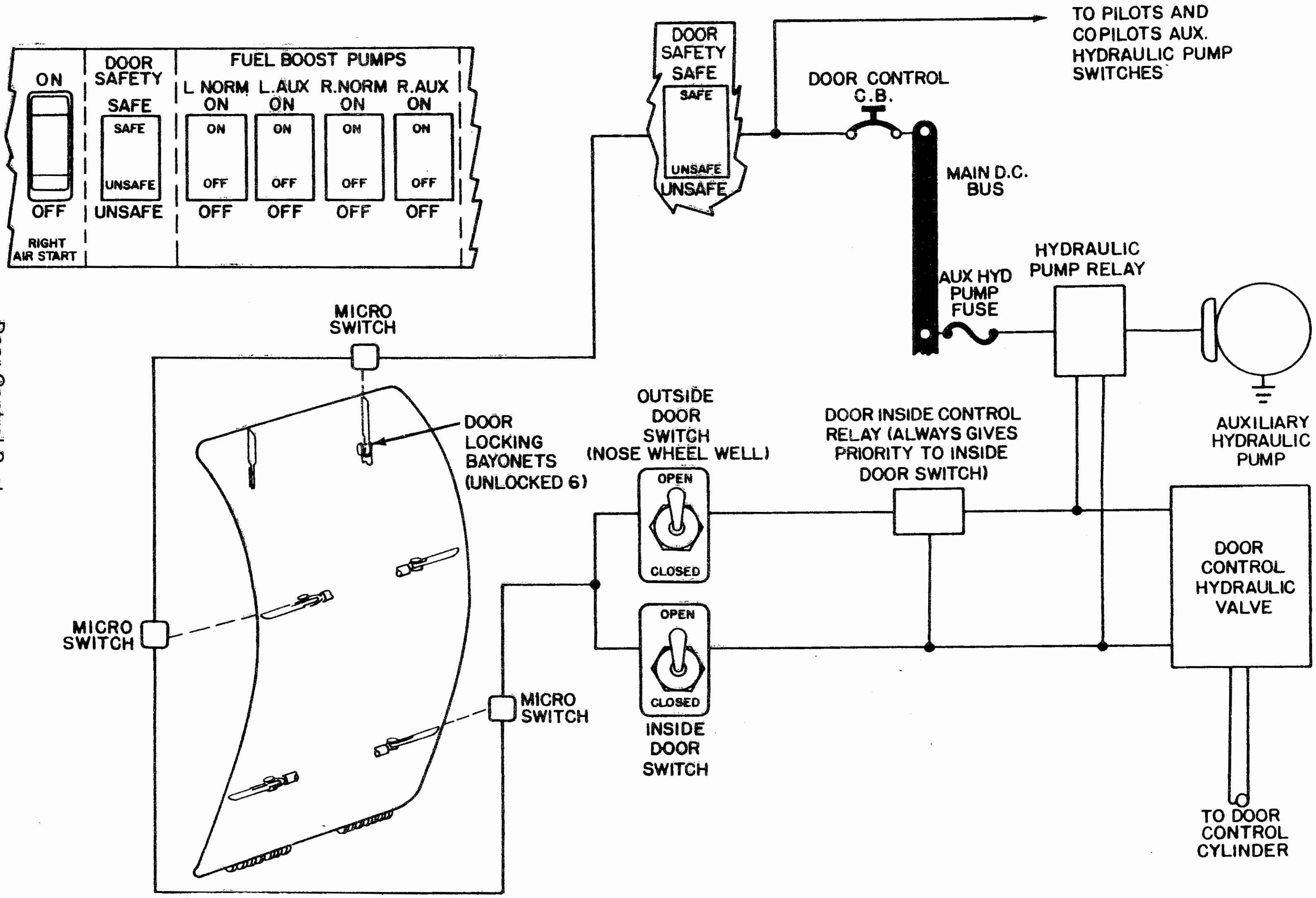
The inside AIRSTAIR SW, located on panel behind the pilots seat, makes door control possible without outside ground handling equipment. The inside door switch control relay in the mid relay box under floorboard No. 8 enables the inside AIRSTAIR SW to override the outside door switch.

Placing DOOR SAFETY switch to UNSAFE with main dc bus energized and door unlocked, will complete the door control power circuit (See Figure 2). Holding inside and outside door control switches to required position will automatically energize the auxiliary hydraulic pump in the nose wheel well and door and stairs selector valve. The pump then supplies auxiliary system pressure to energized open pressure port of the selector valve. The retraction mechanism, operated by the door and stair hydraulic cylinder in the hydraulic compartment, will open or close the door and fold or unfold the stairs. For security purposes on ground, a conventional key type lock is included in the push pull panel in outside door surface.

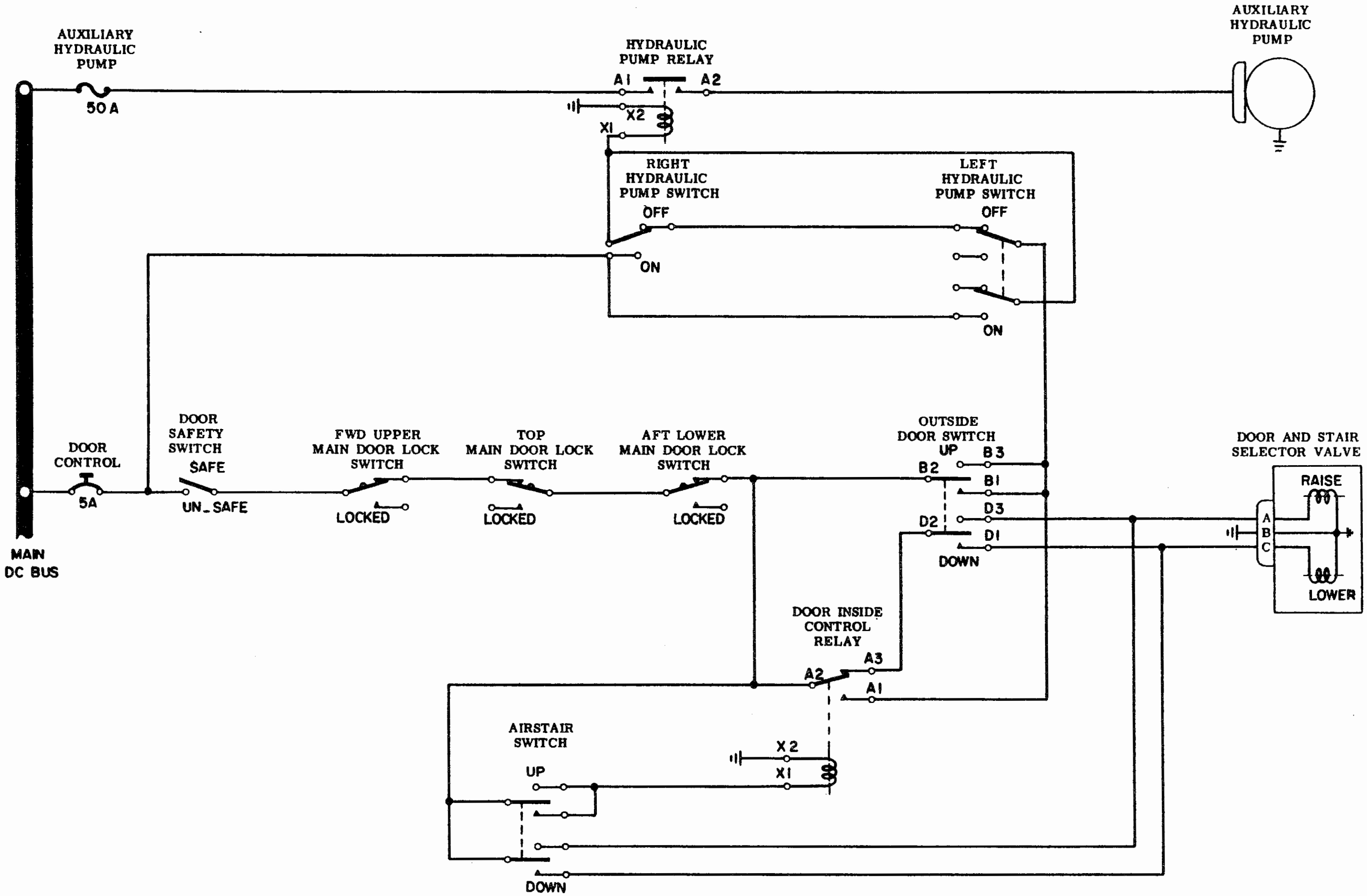
During an extension cycle, it is necessary to hold the AIRSTAIR SW in position. When switch is released it springs to OFF, de-energizing system.

During a retraction cycle, the AIRSTAIR SW is detented. When door reaches its closed position, operating primary door locking mechanism, it de-energizes the control system. The AIRSTAIR SW should then be set to OFF.

When an external source of power is used, the EX PWR switch, located on the pilots overhead panel must be set to EX PWR. If external power is not available, door may be closed by using aircraft battery. To close door from outside, turn on OUTSIDE BAT SW located on the outboard side of the left nacelle. The OUTSIDE DOOR SW in the nose wheel well then is placed in the CLOSE position. After the door is closed and locked, release the OUTSIDE DOOR SW switch, set OUTSIDE BAT SW to center position and close switch access cover.



Door Control System
Figure 1



Airstair Door Control Schematic
Figure 2

"Reference to manufactures in this Tech. Pub. are for identification purposes only and are neither specified nor furnished as a source for obtaining such parts."

MAIN ENTRANCE DOOR HYDRAULIC SYSTEM — MAINTENANCE PRACTICES

1. Main Entrance Door Actuating Cylinder — Removal / Installation

A. Removal

(1) Dissipate hydraulic system pressure.

(2) Remove access cover aft side of Station 169 Bulkhead.

(3) Remove actuator.

B. Installation

(1) Manually place door in closed and locked position as follows:

CAUTION: THE FOLLOWING PROCEDURE IS ONLY USED WHEN ABSOLUTELY NECESSARY AND IS NOT A NORMAL OPERATING MODE.

DO NOT RAISE LOWER FOUR STEPS MORE THAN TEN INCHES ABOVE NORMAL STATIC GROUND POSITION WITHOUT FIRST RELEASING HOOKS AT LOWER FOLD JOINT OF LOWER STAIR. SUPPORT DOOR ABOVE LOWER FOLD JOINT BEFORE RELEASING HOOKS. WITHOUT SUPPORT, DOOR EXTERNAL HANDLE WILL DAMAGE FUSELAGE WHEN HOOKS ARE RELEASED.

(a) Release hooks at lower fold joint of lower stair.

(b) Raise main entrance door to closed position (Use sufficient personnel).

(2) Install head end of actuator and connect hydraulic lines.

(3) Hydraulically operate actuator to full length.

CAUTION: USE EXTREME CARE WHEN EXTENDING ACTUATOR TO PREVENT DAMAGE TO STRUCTURE OR ACTUATOR TERMINAL.

(4) Adjust actuator terminal so that extended length is 1/16 inch longer than required to align terminal with hole in door jack shaft.

NOTE: Door actuator may require additional adjustment to obtain proper snubbing action. If door slams upon initial retraction, shorten door actuator rod end as required to obtain proper snubbing.

(5) Tighten locking nut and safety with lockwire.

(6) Perform Main Entrance Door — Operational Test, See Section 52-1-0.

(7) Inspect for presence of foreign objects then install access cover.

2. Main Entrance Door Actuating Cylinder — Adjustment

NOTE: To ensure door operating lever is not overloaded in closed position through action of manually engaging locking bayonets, adjust door actuating cylinder as follows:

A. Open main entrance door and disconnect terminal end of actuating cylinder from jackshaft arm.

B. Manually place door in closed and locked position.

C. Hydraulically extend actuating cylinder to its full length. Adjust rod end terminal by screwing it in or out so that length of cylinder is 1/16 inch longer than required to align terminal with corresponding hole in jackshaft arm when door is closed and locked. After adjustment, tighten terminal locking nut and key and secure with lockwire.

NOTE: Door actuator may require additional adjustment to obtain proper snubbing action. If door slams upon initial retraction, shorten door actuator rod end as required to obtain proper snubbing.

D. Connect cylinder terminal to jackshaft arm. Observe all precautions when opening and closing door manually.

3. **Main Entrance Door Selector Valve — Removal / Installation**

(See Figure 201)

A. Removal

(1) Gain access to selector valve in hydraulic compartment aft of entrance door.

(2) Disconnect electrical connector from selector valve.

(3) Disconnect hydraulic lines from selector valve.

(4) Install protective caps on electrical and hydraulic connector, lines and fittings.

(5) Remove selector valve and retain hardware.

B. installation

(1) Install selector valve with hardware previously removed.

(2) Remove all protective caps from hydraulic lines and ports, and electrical connector.

(3) Connect hydraulic lines and electrical connector to valve.

(4) Energize main and essential dc buses.

(5) Raise and lock entrance door using inside or outside door switch. Check for proper operation.

(6) Unlock and lower entrance door using inside or outside door switch.

(7) Operate several times to eliminate any trapped air. Leave door open. Shut off electrical power.

(8) Replace access panels.

4. **Main Entrance Door Restrictor — Removal / Installation**

(See Figure 201)

A. Removal

(1) Gain access to restrictor in hydraulic compartment aft of entrance door.

(2) Disconnect hydraulic lines and remove restrictor.

(3) Install protective caps in hydraulic lines.

B. Installation

(1) Remove protective caps from hydraulic lines.

(2) Install restrictor between hydraulic lines.

(3) Energize main and essential dc buses.

(4) Place DOOR SAFETY switch in UNSAFE position.

(5) Raise and lock entrance door using inside airstair switch. Check for leakage and proper door operation.

(6) Operate door several times to eliminate trapped air.

(7) Return door to position desired and turn off electrical power.

(8) Close access.

5. **Main Entrance Door Downlatch Cylinder — Removal / Installation**

WARNING: AFTER CHANGING ANY HYDRAULIC COMPONENT IN THE ENTRANCE DOOR SYSTEM ENSURE THAT HYDRAULIC SYSTEM IS COMPLETELY BLED OF AIR BEFORE ALLOWING DOOR TO FREE FALL OPEN.

A. Removal

(See Figure 201)

(1) With entrance door in down position, use a hammer handle or other firm piece of wood to unlatch two latches at lower stair.

(2) Fold lower stair over upper stair.

(3) Place a padded stand under door to support entire assembly.

(4) Disconnect hydraulic line at downlatch cylinder. Install protective caps on open port and line.

(5) Remove cylinder. Retain hardware.

NOTE: Measure extended length of cylinder after removal if being replaced.

B. Installation

(1) Adjust replacement downlatch cylinder to exact extended length measured on removed cylinder.

(2) Install cylinder. Use hardware previously removed.

(3) Remove protective caps and install hydraulic line.

(4) Remove padded stand supporting door.

(5) Carefully unfold lower stair and engage latches.

(6) Energize main and essential dc buses.

(7) Using OUTSIDE DOOR switch begin to close door. When lower stair is folded over upper stair, release switch. Check downlatch cylinder and hydraulic line for fluid leakage.

(8) After leakage check, operate OUTSIDE DOOR switch to unfold and latch lower stair and fully extend door. Check for audible latch engagement.

(9) Remove electrical power.

6. Main Entrance Door and Floor Swivels — Removal / Installation

(See Figure 201)

WARNING: AFTER CHANGING ANY HYDRAULIC COMPONENT IN THE ENTRANCE DOOR SYSTEM ENSURE HYDRAULIC SYSTEM IS COMPLETELY BLED OF AIR BEFORE ALLOWING DOOR TO FREE FALL OPEN.

A. Removal

(1) Unlatch and lower entrance door.

(2) Remove fairing cover over door swivels at top forward end of entrance door.

(3) Disconnect hydraulic and pneumatic lines from swivel.

NOTE: The upper swivel is the floor swivel while the lower swivel is the door swivel. (4) Install protective caps on lines and ports.

(4) Remove swivel. Retain hardware.

B. Installation

(1) Install swivel using hardware previously removed.

(2) Remove protective caps from lines and fittings.

(3) Install hydraulic and pneumatic lines on swivel.

(4) Energize main and essential dc buses.

(5) Provide at least 15 psi of pressure in the de-icing system pressure manifold (as read on the cockpit gage) by either connecting an air supply to pressure manifold in nacelle or operating the APU with air on and set cabin and cockpit temperature controls to full cooling.

(6) Begin to raise entrance door with inside AIRSTAIR switch. Ensure there is no fluid leakage from swivel or hydraulic connection during door operation.

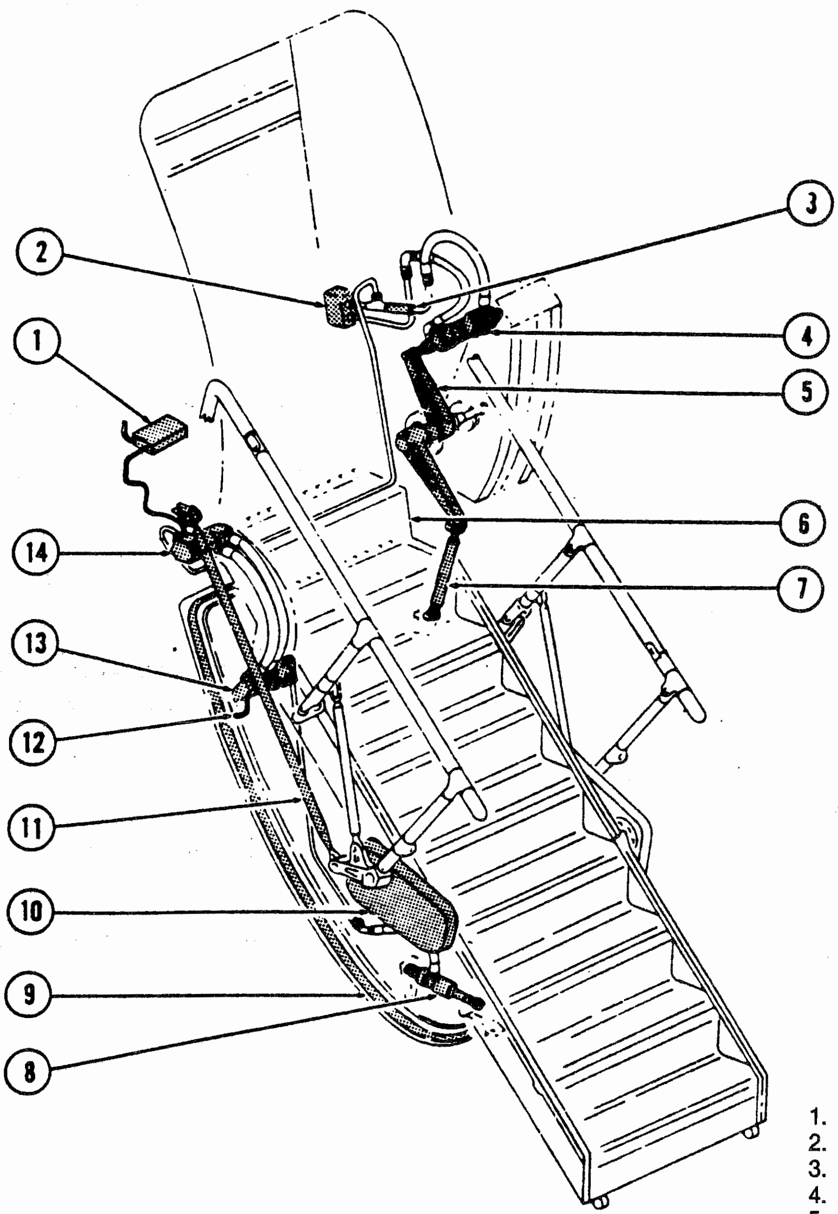
(7) Fully raise and lock door.

(8) Observe that door seal inflates properly when bayonets are extended. Ensure there is no air leakage at swivel or pneumatic line connection.

(9) Open and fully extend door. Check for fluid leakage.

(10) Remove air supply and electrical power if no longer needed.

(11) Install fairing cover.



1. Entrance Door Seal Control Valve
2. Entrance Door Selector Valve
3. Restrictor
4. Actuating Cylinder
5. Jackshaft Assembly
6. Upper Step
7. Aft Pushrod
8. Downlatch Cylinder
9. Door Seal
10. Sector Cable Housing
11. Forward Pushrod
12. Door Seal Inflation Tube
13. Door Swivel
14. Floor Swivel

Main Entrance Door
Figure 201.

MAIN ENTRANCE DOOR ELECTRICAL CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

(See Figure 1 and Figure 2)

The airstair door electrical control system consists of a pilots door safety switch, main door lock switches, inside and outside door switches, inside switch control relay and auxiliary hydraulic pump switches and control relay. The system is powered from the main dc bus through a 5A DOOR CONT circuit breaker, pilots circuit breaker panel. Moving DOOR SAFETY switch, located on the lower overhead panel, to SAFE causes the circuit between the power source and door control switches to open. When the DOOR SAFETY switch is set to UNSAFE, a door warning light on the master panel will come on. Door lock switches, located within the structure surrounding the door, are actuated by bayonets in the door lock mechanism. When the door is unlocked, the released lock switches will complete the power circuit between the door safety switch and the inside and outside door control switches. The outside door switch, mounted on the nose wheel well junction box, allows the door to be operated only when the door safety switch is set to UNSAFE and bayonets retracted.

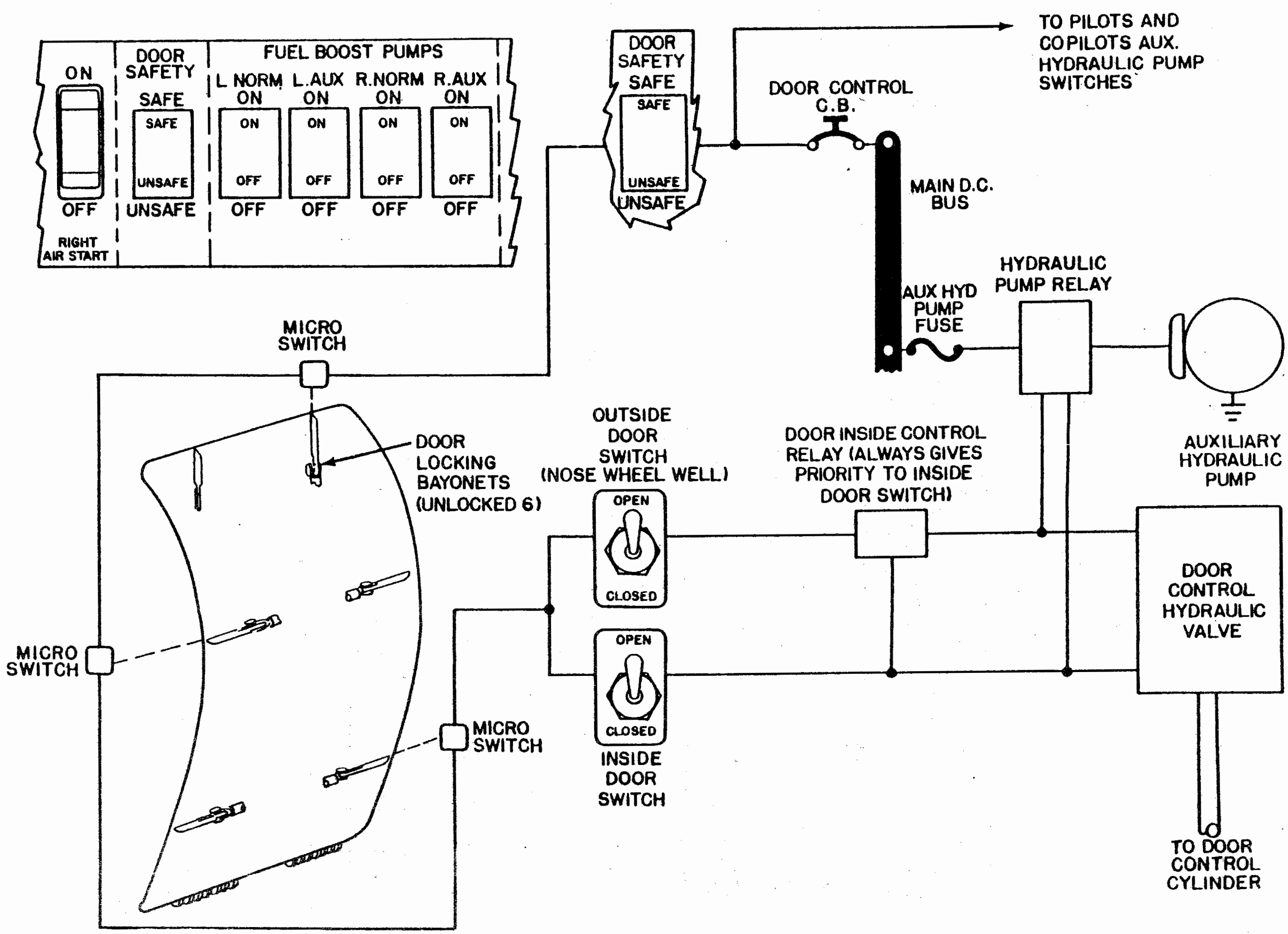
The inside AIRSTAIR SW, located on panel behind the pilots seat, makes door control possible without outside ground handling equipment. The inside door switch control relay in the mid relay box under floorboard No. 8 enables the inside AIRSTAIR SW to override the outside door switch.

Placing DOOR SAFETY switch to UNSAFE with main dc bus energized and door unlocked, will complete the door control power circuit (See Figure 2). Holding inside and outside door control switches to required position will automatically energize the auxiliary hydraulic pump in the nose wheel well and door and stairs selector valve. The pump then supplies auxiliary system pressure to energized open pressure port of the selector valve. The retraction mechanism, operated by the door and stair hydraulic cylinder in the hydraulic compartment, will open or close the door and fold or unfold the stairs. For security purposes on ground, a conventional key type lock is included in the push pull panel in outside door surface.

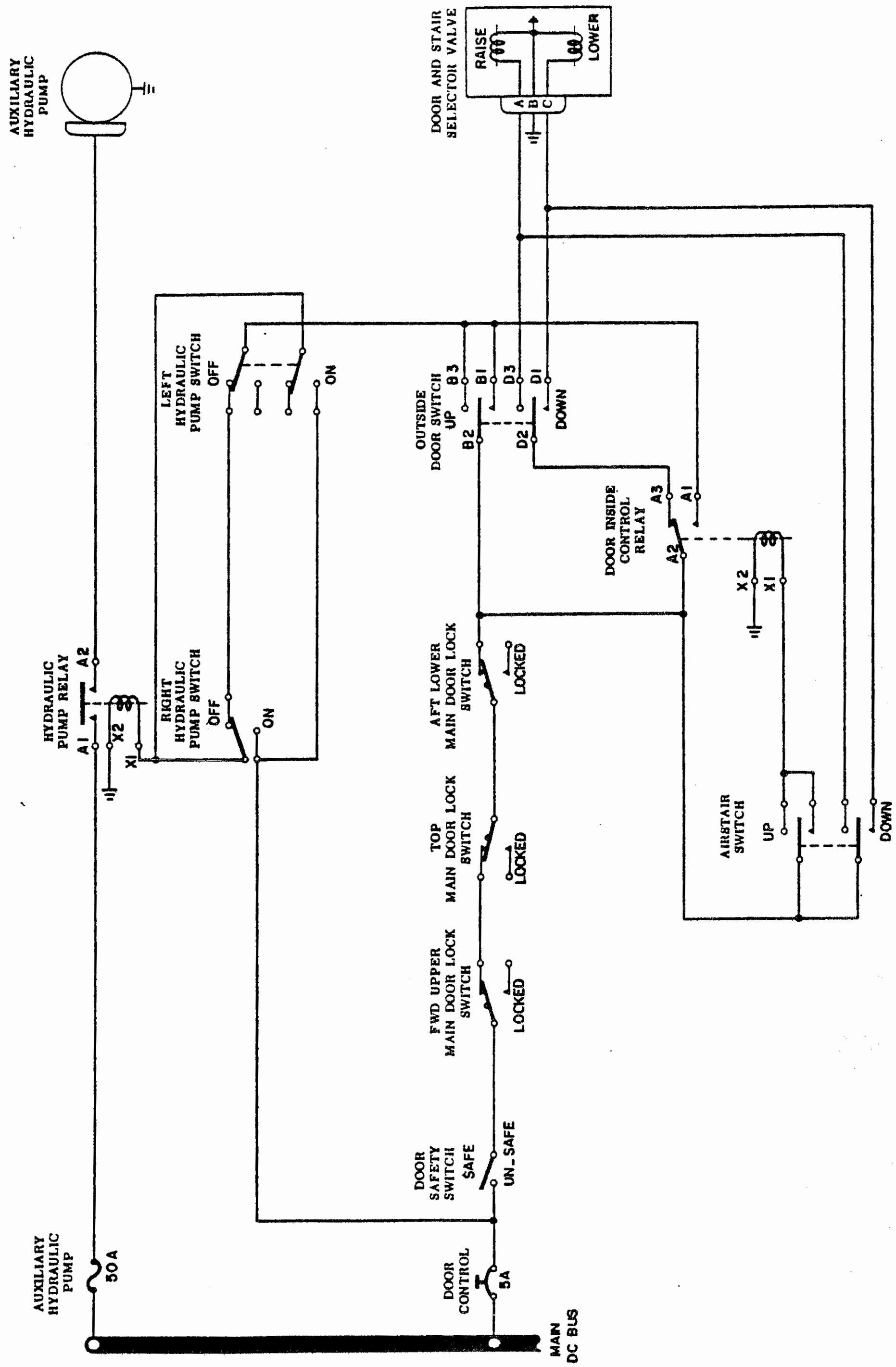
During an extension cycle, it is necessary to hold the AIRSTAIR SW in position. When switch is released it springs to OFF, de-energizing system.

During a retraction cycle, the AIRSTAIR SW is detented. When door reaches its closed position, operating primary door locking mechanism, it de-energizes the control system. The AIRSTAIR SW should then be set to OFF.

When an external source of power is used, the EXT PWR switch, located on the pilots overhead panel must be set to EXT PWR. If external power is not available, door may be closed by using aircraft battery. To close door from outside, turn on OUTSIDE BATT SW located on the outboard side of the left nacelle. The OUTSIDE DOOR SW in the nose wheel well then is placed in the CLOSE position. After the door is closed and locked, release the OUTSIDE DOOR SW switch, set OUTSIDE BATT SW to center position and close switch access cover.



Main Entrance Door Control System
Figure 1.



Main Entrance Door Control Schematic
Figure 2.

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BAGGAGE COMPARTMENT DOOR — DESCRIPTION / OPERATION

1. Description

(See Figure 1)

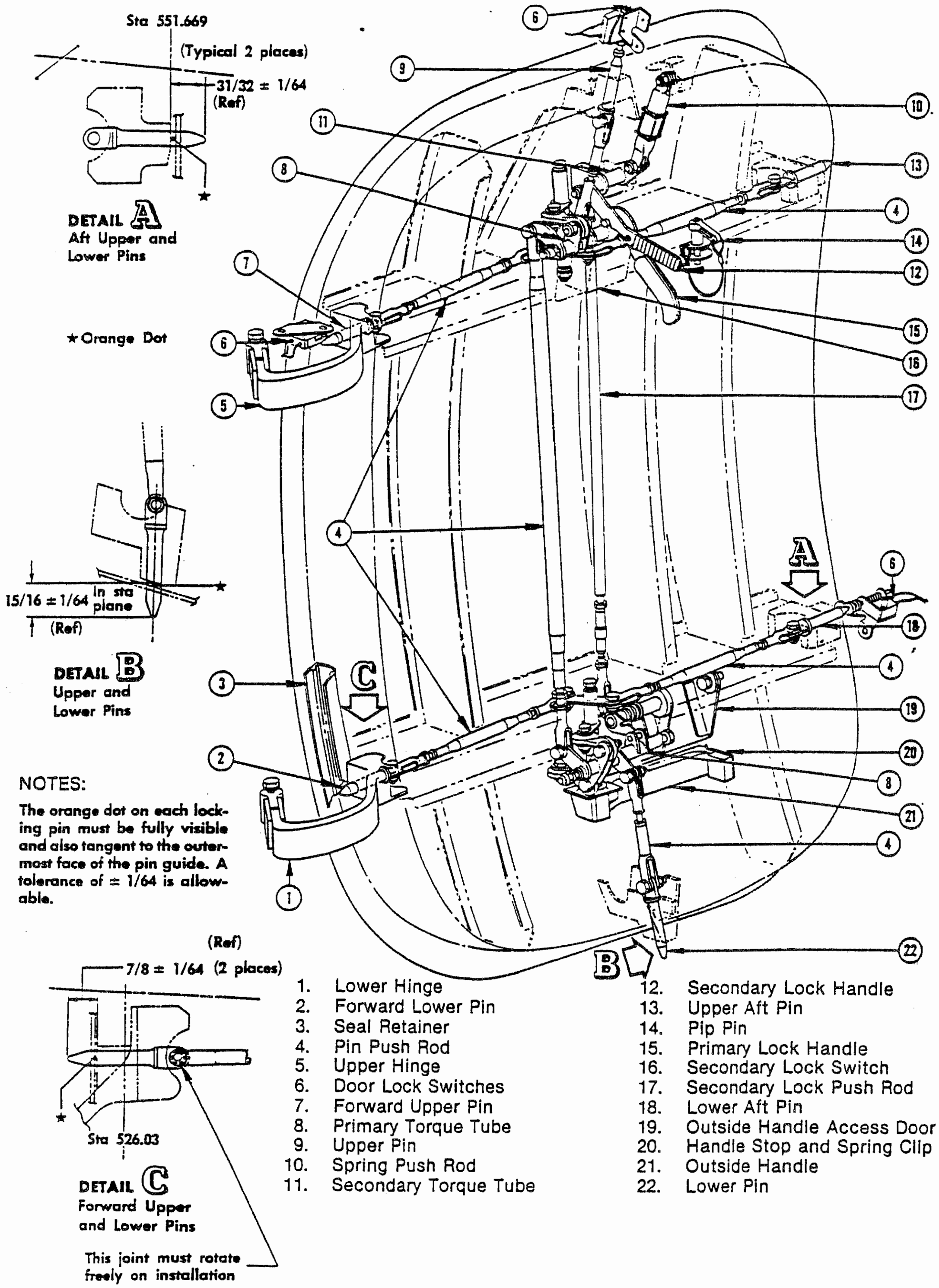
The baggage compartment door provides access for loading and unloading baggage and provides an emergency exit from rear section of the aircraft. It is located on the right side of the fuselage between Fuselage Station 525 and 553, consists of an outer frame, horizontal and vertical channel stiffeners and stressed skin covering. A manually controlled locking mechanism secures the door to surrounding structure. The door is hinged at two points on the forward side so that it travels out and forward when opened. A flush primary pull handle, located at the bottom of the door, is provided for outside operating of locking mechanism. The locking mechanism secures the door using six bayonets; two bayonets on either side, one on top and one on bottom. The inside primary handle is held in the closed position by the spring-loaded secondary latch handle and pin assembly. An inflatable seal is installed around door to maintain cabin pressurization. This seal is actuated by the forward lower bayonet through a door seal control valve.

There are three microswitches installed on the baggage door frame and one in the door locking mechanism. The three microswitches in the door frame are operated by the bayonets on the forward upper, top and rear lower portions of door. The door locking mechanism switch is actuated by using primary locking handle. These microswitches are connected with main door microswitches to form the door warning system.

The baggage door is locked by a pip pin. To lock door, insert pip pin through hole provided in the primary locking handle and through hole in the secondary locking handle, shown in Figure 1.

If the pip pin is installed in the baggage door secondary lock handle, the lock handle switch will be closed and an unsafe condition will be indicated by the lighting of the cabin open light. This is to ensure against flight operations with this pin in place. With the pin in place, emergency exit or entrance through the baggage door is impossible. This pin is only to be used to lock the baggage door handles when the aircraft is on the ground.

With the baggage door in the open position, ensure that the primary control handle is not rotated to extend the bayonets. If this occurs and the door is closed, the extended bayonets will strike the fuselage, denting the skin.



1. Lower Hinge
2. Forward Lower Pin
3. Seal Retainer
4. Pin Push Rod
5. Upper Hinge
6. Door Lock Switches
7. Forward Upper Pin
8. Primary Torque Tube
9. Upper Pin
10. Spring Push Rod
11. Secondary Torque Tube
12. Secondary Lock Handle
13. Upper Aft Pin
14. Pip Pin
15. Primary Lock Handle
16. Secondary Lock Switch
17. Secondary Lock Push Rod
18. Lower Aft Pin
19. Outside Handle Access Door
20. Handle Stop and Spring Clip
21. Outside Handle
22. Lower Pin

Baggage Compartment Door
Figure 1.

BAGGAGE COMPARTMENT DOOR — DESCRIPTION

1. **Description** (See Figure 1)

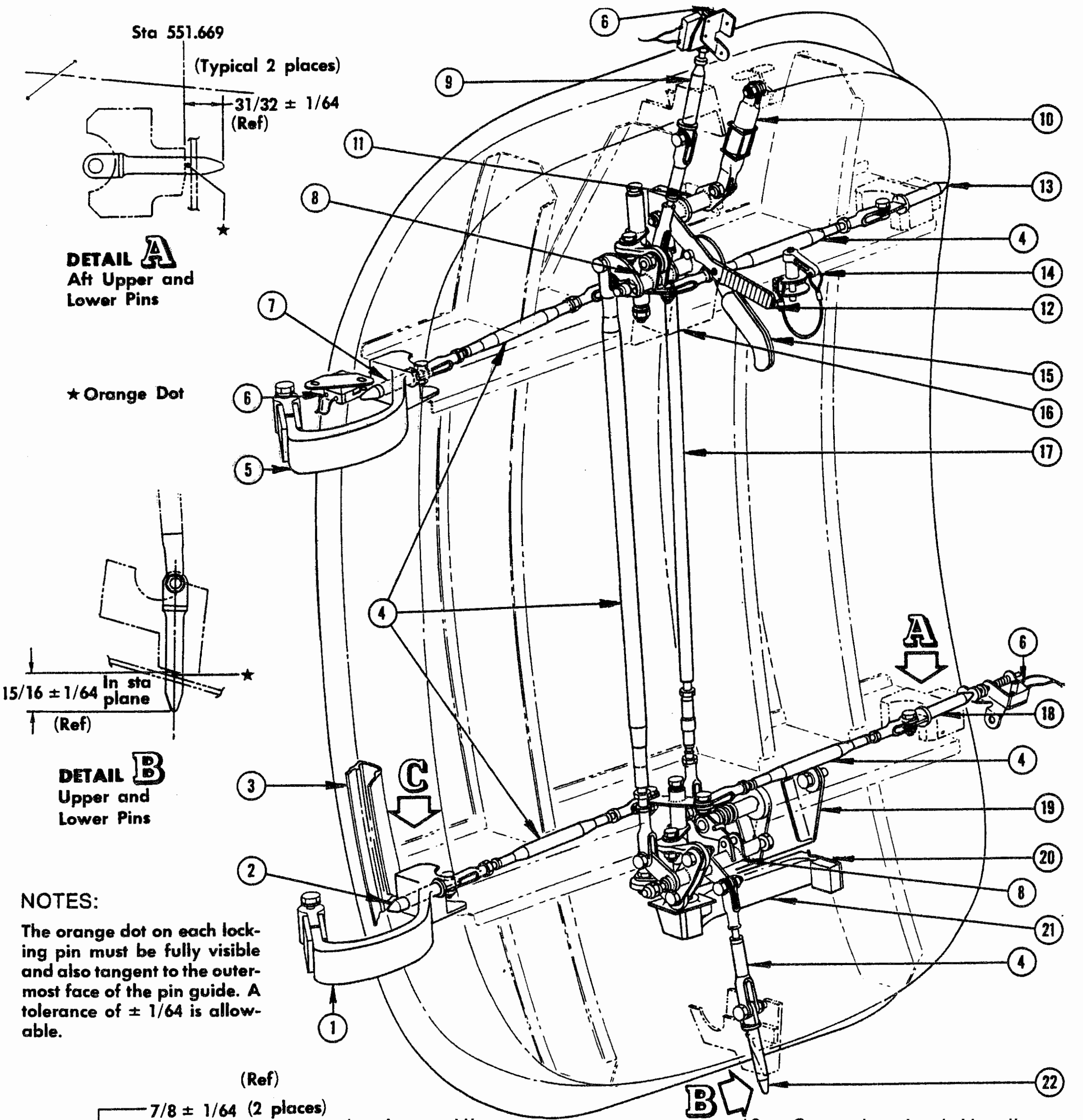
The baggage compartment door provides access for loading and unloading baggage and provides an emergency exit from rear section of the aircraft. It is located on the right side of the fuselage between Fuselage Station 525 and 553, consists of an outer frame, horizontal and vertical channel stiffeners and stressed skin covering. A manually controlled locking mechanism secures the door to surrounding structure. The door is hinged at two points on the forward side so that it travels out and forward when opened. A flush primary pull handle, located at the bottom of the door, is provided for outside operating of locking mechanism. The locking mechanism secures the door using six bayonets; two bayonets on either side, one on top and one on bottom. The inside primary handle is held in the closed position by the spring-loaded secondary latch handle and pin assembly. An inflatable seal is installed around door to maintain cabin pressurization. This seal is actuated by the forward lower bayonet through a door seal control valve.

There are three microswitches installed on the baggage door frame and one in the door locking mechanism. The three microswitches in the door frame are operated by the bayonets on the forward upper, top and rear lower portions of door. The door locking mechanism switch is actuated by using primary locking handle. These microswitches are connected with main door microswitches to form the door warning system.

The baggage door is locked by a pip pin. To lock door, insert pip pin through hole provided in the primary locking handle and through hole in the secondary locking handle, shown in Figure 3.

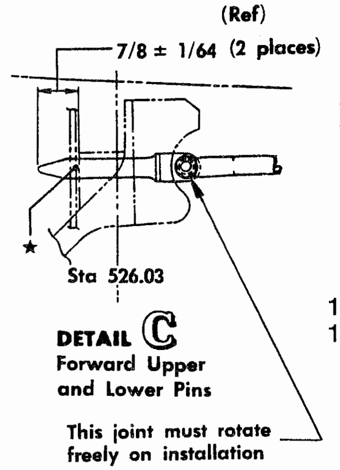
If the pip pin is installed in the baggage door secondary lock handle, the lock handle switch will be closed and an unsafe condition will be indicated by the lighting of the cabin open light. This is to ensure against flight operations with this pin in place. With the pin in place, emergency exit or entrance through the baggage door is impossible. This pin is only to be used to lock the baggage door handles when the aircraft is on the ground.

With the baggage door in the open position, ensure that the primary control handle is not rotated to extend the bayonets. If this occurs and the door is closed, the extended bayonets will strike the fuselage, denting the skin.



NOTES:

The orange dot on each locking pin must be fully visible and also tangent to the outermost face of the pin guide. A tolerance of ± 1/64 is allowable.



1. Lower Hinge
2. Forward Lower Pin
3. Seal Retainer
4. Pin Push Rod
5. Upper Hinge
6. Door Lock Switches
7. Forward Upper Pin
8. Primary Torque Tube
9. Upper Pin
10. Spring Push Rod
11. Secondary Torque Tube
12. Secondary Lock Handle
13. Upper Aft Pin
14. Pip Pin
15. Primary Lock Handle
16. Secondary Lock Switch
17. Secondary Lock Push Rod
18. Lower Aft Pin
19. Outside Handle Access Door
20. Handle Stop and Spring Clip
21. Outside Handle
22. Lower Pin

Baggage Compartment Door
Figure 1

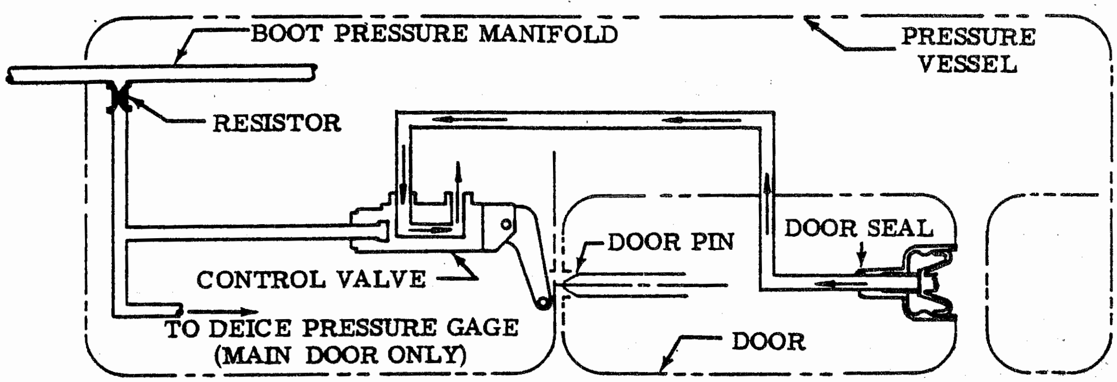
INFLATABLE DOOR SEALS SYSTEM — DESCRIPTION / OPERATION

1. **Description**

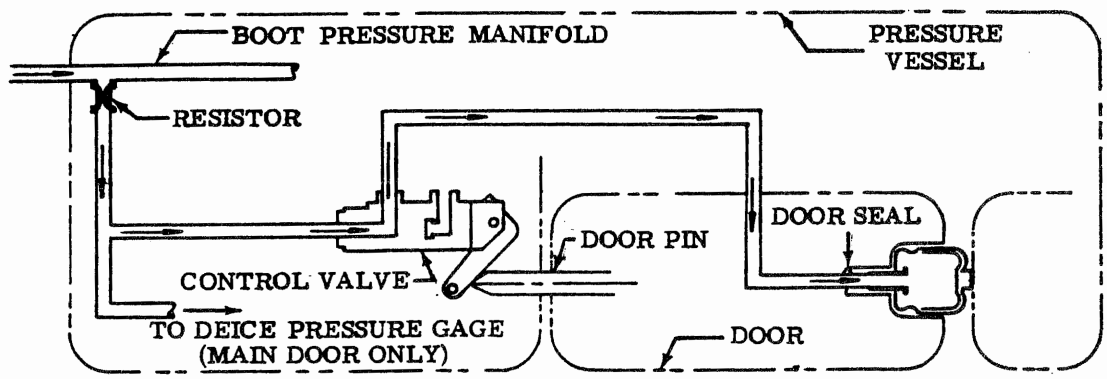
(See Figure 1)

The inflatable door seals on the airstair and baggage compartment doors are supplied with bleed air pressure from the pneumatic de-icing system. Inflatable seals for doors are made of natural or synthetic rubber compound, reinforced with knitted elastic fabric. They are entirely coated with neoprene rubber. The serrated free side of each seal is pushed out against the door coaming when systems are pressurized. Seals are inflated when seal valve of each door is actuated by lower forward bayonet of the locking mechanisms, when deicing manifold pressure is available.

Aircraft having ASC 235 installed are equipped with a check valve in the line to the control valve. This allows air pressure to inflate seal but not to escape through the same line in case of failure. Air can be discharged to ambient only through the door seal control valve.



DOOR PIN RETRACTED
SEAL DEFLATED



DOOR PIN ENGAGED
SEAL INFLATED

MAIN DOOR SHN.
BAGGAGE DOOR SIMILAR EXCEPT AS NOTED

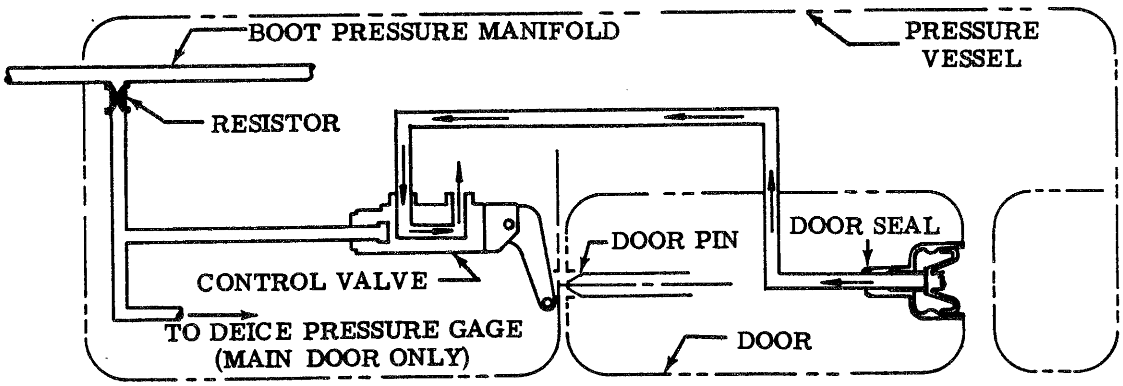
Inflatable Door Seal System
Figure 1.

INFLATABLE DOOR SEALS SYSTEM — DESCRIPTION

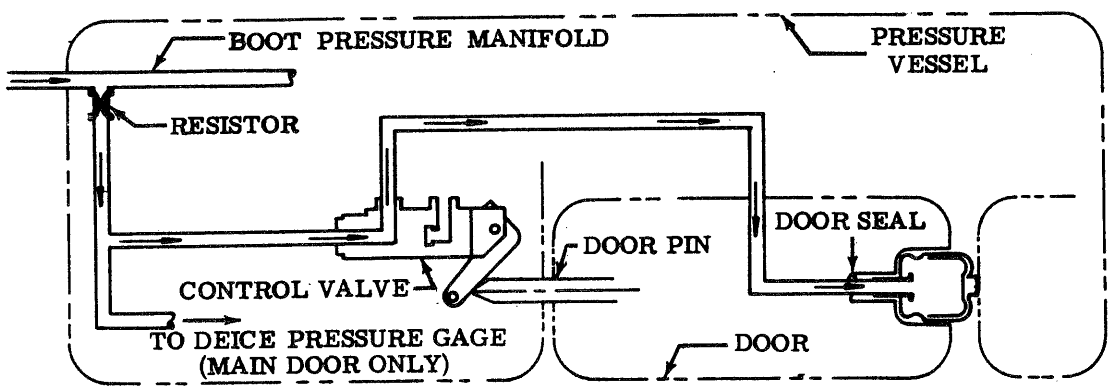
1. **Description** (See Figure 1)

The inflatable door seals on the airstair and baggage compartment doors are supplied with bleed air pressure from the pneumatic de-icing system. Inflatable seals for doors are made of natural or synthetic rubber compound, reinforced with knitted elastic fabric. They are entirely coated with neoprene rubber. The serrated free side of each seal is pushed out against the door coaming when systems are pressurized. Seals are inflated when seal valve of each door is actuated by lower forward bayonet of the locking mechanisms, when deicing manifold pressure is available.

Aircraft having ASC 235 installed are equipped with a check valve in the line to the control valve. This allows air pressure to inflate seal but not to escape through the same line in case of failure. Air can be discharged to ambient only through the door seal control valve.



DOOR PIN RETRACTED
SEAL DEFLATED



DOOR PIN ENGAGED
SEAL INFLATED

MAIN DOOR SHN.
BAGGAGE DOOR SIMILAR EXCEPT AS NOTED

Inflatable Door Seal System
Figure 1

INFLATABLE DOOR SEALS SYSTEM — MAINTENANCE PRACTICES

1. **Main Entrance and Baggage Door Inflatable Seal System — Operational Test**

(Aircraft not having ASC 235)

A. Create a pressure of 15 psi to 25 psi in the pneumatic boot system pressure manifold (as shown on the pressure gage) by accomplishing one of the following:

(1) Connect air pressure to pneumatic de-icing system upstream of the regulator in either nacelle.

(2) Run either engine, but door must be closed if left engine is run.

(3) Operate APU with full air output to cabin, and cabin and cockpit temperature controls set for full cooling.

B. Close door and operate handle to fully extend door locking pins into striker plates. Seal should inflate. Make visual inspection.

C. With door in closed position, operate handle to fully retract door locking pins. Seal should deflate.

Main Door Seal (Aircraft having ASC 235)

A. Create a pressure of 15 psi to 25 psi in the pneumatic boot system pressure manifold (as shown on the pressure gage) by accomplishing one of the following:

B. Close both doors and check that seals inflate.

C. Remove pressure from de-ice system manifold. Ensure that seals remain inflated.

D. Operate handle to fully retracted position. Seals should deflate.

Baggage Door Seal (Aircraft not having ASC 235)

A. Create a pressure of 15 psi to 25 psi in the pneumatic boot system pressure manifold (as shown on the pressure gage) by accomplishing one of the following:

B. Connect air pressure to pneumatic de-icing system upstream of the regulator in either nacelle.

C. Run either engine, but door must be closed if left engine is run.

D. Operate APU with full air output to cabin, and cabin and cockpit temperature controls set for full cooling.

E. Close door and operate handle to fully extend door locking pins into striker plates. Seal should inflate. Make visual inspection.

F. With door in closed position, operate handle to fully retract door locking pins. Seal should deflate.

Baggage Door Seal (Aircraft having ASC 235)

A. Create a pressure of 15 psi to 25 psi in the pneumatic boot system pressure manifold (as shown on the pressure gage) by accomplishing one of the following:

B. Close both doors and check that seals inflate.

C. Remove pressure from de-ice system manifold. Ensure that seals remain inflated.

D. Operate handle to fully retracted position. Seals should deflate.

2. **Main Entrance Door Seal — Removal / Installation**

(See Figure 201)

A. Removal

(1) Extend entrance door fully and disconnect forward pushrod (11) from bellcrank drive shaft and aft pushrod (7) from jackshaft.

(2) Disconnect both handrails at upper joints.

(3) Remove fairing that covers both swivels adjacent to hinge at forward end of door.

(4) Disconnect hydraulic and pneumatic flexible lines at upper swivel.

(5) Install protective caps on lines and ports of swivels.

(6) Pull piano wire out of lower hinge on top step and fold step up.

(7) Remove sector housing cover and fairing that runs the length of leading edge of door.

(8) Work seal out of seal retaining channel using a blunt piece of micarta or equivalent.

(9) Remove seal inflation tube from pneumatic line on door.

(10) Remove door seal.

B. Installation

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(1) Clean door mating surface with MEK or TOLUENE.

(2) Wipe off solvent immediately with clean lint free cloth.

(3) Apply one uniform brush coat of E.C.-1300-L, 1357 or equivalent to door channel and door seal.

(4) Allow sufficient time for adhesive to become tacky, but not transferable to the knuckle, when touched.

NOTE: DO NOT STRETCH when parts are joined.

(5) Place inflatable door seal around door so that inflation tube on seal lines up with air line indoor. Connect inflation tube to air line.

CAUTION: DO NOT USE ANY SHARP INSTRUMENTS ON THE INFLATABLE SEAL WHEN WORKING SEAL INTO CHANNEL.

(6) Work seal into channel with a blunt piece of micarta or its equivalent.

(7) Apply hand pressure all around seal.

(8) Allow bonded assembly to dry for minimum of 30 minutes before inflation.

(9) Fold top step down and install hinge pin (piano wire) into hinge.

(10) Remove protective caps and install pneumatic and hydraulic lines on upper swivel.

(11) Connect both handrails.

(12) Connect forward pushrod (11) to bellcrank drive shaft and aft pushrod (7) to jack shaft.

(13) Energize main and essential dc buses.

(14) Provide at least 15 psi of pressure in the de-icing system pressure manifold (as read on the cockpit gage) by connecting air supply to pressure manifold in nacelle or operating APU with air on and set cabin and cockpit temperature controls to full cooling.

(15) Raise and lock entrance door using the inside AIRSTAIR switch.

(16) Observe that door seal inflates properly when bayonets are extended. Check that seal is not twisted and contacts door frame properly around entire periphery.

(17) Open door and observe that door seal deflates properly when the bayonets are pulled.

(18) Remove air supply and electrical power if no longer required.

(19) Inspect area for foreign objects, security of all attachments.

(20) Close all panels and access.

3. Main Entrance Door Seal Air Valve — Removal / Installation

A. Removal

(1) Gain access to entrance door air seal valve, at extreme aft end of pilots console, forward of Fuselage Station 133.

(2) Remove pilots seat, aft side plate of console, and as many top console panels as required to gain access to valve.

(3) Disconnect pneumatic lines at air valve. Install protective caps on lines and ports.

(4) Remove air valve and retain hardware.

B. Installation

(1) Install air valve using hardware previously removed.

(2) Remove protective caps and connect pneumatic lines on valve.

(3) Energize main and essential dc buses.

(4) Provide at least 15 psi of pressure in the de-icing system pressure manifold (as read on cockpit gage) by connecting air supply to pressure manifold in nacelle or operating APU with air on and set cabin and cockpit temperature control to full cooling.

(5) Raise and lock entrance door using the inside AIRSTAIR switch.

(6) Observe that door seal inflates properly when bayonets are extended.

(7) Open door and observe that door seal deflates properly when the bayonets are pulled.

(8) Remove air supply and electrical power if no longer required.

(9) Inspect area for foreign objects, security of all attachments.

(10) Close all panels and accesses.

(11) Install pilots seat.

4. Baggage Door Seal — Removal / Installation

(See Figure 202)

A. Removal

(1) Open baggage door.

(2) Remove access cover on inside of door below upper hinge.

(3) Remove seal inflation tube from pneumatic line on door.

(4) Remove door seat.

B. Installation

(1) Clean door mating surface with MEK or TOLUENE.

(2) Wipe off solvent immediately with clean lint free cloth.

(3) Apply one uniform brush coat of E.C.-1300-L, 1357 or equivalent to door channel and door seal.

(4) Allow sufficient time for adhesive to become tacky, but not transferable to the knuckle, when touched.

NOTE: DO NOT STRETCH when parts are joined.

(5) Place inflatable door seal around door so that inflation tube on seal lines up with air line indoor. Connect inflation tube to air line.

CAUTION: DO NOT USE ANY SHARP INSTRUMENTS ON THE INFLATABLE SEAL WHEN WORKING SEAL INTO CHANNEL.

(6) Work seal into channel with a blunt piece of micarta or its equivalent.

(7) Apply hand pressure all around seal.

(8) Allow bonded assembly to dry for minimum of 30 minutes before inflation.

(9) Connect inflation tube to pneumatic line.

(10) Provide 15 psi pressure in the de-icing system pressure manifold (as read on the cockpit gage) by connecting air supply to pressure manifold in nacelle or operating APU with air on and set cabin and cockpit temperature controls to full cooling.

(11) Close and lock baggage door from inside of aircraft.

(12) Observe that door seal inflates properly when bayonets are extended. Check that seal is not twisted and contacts door frame properly around entire periphery.

(13) Open door and observe that door seal deflates properly when bayonets are pulled.

(14) Secure air supply.

(15) Inspect area for foreign objects, security of all attachments.

(16) Close all panels and access.

5. Baggage Door Seal Air Valve — Removal / Installation

A. Removal

NOTE: Air valve is located immediately forward of baggage door in lavatory compartment.

(1) Gain access to baggage door seal air valve by removing paneling or access cover on forward side of bulkhead adjacent to baggage door lower hinge.

(2) Disconnect pneumatic lines at air valve; remove air valve.

(3) Install protective caps on lines and ports.

B. Installation

(1) Remove protective caps from lines and ports.

(2) Install air valve on structure; connect pneumatic lines to air valve.

(3) Provide 15 psi pressure in the de-icing system pressure manifold (as read on the cockpit gage) by connecting air supply to pressure manifold in nacelle or operating APU with air on and set cabin and cockpit temperature controls to full cooling.

(4) Close and lock baggage door from inside aircraft.

(5) Check that door seal inflates properly when bayonets are extended; check that seal is not twisted and contacts door frame properly around entire periphery.

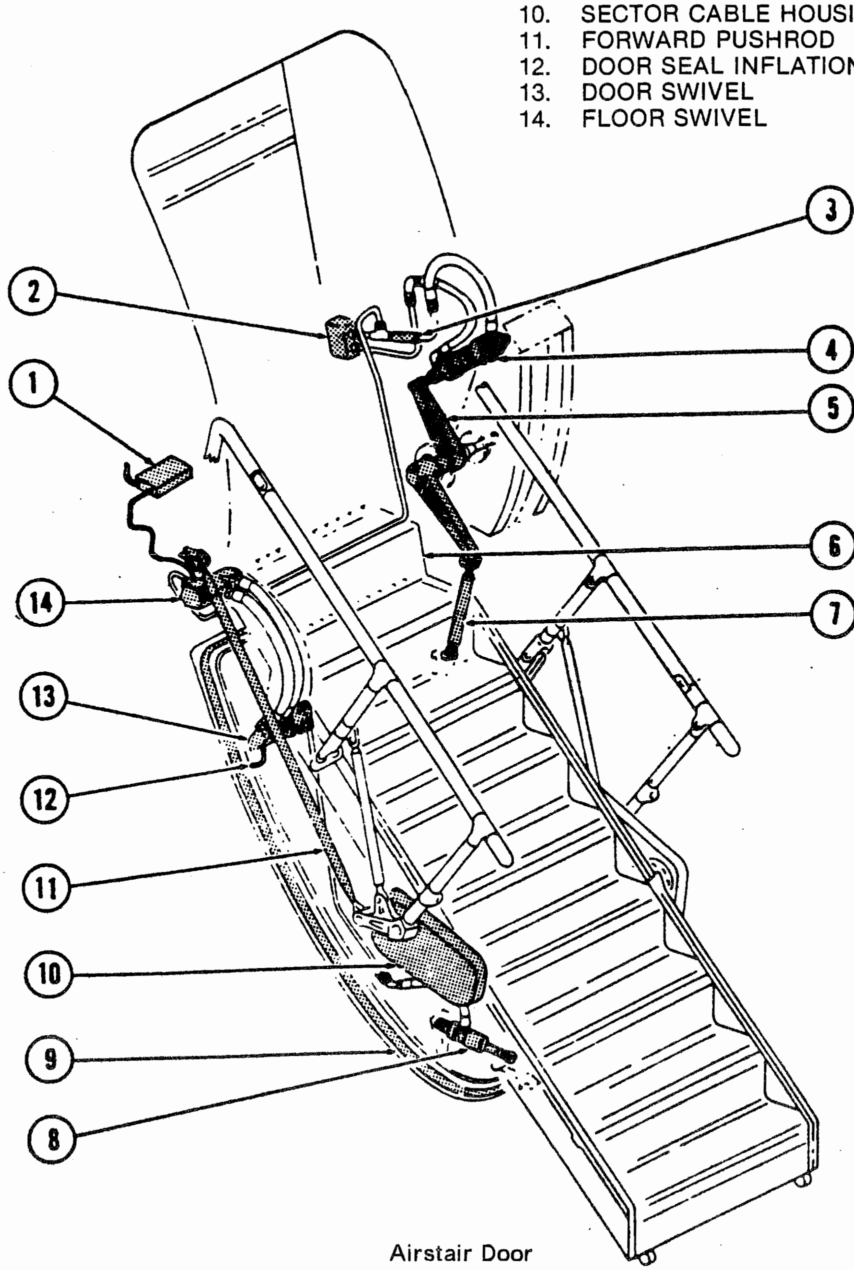
(6) Open door and check that door seal deflates properly when the bayonets are pulled.

(7) Turn off and disconnect air (or shut down APU).

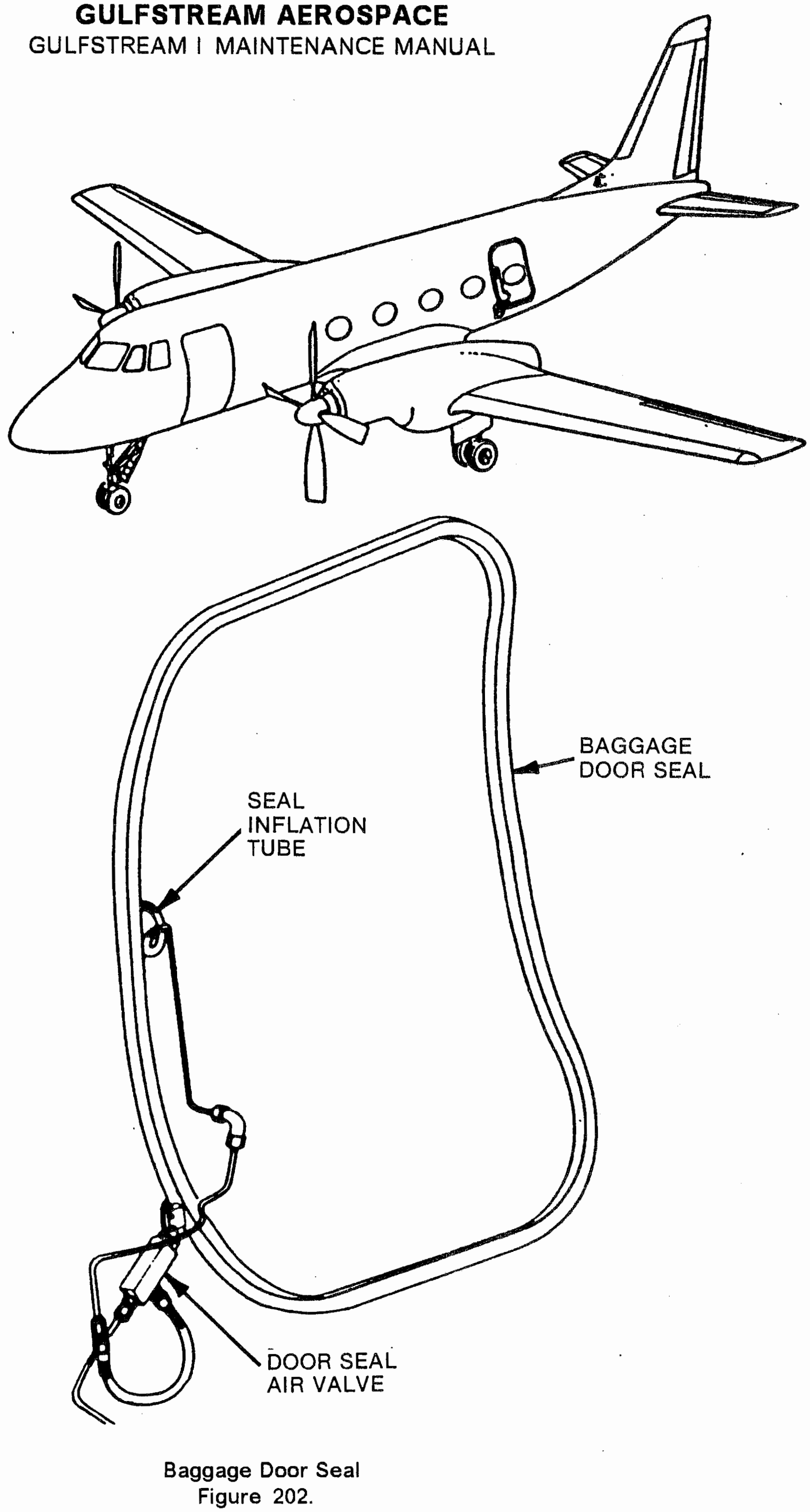
(8) Inspect area for foreign objects, security of all attachments.

(9) Close access.

1. ENTRANCE DOOR SEAL CONTROL VALVE
2. ENTRANCE DOOR SELECTOR VALVE
3. RESTRICTOR
4. ACTUATING CYLINDER
5. JACKSHAFT ASSEMBLY
6. UPPER STEP
7. AFT PUSHROD
8. DOWNLATCH CYLINDER
9. DOOR SEAL
10. SECTOR CABLE HOUSING
11. FORWARD PUSHROD
12. DOOR SEAL INFLATION TUBE
13. DOOR SWIVEL
14. FLOOR SWIVEL



Airstair Door
Figure 201.



Baggage Door Seal
Figure 202.

INFLATABLE DOOR SEALS — MAINTENANCE PRACTICES

1. **Main Door Seal — Checkout** (Aircraft not having ASC 235)
   A. Create a pressure of 15 psi to 25 psi in the pneumatic boot system pressure manifold (as shown on the pressure gage) by accomplishing one of the following:
      (1) Connect air pressure to pneumatic de-icing system upstream of the regulator in either nacelle.
      (2) Run either engine, but door must be closed if left engine is run.
      (3) Operate APU with full air output to cabin, and cabin and cockpit temperature controls set for full cooling.
   B. Close door and operate handle to fully extend door locking pins into striker plates. Seal should inflate. Make visual inspection.
   C. With door in closed position, operate handle to fully retract door locking pins. Seal should deflate.

2. **Main Door Seal — Cheokout** (Aircraft having ASC 235)
   A. Comply with Step 1.A. above.
   B. Close both doors and check that seals inflate.
   C. Remove pressure from de-ice system manifold. Ensure that seals remain inflated.
   D. Operate handle to fully retracted position. Seals should deflate.

3. **Baggage Door Seal — Checkout** (Aircraft not having ASC 235)
   A. See Steps 1.A. — 1.C. above for procedure.

4. **Baggage Door Seal — Checkout** (Aircraft having ASC 235)
   A. See Steps 2.A. — 2.D. above for procedure.

5. **Entrance Door Seal — Removal / Installation** (See Figure 201)
   A. Removal
      (1) Extend entrance door fully and disconnect forward pushrod (11) from bellcrank drive shaft and aft pushrod (7) from jackshaft.
      (2) Disconnect both handrails at upper joints.
      (3) Remove fairing that covers both swivels adjacent to hinge at forward end of door.
      (4) Disconnect hydraulic and pneumatic flexible lines at upper swivel.
      (5) Install protective caps on lines and ports of swivels.
      (6) Pull piano wire out of lower hinge on top step and fold step up.
      (7) Remove sector housing cover and fairing that runs the length of leading edge of door.
      (8) Work seal out of seal retaining channel using a blunt piece of micarta or equivalent.
      (9) Remove seal inflation tube from pneumatic line on door.
      (10) Remove door seal.
   B. Installation

      WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

      (1) Clean door mating surface with MEK or TOLUENE.
      (2) Wipe off solvent immediately with clean lint free cloth.
      (3) Apply one uniform brush coat of E.C.-1300-L, 1357 or equivalent to door channel and door seal.
      (4) Allow sufficient time for adhesive to become tacky, but not transferable to the knuckle, when touched.

      NOTE: DO NOT STRETCH when parts are joined.

      (5) Place inflatable door seal around door so that inflation tube on seal lines up with air line indoor. Connect inflation tube to air line.

      CAUTION: DO NOT USE ANY SHARP INSTRUMENTS ON THE INFLATABLE SEAL WHEN WORKING SEAL INTO CHANNEL.

      (6) Work seal into channel with a blunt piece of micarta or its equivalent.
      (7) Apply hand pressure all around seal.
      (8) Allow bonded assembly to dry for minimum of 30 minutes before inflation.
      (9) Fold top step down and install hinge pin (piano wire) into hinge.
      (10) Remove protective caps and install pneumatic and hydraulic lines on upper swivel.

(11) Connect both handrails.
(12) Connect forward pushrod (11) to bellcrank drive shaft and aft pushrod (7) to jack shaft.
(13) Energize main and essential dc buses.
(14) Provide at least 15 psi of pressure in the de-icing system pressure manifold (as read on the cockpit gage) by connecting air supply to pressure manifold in nacelle or operating APU with air on and set cabin and cockpit temperature controls to full cooling.
(15) Raise and lock entrance door using the inside AIRSTAIR switch.
(16) Observe that door seal inflates properly when bayonets are extended. Check that seal is not twisted and contacts door frame properly around entire periphery.
(17) Open door and observe that door seal deflates properly when the bayonets are pulled.
(18) Remove air supply and electrical power if no longer required.
(19) Close all panels and access.

6. **Entrance Door Seal Air Valve — Removal / Installation**

A. Removal

(1) Gain access to entrance door air seal valve, at extreme aft end of pilots console, forward of Fuselage Station 133.
(2) Remove pilots seat, aft side plate of console, and as many top console panels as required to gain access to valve.
(3) Disconnect pneumatic lines at air valve. Install protective caps on lines and ports.
(4) Remove air valve and retain hardware.

B. Installation

(1) Install air valve using hardware previously removed.
(2) Remove protective caps and connect pneumatic lines on valve.
(3) Energize main and essential dc buses.
(4) Provide at least 15 psi of pressure in the de-icing system pressure manifold (as read on cockpit gage) by connecting air supply to pressure manifold in nacelle or operating APU with air on and set cabin and cockpit temperature control to full cooling.
(5) Raise and lock entrance door using the inside AIRSTAIR switch.
(6) Observe that door seal inflates properly when bayonets are extended.
(7) Open door and observe that door seal deflates properly when the bayonets are pulled.
(8) Remove air supply and electrical power if no longer required.
(9) Close all panels and accesses.
(10) Install pilots seat.

7. **Baggage Door Seal — Removal / Installation**

A. Removal

(1) Open baggage door.
(2) Remove access cover on inside of door below upper hinge.
(3) Remove seal inflation tube from pneumatic line on door.
(4) Remove door seat.

B. Installation

(1) Clean door mating surface with MEK or TOLUENE.
(2) Wipe off solvent immediately with clean lint free cloth.
(3) Apply one uniform brush coat of E.C.-1300-L, 1357 or equivalent to door channel and door seal.
(4) Allow sufficient time for adhesive to become tacky, but not transferable to the knuckle, when touched.

NOTE: DO NOT STRETCH when parts are joined.

(5) Place inflatable door seal around door so that inflation tube on seal lines up with air line indoor. Connect inflation tube to air line.

CAUTION: DO NOT USE ANY SHARP INSTRUMENTS ON THE INFLATABLE SEAL WHEN WORKING SEAL INTO CHANNEL.

(6) Work seal into channel with a blunt piece of micarta or its equivalent.
(7) Apply hand pressure all around seal.
(8) Allow bonded assembly to dry for minimum of 30 minutes before inflation.
(9) Connect inflation tube to pneumatic line.
(10) Provide 15 psi pressure in the de-icing system pressure manifold (as read on the cockpit gage) by connecting air supply to pressure manifold in nacelle or operating APU with air on and set cabin and cockpit temeprature controls to full cooling.

(11) Close and lock baggage door from inside of aircraft.
(12) Observe that door seal inflates properly when bayonets are extended. Check that seal is not twisted and contacts door frame properly around entire periphery.
(13) Open door and observe that door seal deflates properly when bayonets are pulled.
(14) Secure air supply.
(15) Close all panels and access.

8. **Baggage Door Seal Air Valve — Removal / Installation**

A. Removal

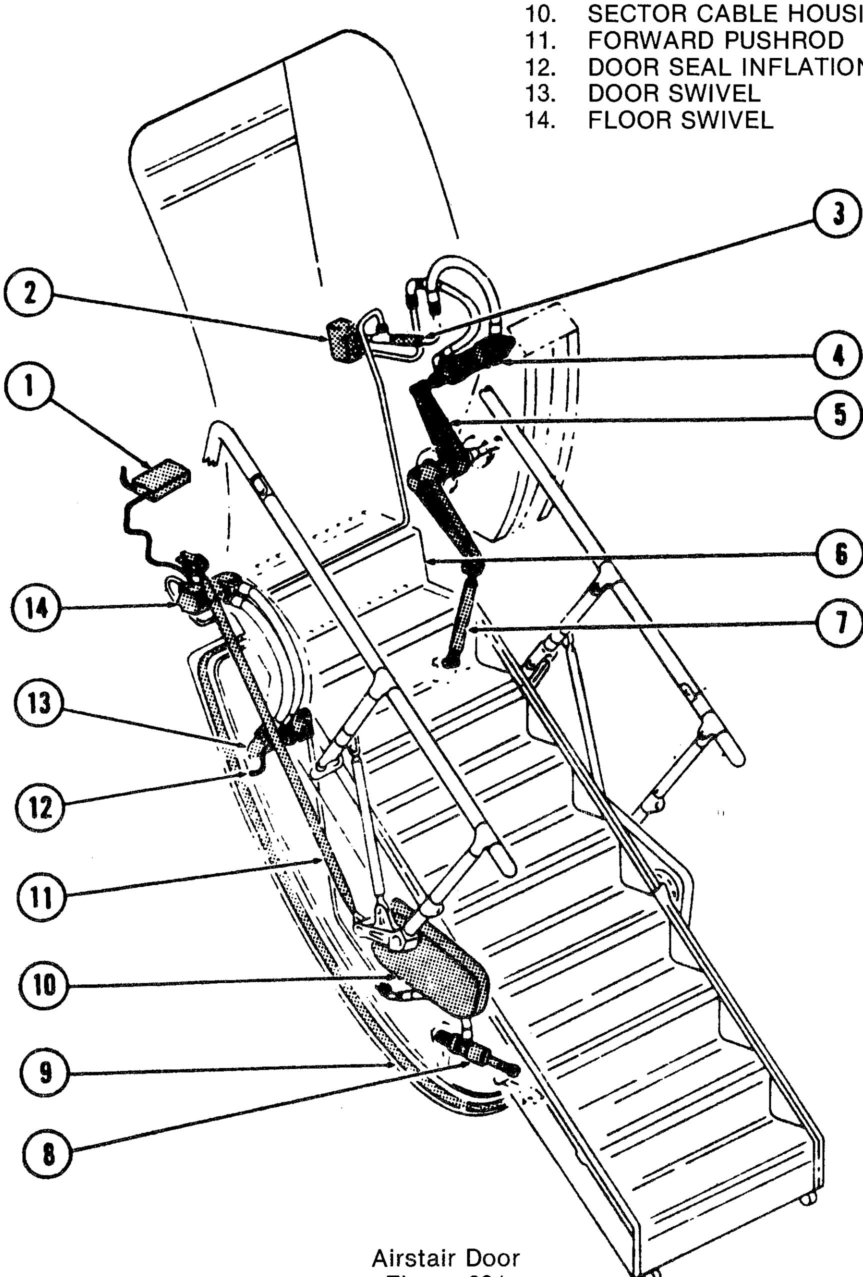
NOTE: Air valve is located immediately forward of baggage door in lavatory compartment.

(1) Gain access to baggage door seal air valve by removing paneling or access cover on forward side of bulkhead adjacent to baggage door lower hinge.
(2) Disconnect pneumatic lines at air valve; remove air valve.
(3) Install protective caps on lines and ports.

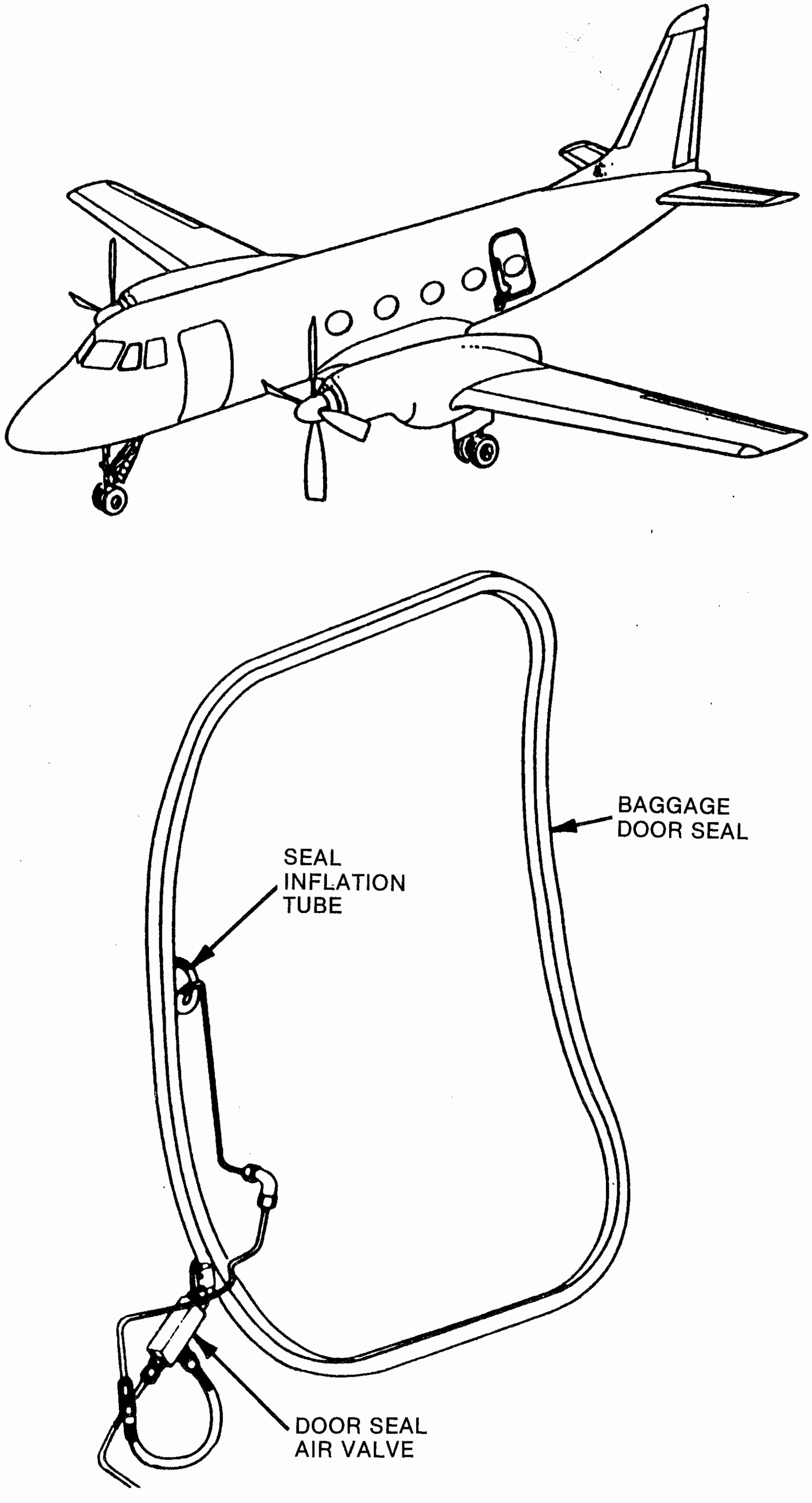
B. Installation

(1) Remove protective caps from lines and ports.
(2) Install air valve on structure; connect pneumatic lines to air valve.
(3) Provide 15 psi pressure in the de-icing system pressure manifold (as read on the cockpit gage) by connecting air supply to pressure manifold in nacelle or operating APU with air on and set cabin and cockpit temperature controls to full cooling.
(4) Close and lock baggage door from inside aircraft.
(5) Check that door seal inflates properly when bayonets are extended; check that seal is not twisted and contacts door frame properly around entire periphery.
(6) Open door and check that door seal deflates properly when the bayonets are pulled.
(7) Turn off and disconnect air (or shut down APU).
(8) Close access.

1. ENTRANCE DOOR SEAL CONTROL VALVE
2. ENTRANCE DOOR SELECTOR VALVE
3. RESTRICTOR
4. ACTUATING CYLINDER
5. JACKSHAFT ASSEMBLY
6. UPPER STEP
7. AFT PUSHROD
8. DOWNLATCH CYLINDER
9. DOOR SEAL
10. SECTOR CABLE HOUSING
11. FORWARD PUSHROD
12. DOOR SEAL INFLATION TUBE
13. DOOR SWIVEL
14. FLOOR SWIVEL



Airstair Door
Figure 201



Baggage Door Seal
Figure 202

DOOR WARNING SYSTEM — DESCRIPTION / OPERATION

1. **Description** (See Figure 1)

The door warning system consists of three primary lock switches and one secondary lock switch on the main entrance door, three primary lock switches and one secondary lock switch on the baggage compartment door, and the pilots DOOR SAFETY switch. The system is powered through the 5A DOOR WARN circuit breaker from the main dc bus. Three lock switches for the main entrance door are located within the structure around the door at the forward upper, top aft and aft lower bayonets. The secondary lock switch for the main door is located below secondary lock handle. The three baggage door lock switches are located within the door coaming at the top, aft lower and upper forward bayonets. The secondary baggage door lock switch is located below the secondary lock handle. The pilots DOOR SAFETY switch is located on the lower overhead panel.

If the pip pin is installed in the baggage door secondary lock handle, the secondary lock handle switch is closed, indicating an unsafe condition. This ensures against flight operations with pin in place. Emergency entrance through the baggage door is impossible. This pin is only to be used to lock the baggage door handles when the aircraft is on the ground.

2. **Operation** (See Figure 2)

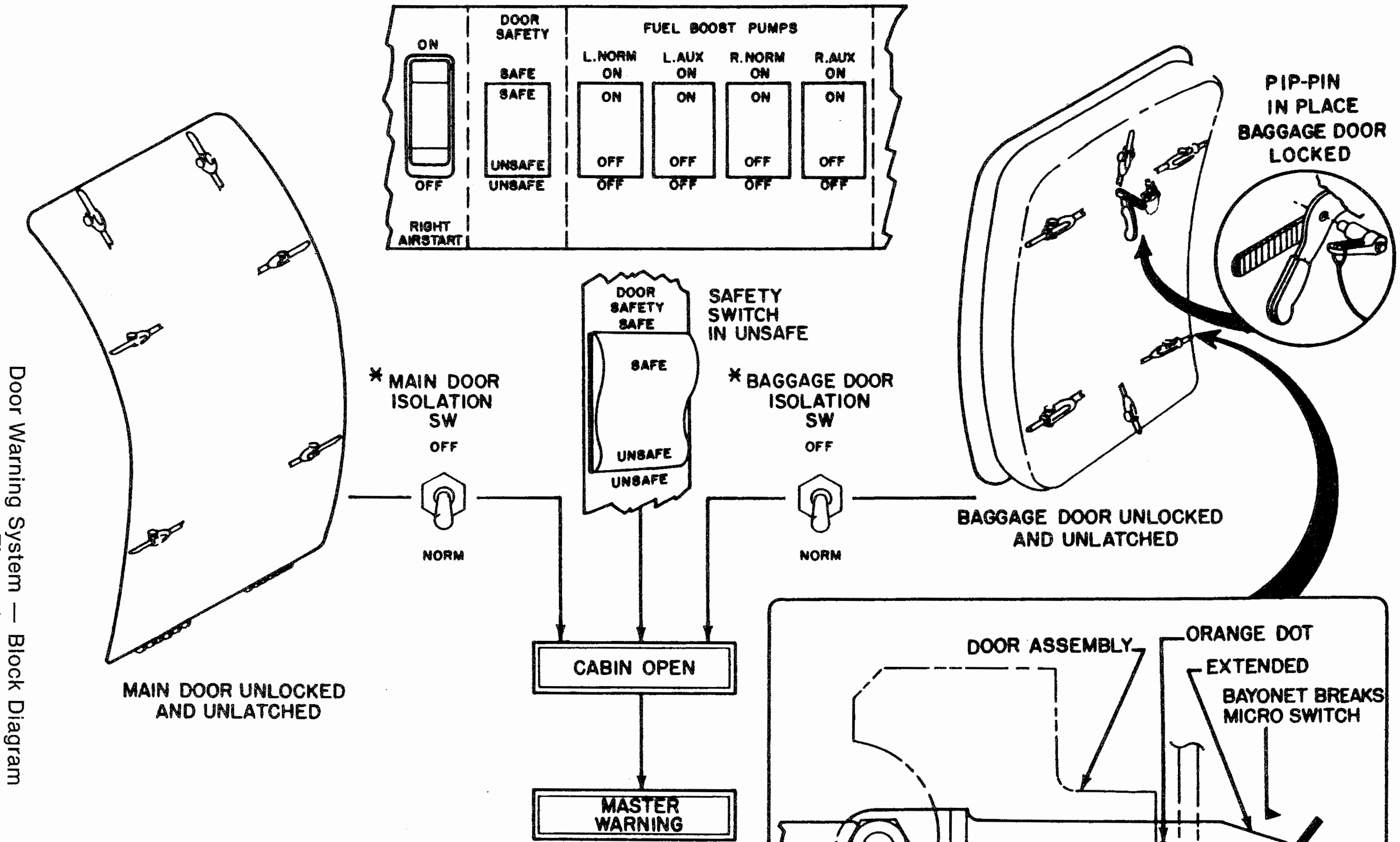
All door warning system switches are connected in parallel in the normally closed position. When one of the door lock switches or secondary switches are not in LOCKED position (bayonets are not extended and locking handle latch not secure), or if the pilots DOOR SAFETY switch is not in SAFE position, the CABIN OPEN light on master caution panel will be on. The light is turned off if all of the following conditions are met:

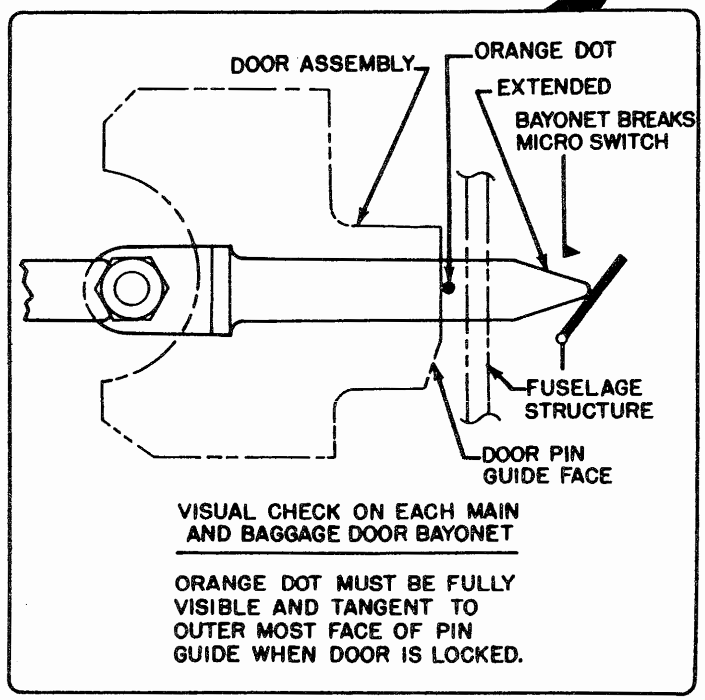
A. The airstair door is up and the bayonets extended.
B. The airstair door locking handle latch is secure.
C. The baggage door is closed and its bayonets are extended.
D. The baggage door locking handle latch is secure.
E. The DOOR SAFETY switch (lower overhead panel) is in the SAFE position.
F. The pip pin is removed from the baggage door latch.

On Aircraft having ASC 219 installed, the door warning light will come on in the event of an electrical malfunction in the door warning circuitry. (See Figure 3).

Aircraft having ASC 110 contain two additional switches (See Figure 4 and 5): the MAIN DOOR WARNING ISOLATION SWITCH and the BAGGAGE DOOR WARNING ISOLATION SWITCH. These are guarded switches located on the outboard side of left Fuselage Station 133 relay panel (located directly behind the pilots seat). They enable crew to ascertain and isolate main or baggage door warning signal for easier identification of an unlocked door or malfunctioning switch. Switches are guarded to their normal position. Placing either switch OFF eliminates that set of switches from affecting the door warning lights for trouble shooting purposes.

WARNING: THE ABOVE ISOLATION SWITCHES SHOULD ALWAYS BE KEPT IN THE NORMAL (GUARD DOWN) POSITION UNLESS ACTUALLY CHECKING THE CIRCUIT FOR TROUBLE.





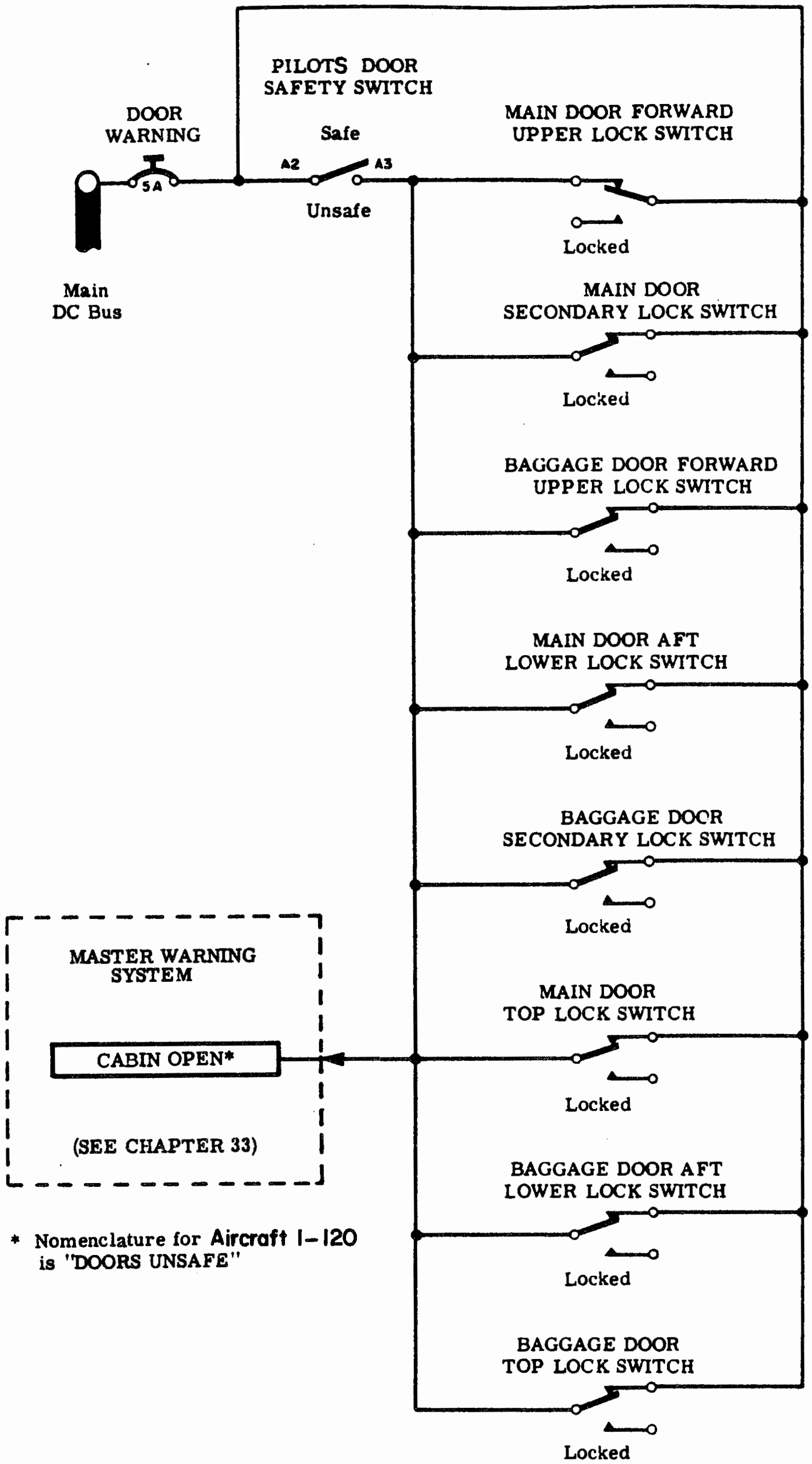
NOTE:

To put cabin open light out the following must be accomplished:

1. Main door locked and latched.
2. Baggage door locked and latched.
3. Pip-pin removed from baggage door.
4. Door safety switch in safe position.
5. If light stays on or comes on in aircraft having ASC 110. Use isolation switches to determine open door or area of malfunction.

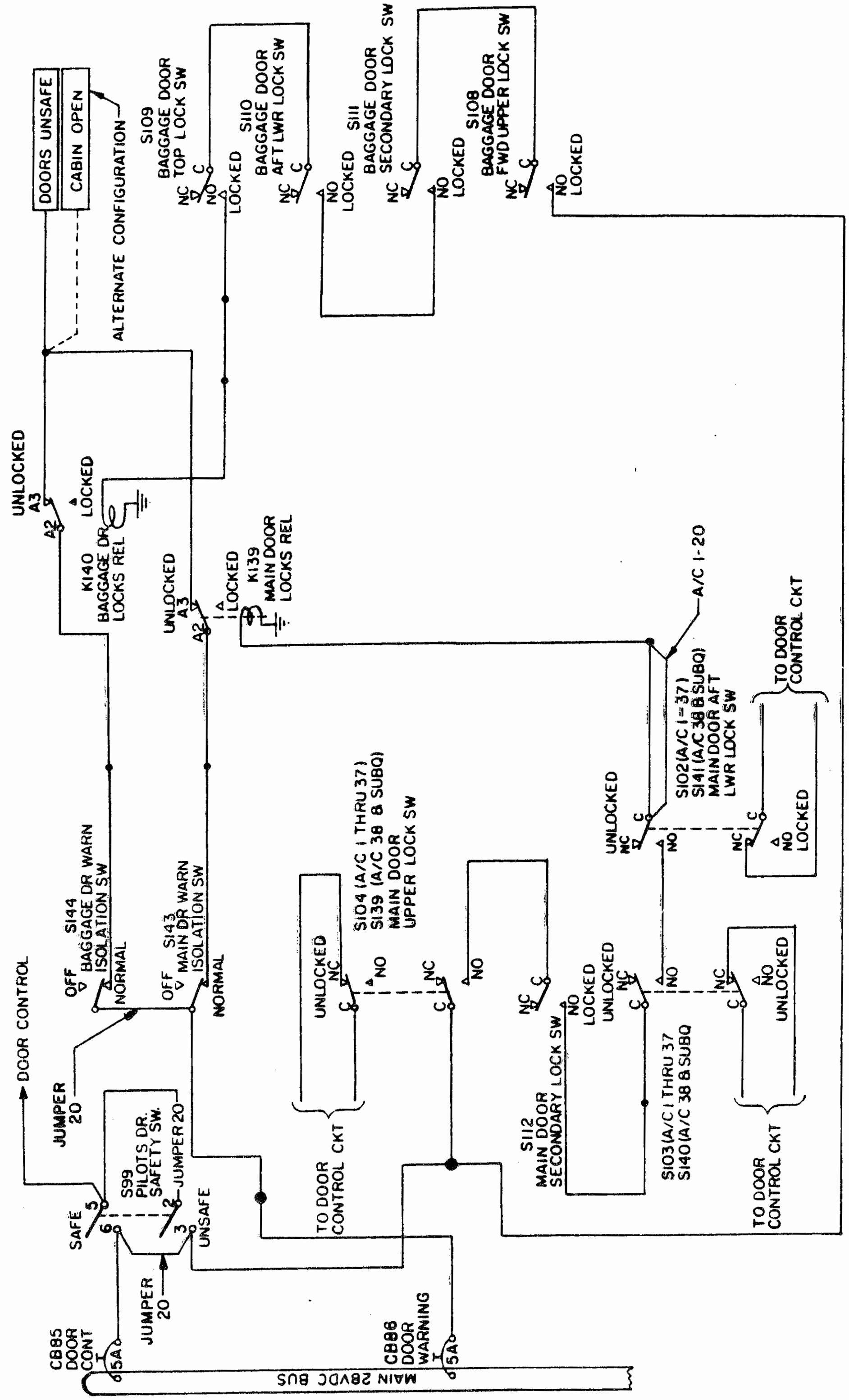
* (On aircraft having ASC 110, only).

Door Warning System — Block Diagram
Figure 1

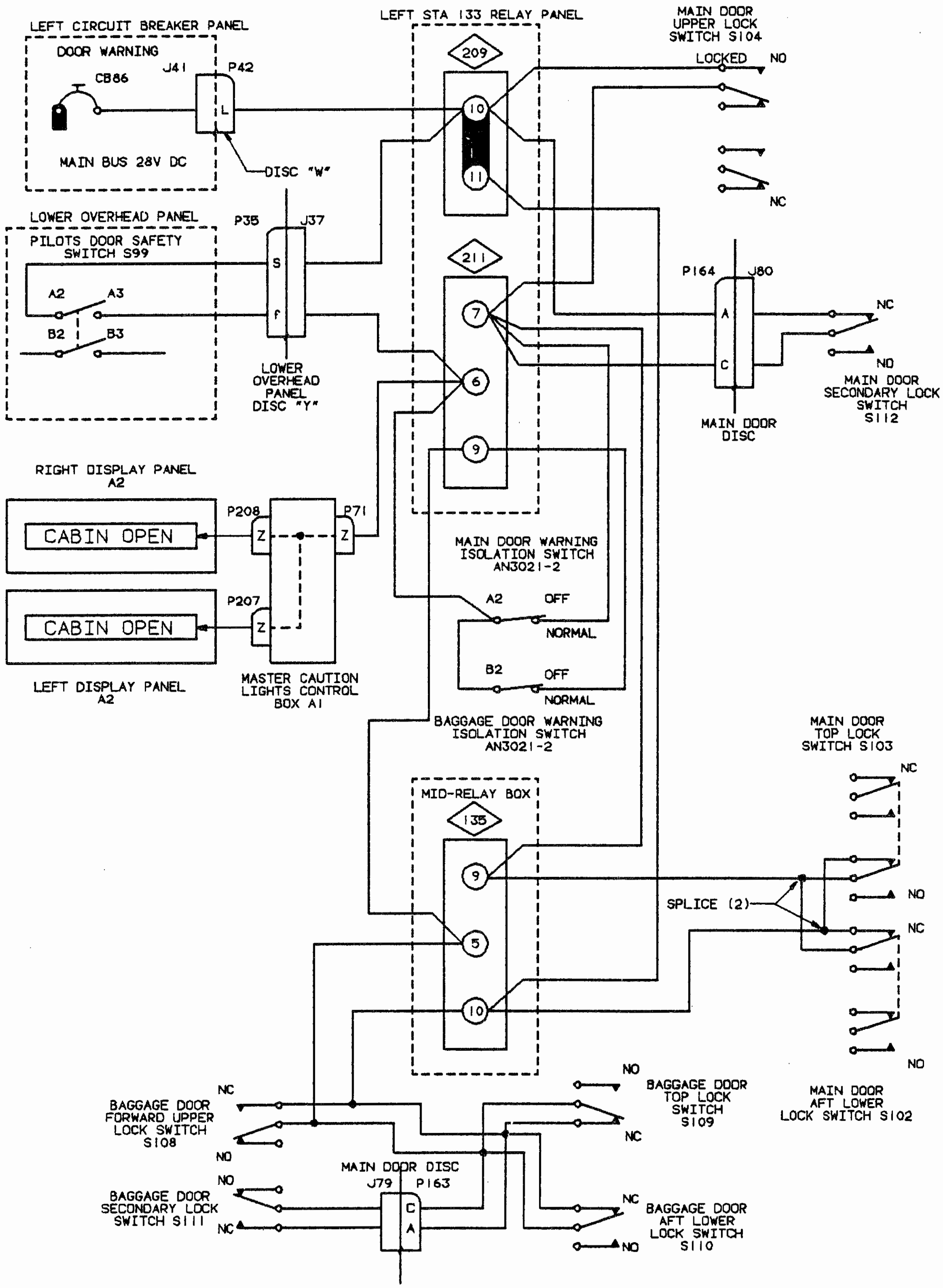


* Nomenclature for Aircraft 1-120 is "DOORS UNSAFE"

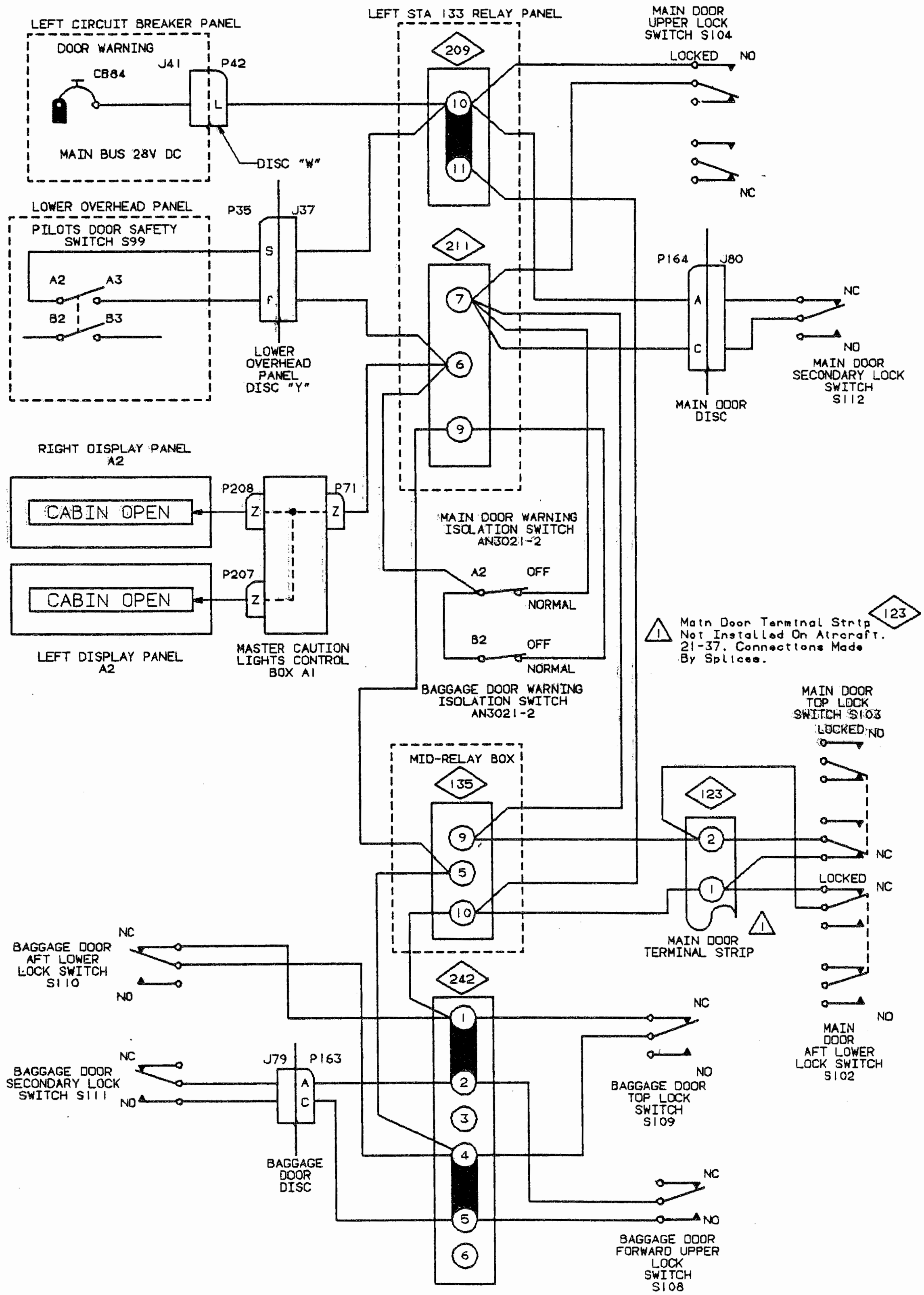
Door Warning System — Schematic
Figure 2



Door Warning System — Schematic
(Aircraft Having ASC 219)
Figure 3



Door Warning System
(Aircraft 1 – 20 Including 114 Having ASC 110)
Figure 4



Door Warning System
(Aircraft 21 – 200 Including 322, 323 Having ASC 110)
Figure 5

DOOR WARNING SYSTEM — DESCRIPTION / OPERATION

1. **Description**

(See Figure 1)

The door warning system consists of three primary lock switches and one secondary lock switch on the main entrance door, three primary lock switches and one secondary lock switch on the baggage compartment door, and the pilots DOOR SAFETY switch. The system is powered through the 5A DOOR WARN circuit breaker from the main dc bus. Three lock switches for the main entrance door are located within the structure around the door at the forward upper, top aft and aft lower bayonets. The secondary lock switch for the main door is located below secondary lock handle. The three baggage door lock switches are located within the door coaming at the top, aft lower and upper forward bayonets. The secondary baggage door lock switch is located below the secondary lock handle. The pilots DOOR SAFETY switch is located on the lower overhead panel.

If the pip pin is installed in the baggage door secondary lock handle, the secondary lock handle switch is closed, indicating an unsafe condition. This ensures against flight operations with pin in place. Emergency entrance through the baggage door is impossible. This pin is only to be used to lock the baggage door handles when the aircraft is on the ground.

2. **Operation**

(See Figure 2)

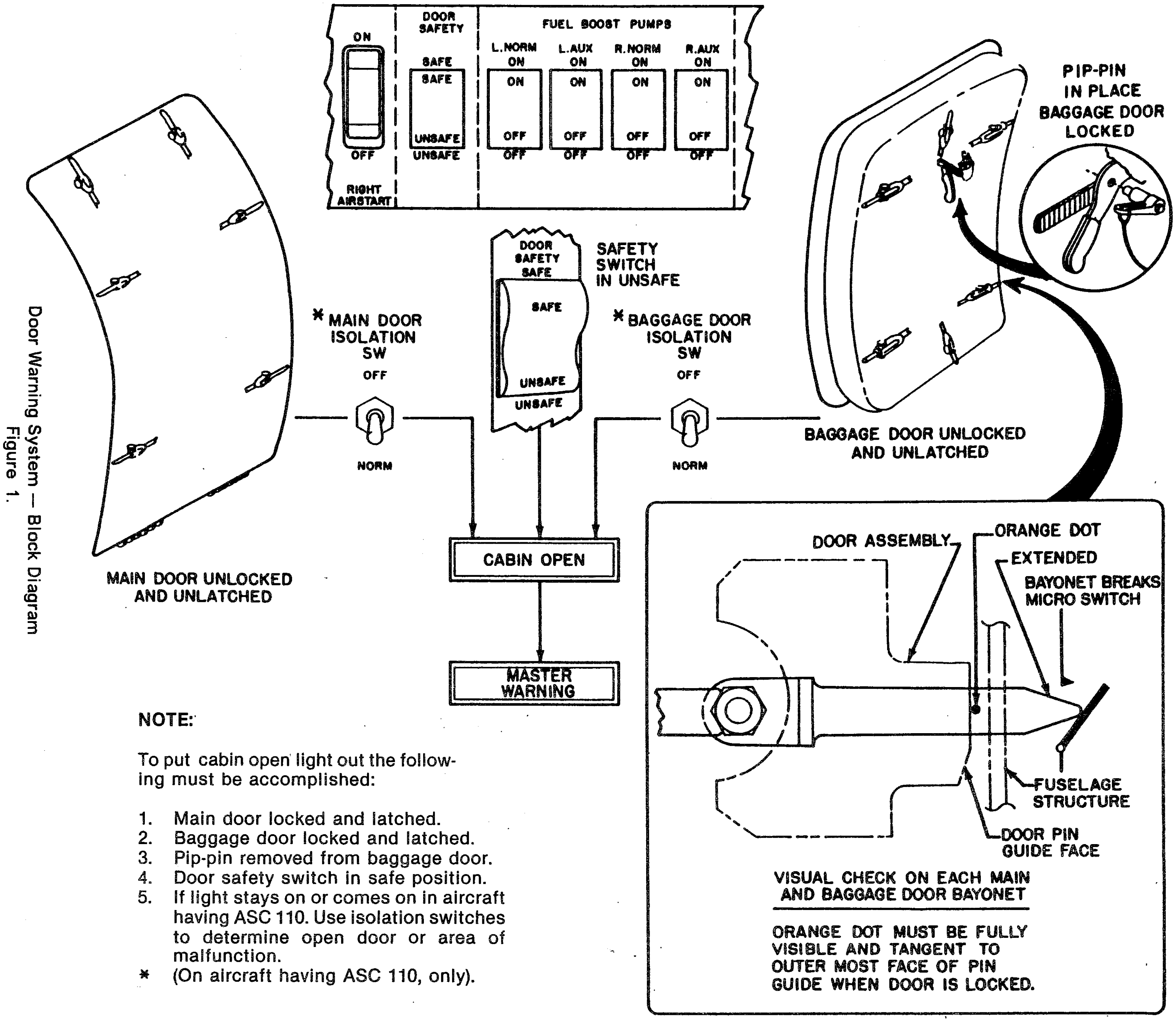
All door warning system switches are connected in parallel in the normally closed position. When one of the door lock switches or secondary switches are not in LOCKED position (bayonets are not extended and locking handle latch not secure), or if the pilots DOOR SAFETY switch is not in SAFE position, the CABIN OPEN light on master caution panel will be on. The light is turned off if all of the following conditions are met:

- The airstair door is up and the bayonets extended.
- The airstair door locking handle latch is secure.
- The baggage door is closed and its bayonets are extended.
- The baggage door locking handle latch is secure.
- The DOOR SAFETY switch (lower overhead panel) is in the SAFE position.
- The pip pin is removed from the baggage door latch.

On Aircraft having ASC 219 installed, the door warning light will come on in the event of an electrical malfunction in the door warning circuitry. (See Figure 3)

Aircraft having ASC 110 contain two additional switches (See Figure 4 and Figure 5) the MAIN DOOR WARNING ISOLATION SWITCH and the BAGGAGE DOOR WARNING ISOLATION SWITCH. These are guarded switches located on the outboard side of left Fuselage Station 133 relay panel (located directly behind the pilots seat). They enable crew to ascertain and isolate main or baggage door warning signal for easier identification of an unlocked door or malfunctioning switch. Switches are guarded to their normal position. Placing either switch OFF eliminates that set of switches from affecting the door warning lights for trouble shooting purposes.

WARNING: THE ABOVE ISOLATION SWITCHES SHOULD ALWAYS BE KEPT IN THE NORMAL (GUARD DOWN) POSITION UNLESS ACTUALLY CHECKING THE CIRCUIT FOR TROUBLE.



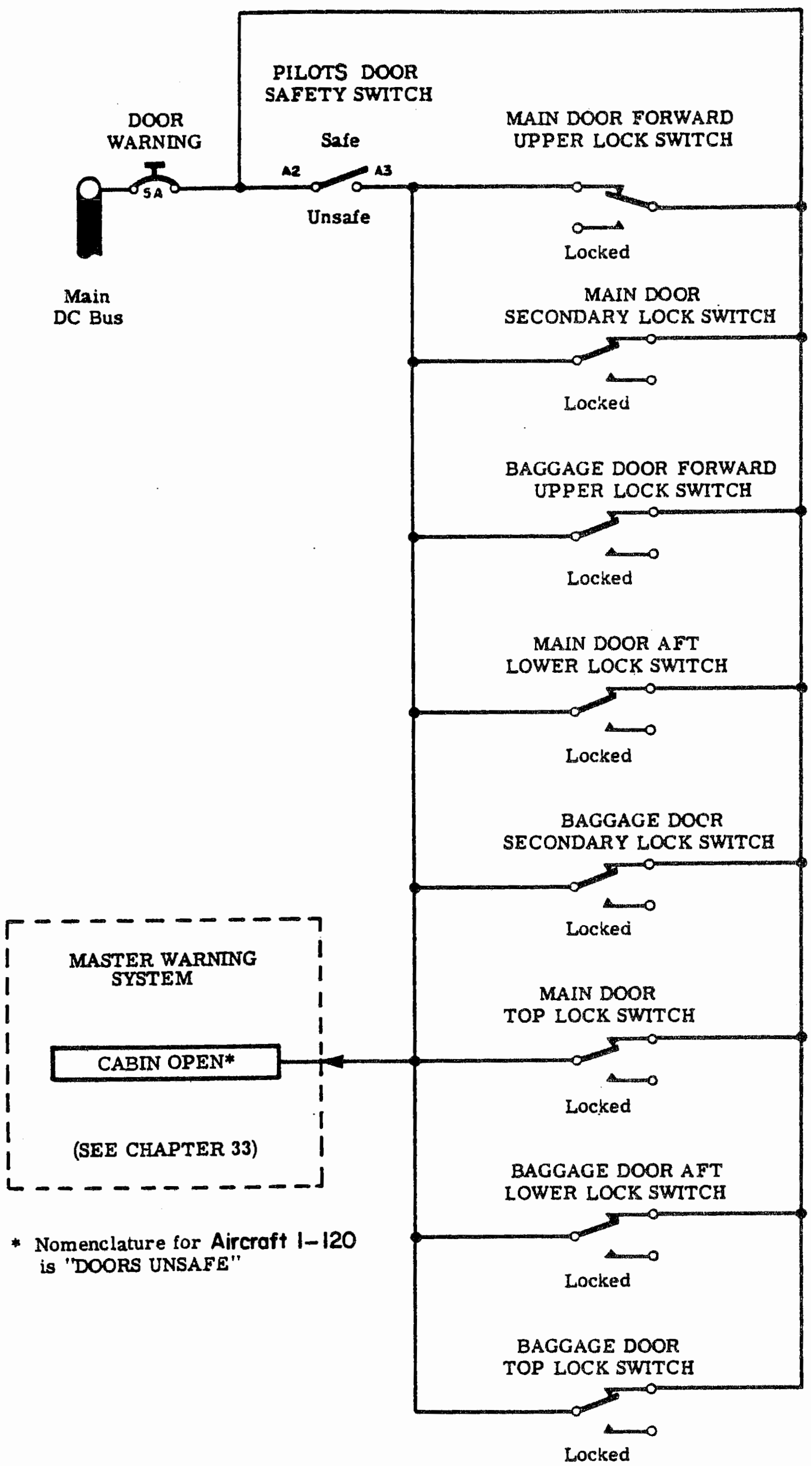
NOTE:

To put cabin open light out the following must be accomplished:

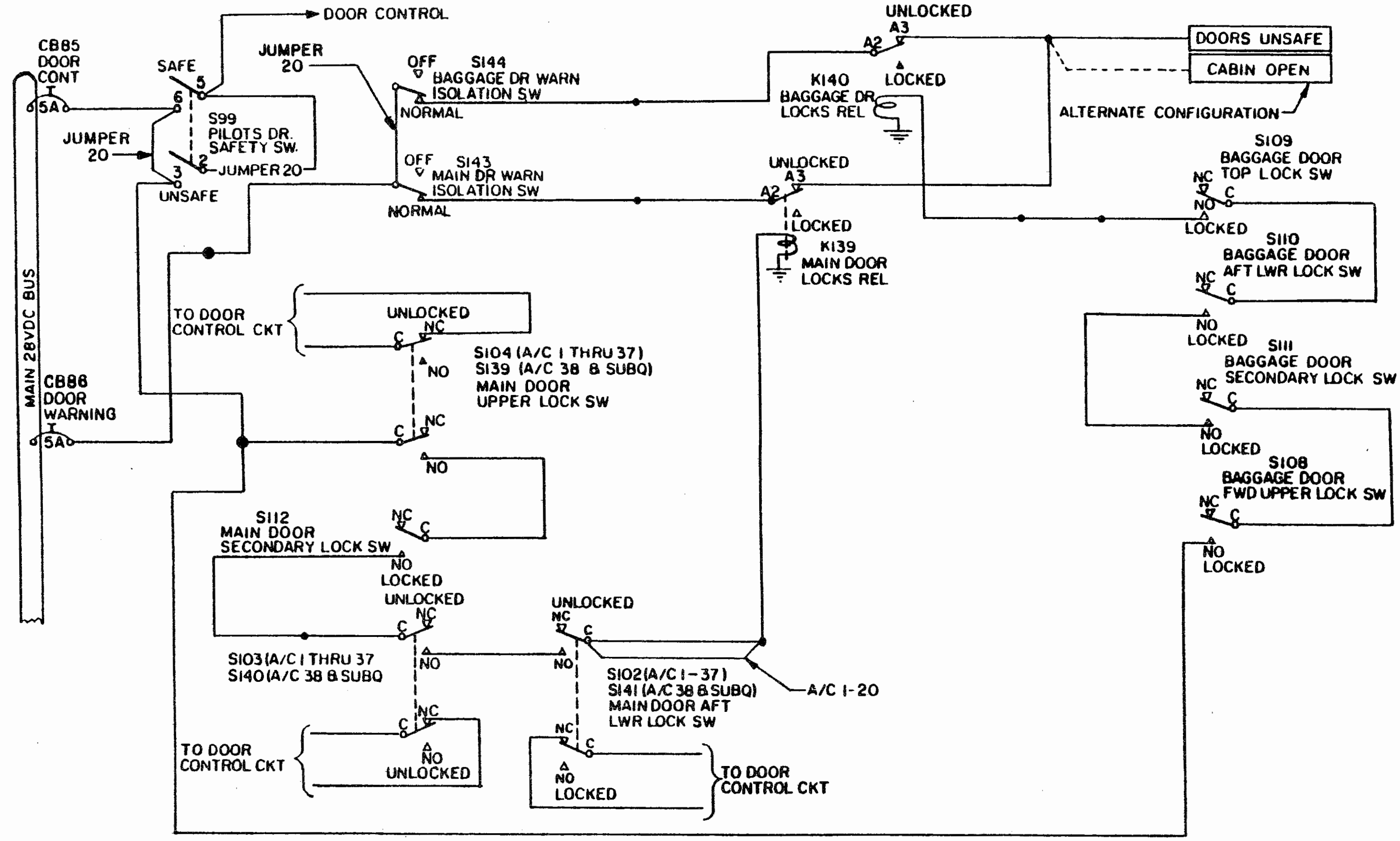
1. Main door locked and latched.
2. Baggage door locked and latched.
3. Pip-pin removed from baggage door.
4. Door safety switch in safe position.
5. If light stays on or comes on in aircraft having ASC 110. Use isolation switches to determine open door or area of malfunction.

* (On aircraft having ASC 110, only).

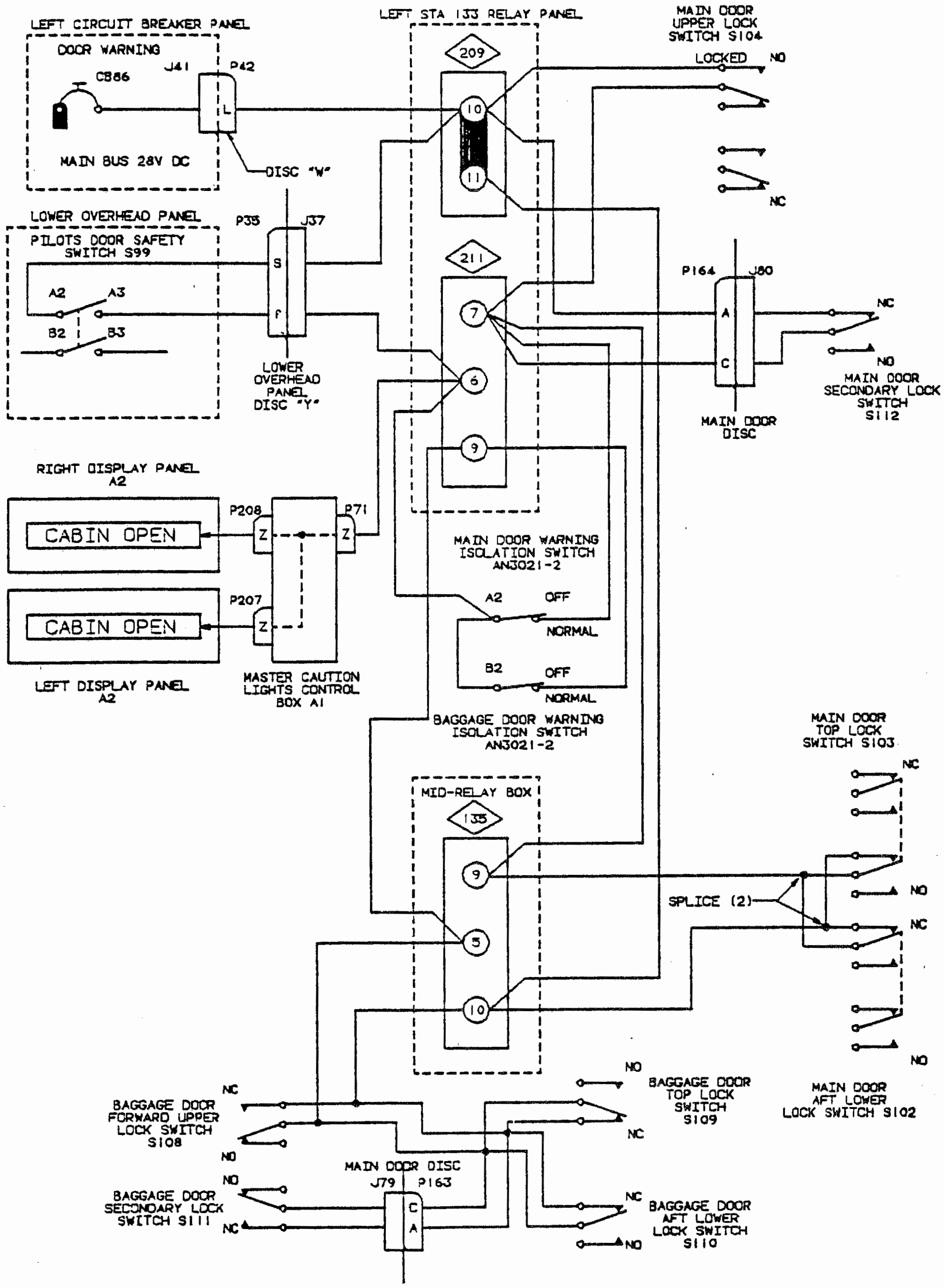
Door Warning System — Block Diagram
Figure 1.



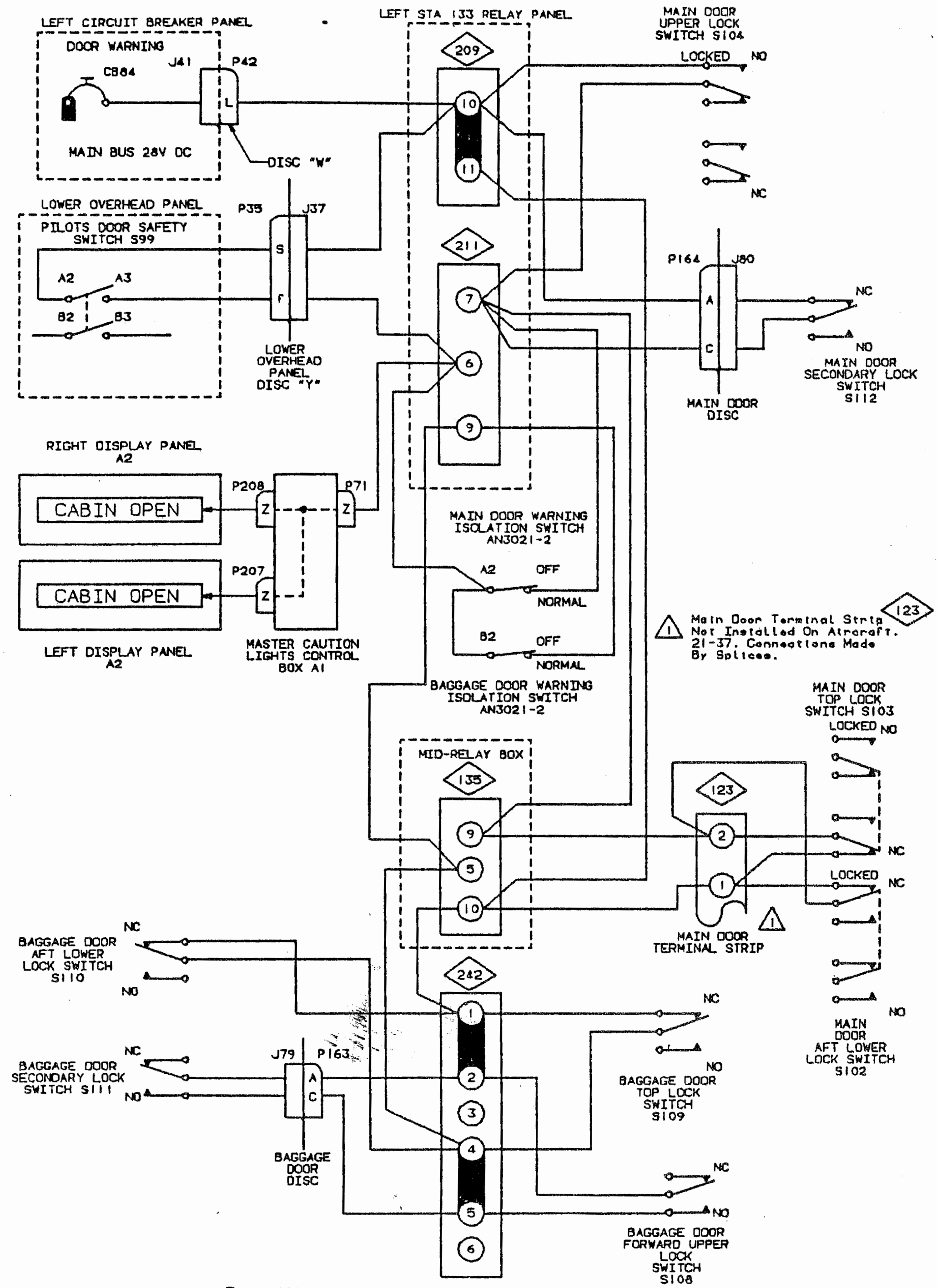
Door Warning System — Schematic
Figure 2.



Door Warning System — Schematic
(Aircraft Having ASC 219)
Figure 3.



Door Warning System — Schematic
(Aircraft 1-20 and 114 Having ASC 110)
Figure 4.



Door Warning System — Schematic
(Aircraft 21-200, 322 and 323 Having ASC 110)
Figure 5.

EMERGENCY ESCAPE HATCH — DESCRIPTION
(Aircraft Not Having ASC 197)

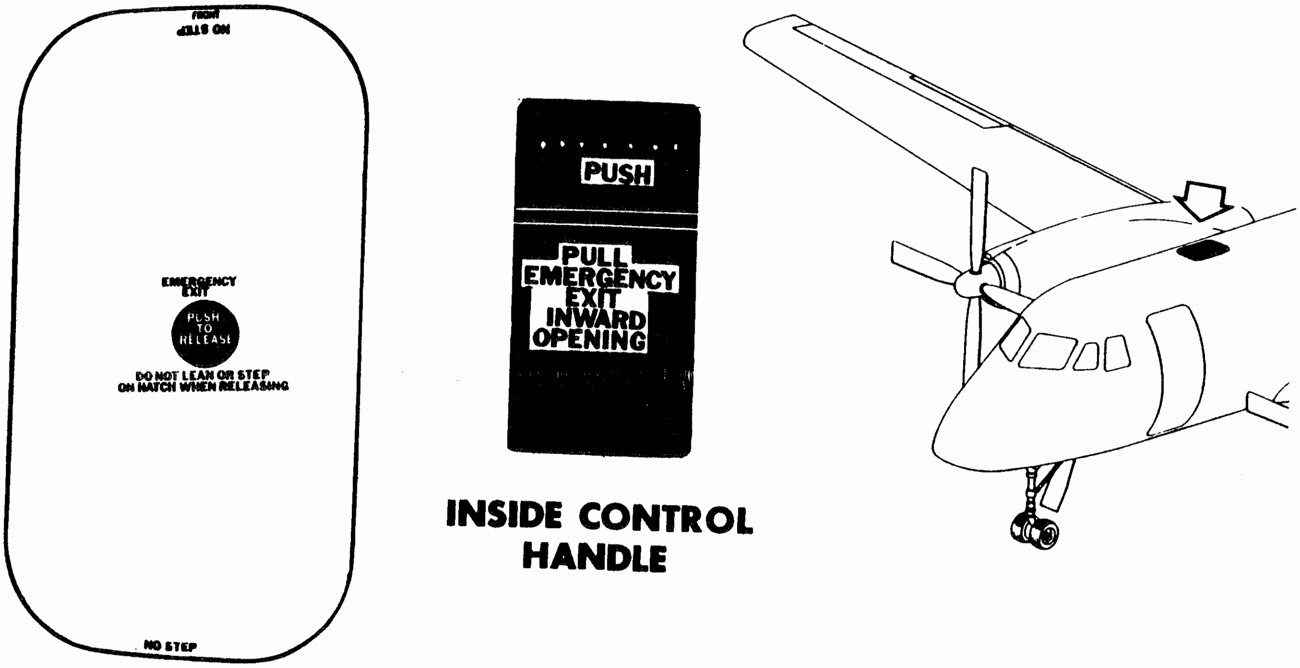
1. **Description** (See Figures 1 and 2)

The overhead emergency escape hatch is located in the forward region of the passenger compartment, between Fuselage Station 193 and 232-1/2. The hatch is unlocked and opened inward by means of a centrally located pull down handle. The inside handle, held in place by a spring clip, is moved from the outside by depressing a flush push plate located directly above the inside handle. The mechanically operated latches located on each side lock the hatch to surrounding structure. A pressurization seal is installed around the hatch. The hatch and fuselage skin joint is sealed with a high adhesion sealing compound.

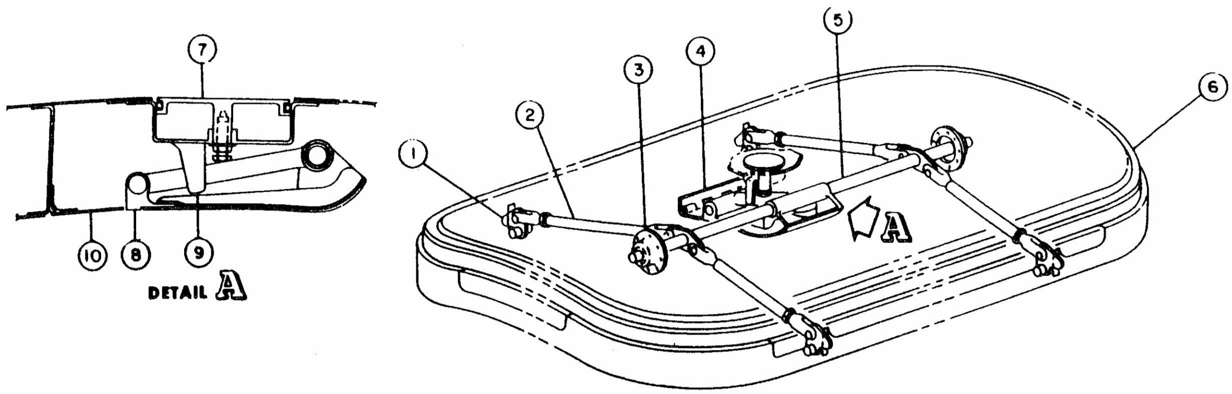
CAUTION: SINCE THIS HATCH IS A PLUG TYPE DESIGN, IT FALLS INWARD WHEN THE LATCHES ARE RELEASED. WHEN OPENING FROM INSIDE, USE BOTH HANDS TO CONTROL THE MOTION OF THE HATCH. WHEN OPENING FROM OUTSIDE, BE CAREFUL NOT TO FALL THROUGH THE OPENING.

The evacuation plan in which the emergency escape hatch is used is shown in Figure 3.

NOTE: Fuselage overhead escape hatch may be removed by ASC 197, provided ASC 153 (maximum passenger certification) has not been accomplished.

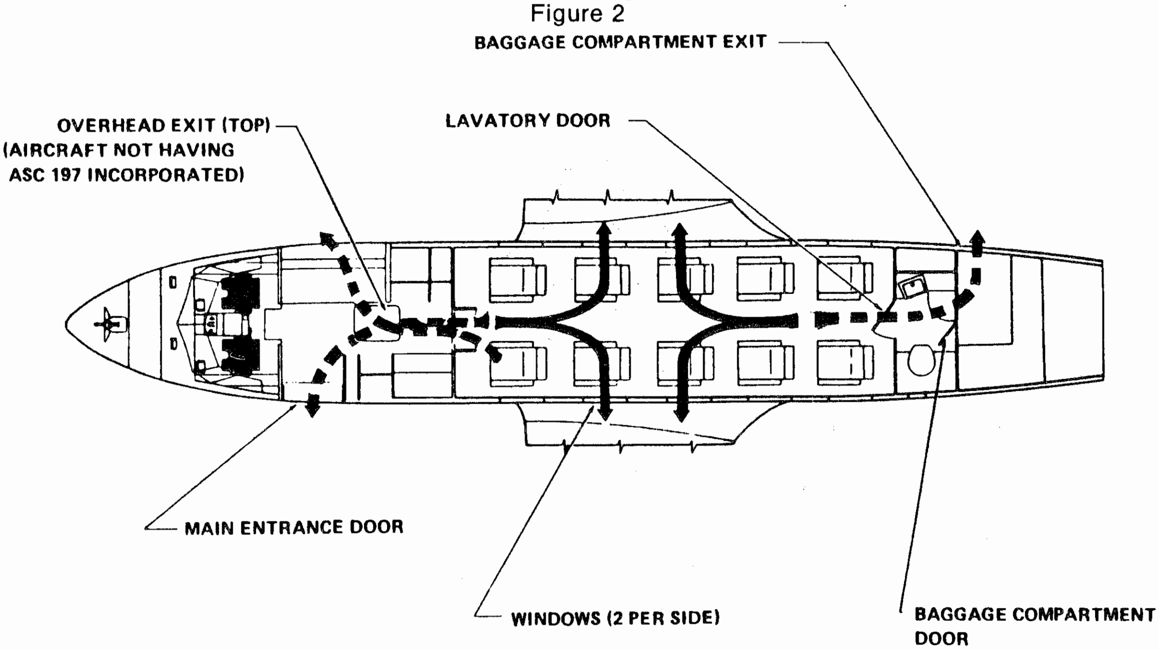


Emergency Escape Hatch
Figure 1



1. Latch
2. Push Rod
3. Housing
4. Handle Assembly
5. Torque Rod
6. Seal
7. Depressor
8. Handle
9. Spring
10. Door

Emergency Escape Hatch Components
(Aircraft Not Having ASC 197)
Figure 2



In the event of an emergency, passengers and crew can exit through the main entrance door, four cabin escape windows, baggage compartment door or overhead hatch (Aircraft not having ASC 197). The four cabin windows are removed into the cabin and are provided with painted (red) pull handles and placarded release instructions. Both the main entrance door and the baggage compartment door open outward and have painted (red) manual lock release controls and placarded instructions for opening the doors from the inside of the aircraft.

If the aircraft is ditched, passengers and crew exit through the overhead escape hatch which is removed into the cabin by means of the placarded control handle located on the hatch (Aircraft not having ASC 197 only).

Emergency Exit Plan
Figure 3

EMERGENCY ESCAPE HATCH — MAINTENANCE PRACTICES

1. **Overhead Escape Hatch — Operational Tset**

WARNING: THIS HATCH IS A PLUG TYPE DESIGN AND WILL FALL INWARD WHEN LATCHES ARE RELEASED. WHEN OPENING FROM INSIDE, USE BOTH HANDS TO CONTROL THE MOTION OF THE HATCH. WHEN OPENING FROM OUTSIDE, A PERSON MUST BE INSIDE TO SUPPORT HATCH WHEN RELEASED.

NOTE: This operational test not applicable to aircraft having ASC 197.

A. Inside Operation
   (1) With one hand supporting hatch, open with other hand as follows:
      (a) Depress PUSH plate on inside of hatch.
      (b) Pull down on handle to release hatch.
      (c) Remove hatch.
   (2) Check for freedom and smoothness of operation of hatch mechanism.
   (3) Check condition of hatch seals for cuts, deterioration or pliability.
   (4) Position hatch into fuselage opening with FRONT placard facing forward.
   (5) Push up on handle to lock mechanism and reset PUSH plate.
   (6) Visually check perimeter of door to ensure proper seating of door into hatch opening.

B. Outside Operation
   (1) Assign one man inside aircraft to support door.
   (2) Gain access to outside top of fuselage.

   WARNING: ENSURE INSIDE MAN IS FIRMLY SUPPORTING HATCH BEFORE RELEASING MECHANISM FROM OUTSIDE OF AIRCRAFT.

   (3) As inside man supports hatch, depress PUSH TO RELEASE disc from outside of aircraft. When mechanism is released hatch will drop and be removed by inside man.
   (4) Inspect and install by performing Steps A.(2) - A.(6) above.

2. **Overhead Escape Hatch Structure — Inspection**

NOTE: This inspection is not applicable to Aircraft having ASC 197. This inspection complies with CB 305.

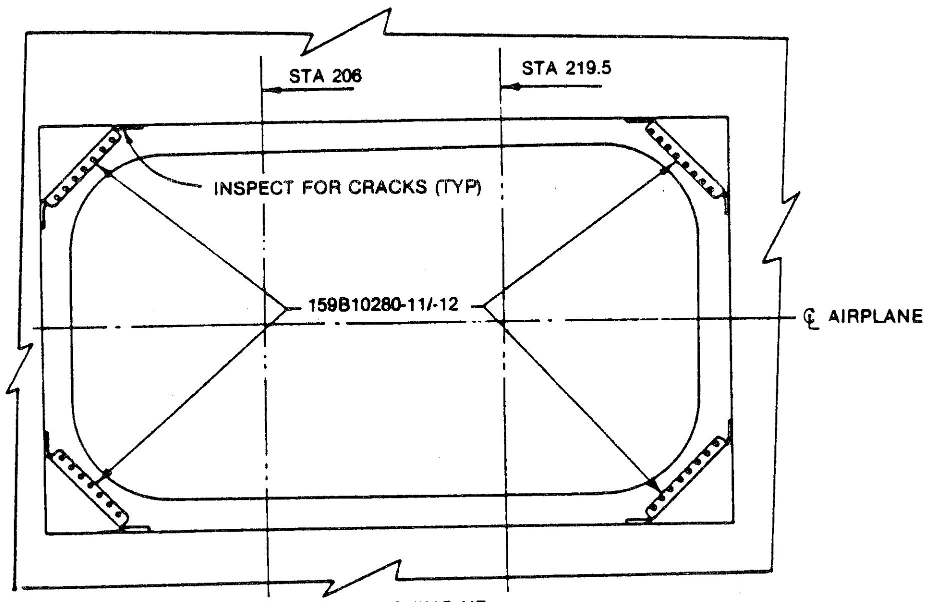
This inspection can be accomplished in conjunction with Overhead Escape Hatch — Operational Test.

A. Remove the overhead escape hatch.
B. Inspect the corner angle brackets in each corner of the framing structure for cracks. (See Figure 201).

   NOTE: Because of the seriousness of the loss of the escape hatch, a thorough inspection should be performed. This should include:
   (1) The escape hatch itself.
   (2) The edges of the fuselage skin forming the exit.
   (3) Fore and aft framing members on each side of the latch including their local attachment to the ring frames at each end.
   (4) The local ring frame structure at the forward and aft ends of the hatch.

C. Any corner angle bracket that is cracked is to be replaced within the next 150 flight hours.

   NOTE: If a corner angle is found cracked the inspection interval is changed to 150 flight hours or 6 months.



Overhead Escape Hatch Framing Structure — Bracket Location
Figure 201

EMERGENCY ESCAPE HATCH (AIRCRAFT NOT HAVING ASC 197) — DESCRIPTION / OPERATION

1. Description

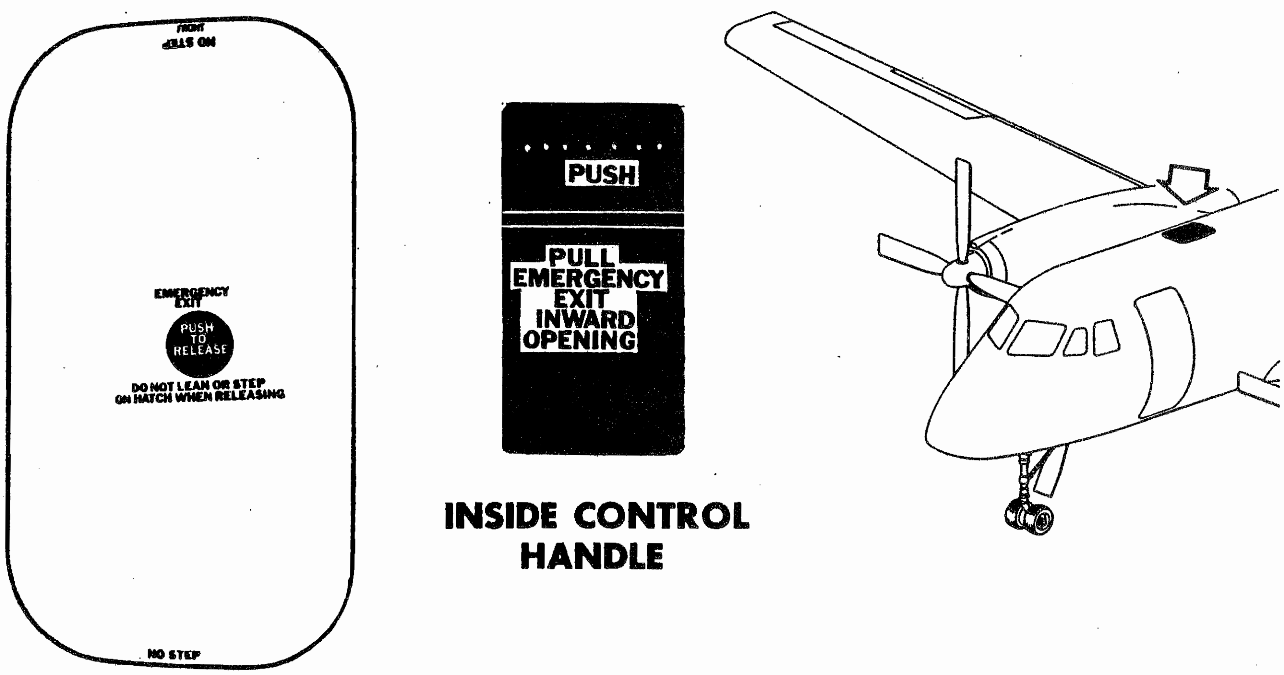
(See Figure 1 and Figure 2)

The overhead emergency escape hatch is located in the forward region of the passenger compartment, between Fuselage Station 193 and 232 1/2. The hatch is unlocked and opened inward by means of a centrally located pull down handle. The inside handle, held in place by a spring clip, is moved from the outside by depressing a flush push plate located directly above the inside handle. The mechanically operated latches located on each side lock the hatch to surrounding structure. A pressurization seal is installed around the hatch. The hatch and fuselage skin joint is sealed with a high adhesion sealing compound.

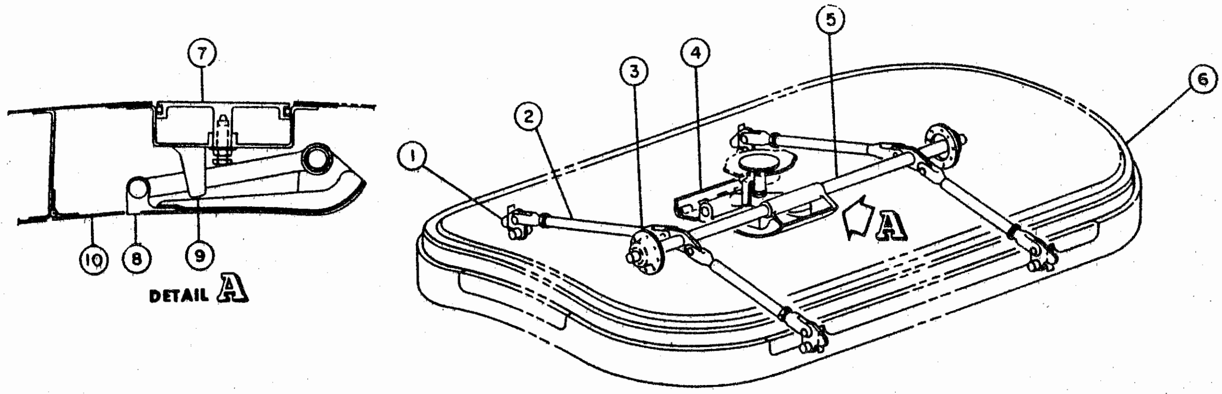
CAUTION: SINCE THIS HATCH IS A PLUG TYPE DESIGN, IT FALLS INWARD WHEN THE LATCHES ARE RELEASED. WHEN OPENING FROM INSIDE, USE BOTH HANDS TO CONTROL THE MOTION OF THE HATCH. WHEN OPENING FROM OUTSIDE, BE CAREFUL NOT TO FALL THROUGH THE OPENING.

The evacuation plan in which the emergency escape hatch is used is shown in Figure 3.

NOTE: Fuselage overhead escape hatch may be removed by ASC 197, provided ASC 153 (maximum passenger certification) has not been accomplished.

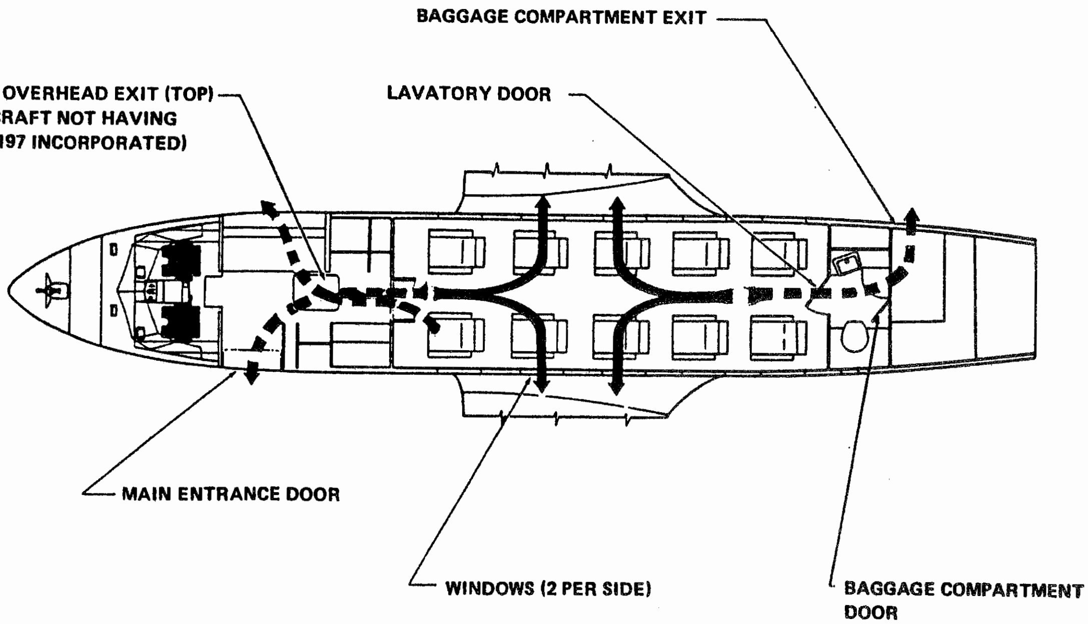


Emergency Escape Hatch
Figure 1.



1.	Latch	6.	Seal
2.	Push Rod	7.	Depressor
3.	Housing	8.	Handle
4.	Handle Assembly	9.	Spring
5.	Torque Rod	10.	Door

Emergency Escape Hatch Components
(Aircraft Not Having ASC 197)
Figure 2.



In the event of an emergency, passengers and crew can exit through the main entrance door, four cabin escape windows, baggage compartment door or overhead hatch (Aircraft not having ASC 197). The four cabin windows are removed into the cabin and are provided with painted (red) pull handles and placarded release instructions. Both the main entrance door and the baggage compartment door open outward and have painted (red) manual lock release controls and placarded instructions for opening the doors from the inside of the aircraft.

If the aircraft is ditched, passengers and crew exit through the overhead escape hatch which is removed into the cabin by means of the placarded control handle located on the hatch (Aircraft not having ASC 197 only).

Emergency Exit Plan
Figure 3.

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EMERGENCY ESCAPE HATCH (AIRCRAFT NOT HAVING ASC 197) — MAINTENANCE PRACTICES

1. **Overhead Escape Hatch — Operational Test**

NOTE: This operational test not applicable to aircraft having ASC 197.

WARNING: THIS HATCH IS A PLUG TYPE DESIGN AND WILL FALL INWARD WHEN LATCHES ARE RELEASED. WHEN OPENING FROM INSIDE, USE BOTH HANDS TO CONTROL MOTION OF HATCH. WHEN OPENING FROM OUTSIDE, A PERSON MUST BE INSIDE TO SUPPORT HATCH WHEN RELEASED.

A. Inside Operation

(1) With one hand supporting hatch, open with other hand as follows:

(a) Depress PUSH plate on inside of hatch.

(b) Pull down on handle to release hatch.

(c) Remove hatch.

(2) Check for freedom and smoothness operation of hatch mechanism.

(3) Check condition of hatch seals for cuts, deterioration or pliability.

(4) Position hatch into fuselage opening with FRONT placard facing forward.

(5) Push up on handle to lock mechanism and reset PUSH plate.

(6) Visually check perimeter of door to ensure proper seating of door into hatch opening.

B. Outside Operation

(1) Assign one man inside aircraft to support door.

(2) Gain access to outside top of fuselage.

WARNING: ENSURE INSIDE MAN IS FIRMLY SUPPORTING HATCH BEFORE RELEASING MECHANISM FROM OUTSIDE OF AIRCRAFT.

(3) As inside man supports hatch, depress PUSH TO RELEASE disc from outside of aircraft. When mechanism is released hatch will drop and be removed by inside man.

(4) Inspect and install.

(5) Check for freedom and smoothness operation of hatch mechanism.

(6) Check condition of hatch seals for cuts, deterioration or pliability.

(7) Position hatch into fuselage opening with FRONT placard facing forward.

(8) Push up on handle to lock mechanism and reset PUSH plate.

(9) Visually check perimeter of door to ensure proper seating of door into hatch opening.

2. **Overhead Escape Hatch Structure — Inspection**

NOTE: This inspection is not applicable to Aircraft having ASC 197.

This inspection can be accomplished in conjunction with Overhead Escape Hatch — Operational Test.

A. Remove overhead escape hatch.

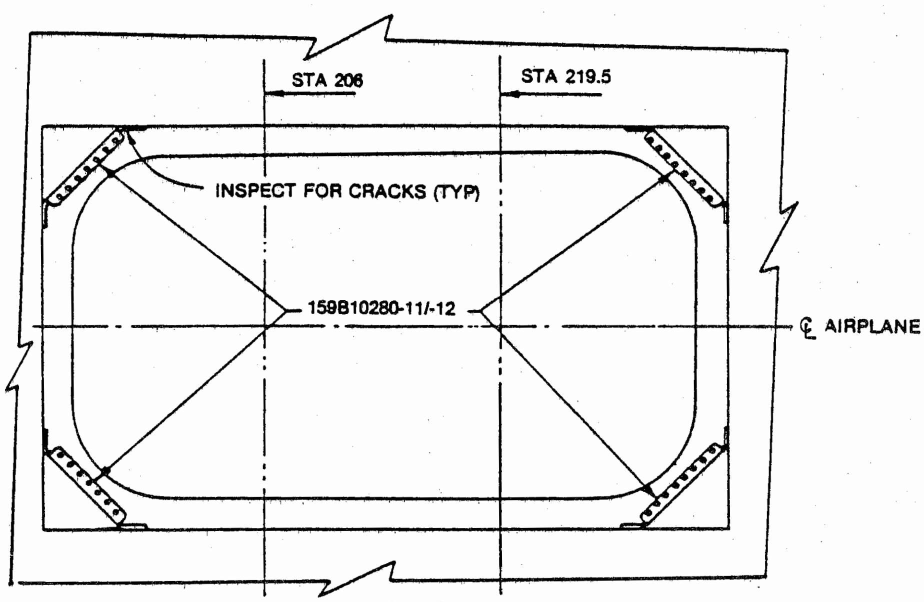
B. Inspect corner angle brackets in each corner of framing structure for cracks. (See Figure 201)

NOTE: Because of seriousness of loss of escape hatch, a thorough inspection should be performed. This should include:

- Escape hatch itself.
- Edges of the fuselage skin forming the exit.
- Fore and aft framing members on each side of the latch including their local attachment to the ring frames at each end.
- Local ring frame structure at the forward and aft ends of the hatch.

C. Any corner angle bracket cracked is to be replaced within next 150 flight hours.

NOTE: If a corner angle is found cracked inspection interval is changed to 150 flight hours or 6 months.



Overhead Escape Hatch Framing Structure — Bracket Location
Figure 201.

APU AND TAIL SECTION ACCESS DOORS — DESCRIPTION / OPERATION

1. Description

The APU access doors are located on the left side of the fuselage between Fuselage Station 585 and 616 1/2. These doors are hinged to stringer No. 15A and move out and down when opened. Two flush type, trigger controlled latches secure each door to the surrounding structure. The APU access panel is located on the right side of the fuselage between Fuselage Station 579 1/2 and 616 1/2. This panel is secured to the fuselage structure by screws and gang channels. The APU inlet door, located on the left side of the fuselage, extends from Fuselage Station 598 to Fuselage Station 616 1/2. The door is hinged to stringer No. 5 and is opened inward by the APU door actuator. The tail section access door, located on the right side of the fuselage, extends from Fuselage Station 649 11/16 to Fuselage Station 668. This door is hinged to stringer No. 20A and moves out and down when opened. Four flush-type trigger controlled latches, one on each side and two on the top, secure the tail section access door to the surrounding structure.

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APU AND TAIL SECTION ACCESS DOORS — DESCRIPTION

1. Description

The APU access doors are located on the left side of the fuselage between Fuselage Station 585 and 616-1/2. These doors are hinged to stringer No. 15A and move out and down when opened. Two flush type, trigger controlled latches secure each door to the surrounding structure. The APU access panel is located on the right side of the fuselage between Fuselage Station 579-1/2 and 616-1/2. This panel is secured to the fuselage structure by screws and gang channels. The APU inlet door, located on the left side of the fuselage, extends from Fuselage Station 598 to Fuselage Station 616-1/2. The door is hinged to stringer No. 5 and is opened inward by the APU door actuator. The tail section access door, located on the right side of the fuselage, extends from Fuselage Station 649-11/16 to Fuselage Station 668. This door is hinged to stringer No. 20A and moves out and down when opened. Four flush-type trigger controlled latches, one on each side and two on the top, secure the tail section access door to the surrounding structure.

MISCELLANEOUS ACCESS DOORS — DESCRIPTION / OPERATION

1. **Description**

A. Fuselage

Unit	No. Per A/C	Location
Control column access	2	Each side of cabin lower structure.
Oxygen bottle access	1	Intersection of stringer No. 14 and Fuselage Station 56.
Hydraulic tank filler access	1	Below upper aft corner of airstair door opening.
Control cable and fairlead access	8	Undersurface of wing center section.
Aft section control cable access	4	Bottom of fuselage between Fuselage Station 567 and 588 1/2
Elevator bellcrank access	1	Tail cone

B. Wings

Unit	No. Per A/C	Location
Window release access	4	Wing to fuselage fillet
Fuel pump access	2	Upper cover of inner panel between Fuselage
Fuel cavity bay access	12	Upper cover of inner panel
Fuel filler cap	2	Upper cover of inner panel at Fuselage Station 265 5/8
Vent valve access	2	Upper cover of inner panel at Fuselage Station 302 1/4
Water/methanol tank access	4	Upper and lower cover of outer panel between Fuselage Station 313 1/4 and 331
Water/methanol filler cap and tank access	2	Upper cover of outer panel between Fuselage Station 331 and 348 1/2
Compass flux valve cable access	2	Upper cover of outer panel between Fuselage Station 348 1/2 and 364
Aileron bellcrank access	2	Leading edge of outer panel upper cover between Fuselage Station 348 1/2 and 364
De-icer boot valve access	2	Leading edge of outer panel upper cover between Fuselage Station 364 and 382 1/2
Compass flux valve access	2	Upper cover of outer panel at Fuselage Station 398
Trim tab cables access	6	Upper cover of left outer panel between Fuselage Station 402 and 455
Aileron seal, trim tab cables and lower stop adjustment bolt access	10	Lower trailing edge of outer panel
Crossfeed lines and antennas access	2	Lower trailing edge of wing center section

Unit	No. Per A/C	Location
Engine, aileron, hydraulic, electrical and de-icing lines and cables access	2	Leading edge of wing center section
Upper aileron stop adjustment bolt access	1	Upper trailing edge of outer panel at Fuselage Station 365

C. Nacelle

Unit	No. Per A/C	Location
Engine access	2	Forward of Vickers Fuselage Station "0"
Accessory gear box and oil filler access	2	Top of nacelle, directly aft of engine access.
Fire extinguisher bottle and electrical panel access	2	Upper right side at wing and nacelle intersection.
Battery access and emergency disconnect	2	Lower outboard side at wing intersection, both nacelles.
Tailpipe clamp access	2	Upper left side at wing intersection.
Main gear attaching bolts	2	Aft lower inboard side at wing intersection.
Tailpipe access	2	Trailing edge of nacelle.
External power supply	1	Lower outboard, left nacelle.
Electrical disconnect at firewall.	2	Upper right side, both nacelles.

D. Empennage

Unit	No. Per A/C	Location
Rudder attachment access	4	Trailing edge of fin, below rudder in fixed fairing at center line of hinge.
Rudder seal access	6	Trailing edge of fin on right side.
Fin hoist access	1	Upper center of fin.
Navigation light conduit access	2	Rudder tip cap and upper trailing edge of fin on right side.
Rudder trim tab control cables access	2	Center of fin and rudder leading edge between Water Line 161 and 171 on right side.
Elevator removal access	8	Trailing edge of each stabilizer and inboard end of each elevator.
Elevator seal and lower stop adjustment bolt access	8	Trailing edge of each stabilizer on bottom side.
Elevator trim tab control	4	Center of each stabilizer and leading edge of each elevator between Stabilizer Station 46 and 58.
Elevator upper stop adjustment bolt access.	2	Upper inboard trailing edge of stabilizer.

MISCELLANEOUS ACCESS DOORS — DESCRIPTION

1. **Description**

A. Fuselage

NOMENCLATURE	QUANTITY	APPROXIMATE LOCATION
Control column access	2	Each side of cabin lower structure.
Oxygen bottle access	1	Intersection of stringer No. 14 and Fuselage Station 56.
Hydraulic tank filler access	1	Below upper aft corner of airstair door opening.
Control cable and fairlead access	8	Undersurface of wing center section.
Aft section control cable access	4	Bottom of fuselage between Fuselage Station 567 and 588-1/2
Elevator bellcrank access	1	Tail cone

B. Wings

NOMENCLATURE	QUANTITY	APPROXIMATE LOCATION
Window release access	4	Wing to fuselage fillet.
Fuel pump access	2	Upper cover of inner panel between Fuselage Station 45 and 55.
Fuel cavity bay access	12	Upper cover of inner panel.
Fuel filler cap	2	Upper cover of inner panel at Fuselage Station 265-5/8
Vent valve access	2	Upper cover of inner panel at Fuselage Station 302-1/4
Water / methanol tank access	4	Upper and lower cover of outer panel between Fuselage Station 313-1/4 and 331.
Water / methanol filler cap and tank access	2	Upper cover of outer panel between Fuselage Station 331 and 348-1/2.
Compass flux valve cable access	2	Upper cover of outer panel between Fuselage Station 348-1/2 and 364.
Aileron bellcrank access	2	Leading edge of outer panel upper cover between Fuselage Station 348-1/2 and 364.
De-icer boot valve access	2	Leading edge of outer panel upper cover between Fuselage Station 364 and 382-1/2
Compass flux valve access	2	Upper cover of outer panel at Fuselage Station 398.
Trim tab cables access	6	Upper cover of left outer panel between Fuselage Station 402 and 455.
Aileron seal, trim tab cables and lower stop adjustment bolt access	10	Lower trailing edge of outer panel.
Crossfeed lines and antennas access	2	Lower trailing edge of wing center section.
Engine, aileron, hydraulic, electrical and de-icing lines and cables access.	2	Leading edge of wing center section.
Upper aileron stop adjustment bolt access	1	Upper trailing edge of outer panel at Fuselage Station 365.

C. Nacelle

NOMENCLATURE	QUANTITY	APPROXIMATE LOCATION
Engine access	2	Forward of Vickers Fuselage Station "O".
Accessory gear box and oil filler access	2	Top of nacelle, directly aft of engine access.

NOMENCLATURE	QUANTITY	APPROXIMATE LOCATION
Fire extinguisher bottle and electrical panel access	2	Upper right side at wing and nacelle intersection.
Battery access and emergency disconnect	2	Lower outboard side at wing intersection, both nacelles.
Tailpipe clamp access	2	Upper left side at wing intersection.
Main gear attaching bolts	2	Aft lower inboard side at wing intersection.
Tailpipe access	2	Trailing edge of nacelle.
External power supply	1	Lower outboard, left nacelle.
Electrical disconnect at firewall.	2	Upper right side, both nacelles.

D. Empennage

NOMENCLATURE	QUANTITY	APPROXIMATE LOCATION
Rudder attachment access	4	Trailing edge of fin, below rudder in fixed fairing at center line of hinge.
Rudder seal access	6	Trailing edge of fin on right side.
Fin hoist access	1	Upper center of fin.
Navigation light conduit access	2	Rudder tip cap and upper trailing edge of fin on right side.
Rudder trim tab control cables access	2	Center of fin and rudder leading edge between Water Line 161 and 171 on right side.
Elevator removal access	8	Trailing edge of each stabilizer and inboard end of each elevator.
Elevator seal and lower stop adjustment bolt access	8	Trailing edge of each stabilizer on bottom side.
Elevator trim tab control	4	Center of each stabilizer and leading edge of each elevator between Stabilizer Station 46 and 58.
Elevator uper stop adjustment bolt access.	2	Upper inboard trailing edge of stabilizer.

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NACELLE

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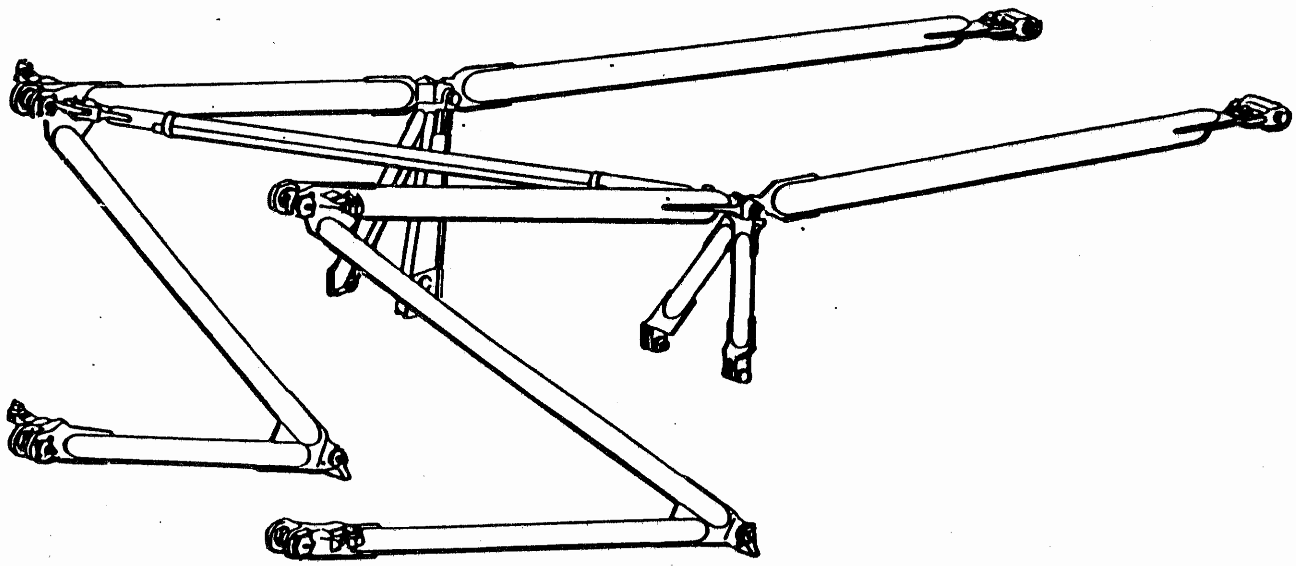
CHAPTER 54

NACELLE

NACELLES — DESCRIPTION / OPERATION

1. Description

Each nacelle is of metal construction, and is permanently attached to the Wing. The structure consists mainly of aluminum alloy members with steel details used when necessary. The main landing gear, when retracted, are housed in the nacelles and are fully enclosed by inboard and outboard doors and an aft mounted fairing. The nacelle also contains the "Grumman" mount assembly (see Figure 1) to which the Quick Engine Change Assembly is attached to the nacelle structure firewall at four fitting points. The engine tailpipe is attached to the inner flange of the firewall, and at two points on each side aft of the firewall above wing.



Grumman Mount Assembly
Figure 1.

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NACELLES — MAINTENANCE PRACTICES

1. Engine Accessory Section / Nacelle — Inspection

A. Open all engine accessory section access panels.

B. Condition, security and evidence of corrosion of structural members and adjacent skin.

C. Condition, security and evidence of failure of engine mount (Grumman) and mount bolts.

D. Condition, security and evidence of failure of firewall. Pay particular attention to inner flange of the cap to the fitting that joins the Vickers and Grumman engine mounts. Inspect center section of firewall for cracks. (See Figure 203). If cracks are found, the cracked part(s) should be replaced (preferred) or repaired within the next 25 flights. Repair drawing 159RDW523 is available through Gulfstream Aerospace Technical Operations.

E. Condition, security and evidence of failure or leakage of accessory gearbox including drive shaft and mounted components.

F. Check gearbox oil level on dipstick.

CAUTION: AFTER CHECKING OIL LEVEL, VERIFY THAT DIPSTICK IS SECURELY IN PLACE BY LIGHTLY PULLING UP ON THE DIPSTICK. ALSO CHECK THAT THE SPRING LATCH IS COMPLETELY SEATED ON TOP OF DIPSTICK.

G. Condition, security and evidence of leakage of hydraulic, fuel, water / methanol, pneumatic de-icing, fire protection and air condition (right engine only) installations including; components, lines, hoses, ducting, fittings and connections.

H. Engine control installations including components, cables, control rods, linkages, bellcranks, pulleys, and fairleads.

I. Accessory gearbox, components and mounting for condition, security and evidence of failure or leakage.

J. Accessory gearbox drive shaft tunnel housing for condition, security and evidence of failure.

K. Inspect for elongation of attachment holes between accessory gearbox mounting brackets and top tubes of engine mount. (See Figure 204)

L. Alternator and generator for signs of overheating, security of components and evidence of scoring.

M. All electrical connections in area for condition, security and safeties as necessary.

N. All wires and cables for condition, evidence of chafing, adequate lacing and security of clamps.

O. Blower (right gearbox only) for condition, security and evidence of failure or leakage.

P. Flex hose (pressure) from hydraulic pump to filter (right engine nacelle only) for evidence of chafing and condition.

Q. Inspect electrical installations including components, wiring, conduits, plugs, connectors, terminal strips and junction boxes for condition, security and evidence of overheating.

R. Inspect the battery compartment for condition of skin and evidence of structural failure.

S. Inspect nacelle fire extinguisher installation including components, tubing, blowout plugs (yellow aft, red fwd), fittings and connectors for condition, security and evidence of leakage.

T. Inspect the tailpipe compartment for condition and evidence of structural failure.

U. Inspect forward tailpipe blanket for condition, security and damage.

V. Inspect for presence of foreign objects then close access panels.

2. Nacelle — Inspection

A. Open access panels (shaded) in Figure 201.

B. Nacelle skin and structure for general condition and evidence of structural failure

C. Wing heat shield for proper installation, condition and security.

D. Nacelle fire extinguisher installation, including components, tubing, blowout plugs, fittings and connectors for condition, security and evidence of leakage

E. Check date of last weight check of fire extinguisher and record date on attached card.

F. Check date of last hydrostatic test of fire extinguisher and record date on attached card.

G. Fire detector sensing elements for condition, security, corrosion and evidence of chafing or sharp bends. Replace element if any of the above conditions are found.

H. Blower duct OVERHEAT WARN switch for security, condition of plug and proper safety.

I. Tailpipe and blanket for condition and security.

J. Tailpipe mounting brackets, forward and aft, for condition and security.

K. Tailpipe clamp and tailpipe for correct installation.

L. Tailpipe compartment for condition and security, evidence of overheating of wing skin below heat shield, as seen thru heat shield-to-shim gap area.

M. Electrical wires for condition, evidence of overheating and chafing.

N. Inspect for chafing of hoses and hydraulic lines.

O. Condition of flight and engine control cables, particularly the left throttle control cable between two pulleys on inboard side of left hand nacelle, for chafing and wear. (If cable is worn or chafed, see Throttle / HP Cock Control Cables - Wear Limits, Section 76-0 for limits and corrective actions).

P. Air conditioning installation including components, ducting and connections for condition, security and evidence of leakage.

Q. Spill valve for condition, security and leakage Lubricate valve shaft with anti-corrosion solution,

R. Engine breather line and drain line installation for condition, security and evidence of leakage.

S. Right main hydraulic filter bracket (159H10158-1) for evidence of cracking. If cracks are discovered, replace bracket.

T. Hydraulic flex hose 159H10165-17P-8, located in right nacelle between hydraulic pump and filter for chafing, replace line if chafed.

U. Inspect first bulkhead aft of wing front beam in upper nacelle fairing for condition and cracks.

V. Bellows assembly for condition and security.

W. Tadpole seal installation for appearance and general condition. Replace if required.

X. Battery compartment for condition, skin for structural failure and evidence of corrosion.

Y. Battery compartment components including battery, connections and vent lines for condition, security,

Z. Cleanliness and evidence of corrosion.

AA. Outside battery switch for security.

AB. Cowling and cowling fasteners for condition and security.

3. Engine Mount Truss — Inspection

NOTE: Inspection accomplished at engine change only.

A. Inspect engine mount truss for condition and security.

4. Engine Mount Chafe Guard — Inspection

A. For Aircraft having ASC 180.

(1) Inspect engine mount for signs of chafing, damage and security.

(2) Replace chafe guard if chafing is evident.

NOTE: This Step complies with requirements of AD 67-17-5.

B. For Aircraft not having ASC 180, (see Figure 202).

(1) Perform a thorough inspection of engine mount for evidence of chafing or interference with engine tailpipe blanket.

NOTE: If chafing is evident, see ASC 180 for recommended instructions / repairs.

5. Firewall Receptacle — Inspection

A. Gain access to front and rear of engine firewall receptacles of each engine by removing access panels, see Figure 205 and Figure 206 and open engine cowling.

B. Remove and retain mounting hardware from the No. 2, 15 thru 20 and 24 thru 28 receptacles, see Figure 207)

C. Disassemble connectors and inspect for corrosion.

NOTE: If no corrosion is found, assemble and install removed receptacles.

D. Inspect area for presence of foreign objects and security of all attachments.

E. Secure access panels and close cowlings.

6. Engine Grumman Mount — Inspection

NOTE: Compliance with the following procedures meet requirements of A.D. 84-20-08.

A. Mounts with less than 7500 landings

Comply with inspection criteria of Customer Bulletins 241C and Amendments for inspection of engine mounts.

B. Mounts with more than 7500 landings

The following procedure applies to aircraft with engine mounts that have accumulated 7500 landings and are classified as Category A in Step A. above.

Inspection Of Engine Mounts (Grumman Mounts) For Service Beyond 7500 Landings

Engine mounts are known to be susceptible to the effects of corrosion of both the scale and pitting variety. The presence of corrosion affects the fatigue life of the mount. Any "Grumman Mounts" known to have corrosion (CB 241C, Categories A through E) are required to continue on the existing requirements of AD 84-20-08 (previously 74-03-01) and amendments, plus CB 241C and Amendments.

The redesigned "Grumman Mount," featuring the open end tube design, has improved resistance to corrosion and can take advantage of the results of a recently completed fatigue test (20,000 landing fatigue life). The closed end mount design is also eligible for this advantage; however, it has been our experience that internal corrosion will develop and disqualify these mounts.

The CB 241C and Amendments mount wall thickness categories have been revised. Some mounts previously classified as "Category B" mounts may now come under the "Category A" classification. A review of the revised "A" and "B" categories should be conducted to determine if any mount classification changes are required.

Mounts classified as "Category B" cannot qualify for the 20,000 landing life limit and must continue compliance with the requirements of CB 241C and Amendments and FAA Airworthiness Directive 84-20-08 and Amendments.

The new life limit of all "Category A" "Grumman Mounts" without corrosion is 20,000 landings. The inspection requirements to continue to qualify for this life limit are as follows:

Compliance

Initial 20,000 Landing Qualification Inspection

- **Mounts with less than 7500 landings** — No later than 30 days after accumulation of 7500 landings.

  NOTE: To take full advantage of the longer recurring inspection interval of CB 241C, initial compliance should not be accomplished until Grumman Mount landing accumulation approaches 7500 landings.

- **Mounts with 7500 or more landings** — No later than 12 months after the last successful inspection per CB 289, Amendment 1.

Recurring 20,000 Landing Qualification Inspections

- At intervals no greater than 12 months after the initial inspection.

20,000 Landing Qualification Inspection

On mounts classified by CB 241C as "Category A", perform an external visual inspection to ensure no cracks are present. Pay particular attention to the junction of the tubes and end fittings where the tubes are slotted to accept the fittings, and to where all non-welded attachments have been made to the tubes.

NOTE: A change in the fit or ease of opening of cowling panels/doors or main wheel doors could be an indication of a failure in the engine mount truss-work. If, at any time, such a change is noted, the above visual inspection should be performed before further flight.

Perform Engine Mount Internal Inspection (NDT) Procedure below. If the internal NDT inspection indicates that corrosion is negligible or does not exist and visual inspection does not reveal any cracks in any mount subassemblies, those mount subassemblies may continue service beyond 7500 landings with these inspections successfully repeated at 12 month inspection intervals, up to the life limit of 20,000 landings (wet or dry power applications). Also, for these Grumman Mount subassemblies, further compliance with CB 241C and Amendments will no longer be necessary.

NOTE: Negligible corrosion is defined as that slight amount of corrosion that can be regarded as having no significant effect on mount fatigue life. Determination of the level of internal mount corrosion (i.e. non-existent, negligible, or non-negligible) is to be made only by the following Gulfstream Aerospace Corporation ultrasonic inspection (as performed by a trained/qualified technician), or in an equivalent manner approved by the Manager, Atlanta Aircraft Certification Office, FAA, Central Region.

If any internal NDT inspection indicates that corrosion is present and is in excess of "negligible corrosion", the mount or affected mount subassembly can no longer qualify for the 20,000 landing life limit and the mount or affected mount subassembly must be replaced within the next 25 landings or 30 days, whichever occurs first.

If a mount crack is found by any inspection, notify Gulfstream Aerospace Corp. Tech Services Department, for disposition before further flight. Also, notify the Manager, Atlanta Aircraft Certification Office, FAA, Central Region.

Engine Mount Internal Inspection (NDT) Procedure

This procedure establishes the test required to determine if the level of internal corrosion in G-159 Grumman engine mount tubes is negligible or non-negligible with regard to Grumman engine mount life limit considerations. The procedure must only be performed by trained/qualified technicians.

(1) Applicability: This test procedure is applicable to the following components:

Original Closed End Mounts	Drilled End Mounts	Open End Mounts	
159W10170-1	-901	*	Truss Assy, Zee - L/H Inbd
159W10170-3	-903	*	Truss Assy, Zee- R/H Inbd
159W10171-1	-901	*	Truss Assy, Zee - L/H Outbd
159W10171-3	-903	*	Truss Assy, Zee- R/H Outbd
159W10172-5	-905	*	Tube Assy, Upper Diagonal L/H
159W10172-7	-907	*	Tube Assy, Upper Diagonal R/H
159W10181-1	-901	*	Tube Assy, Overwing - L/H Outbd
159W10182-1	-901	*	Tube Assy, Overwing - L/H Inbd
15910183-1	-901	*	Tube Assy, Overwing . R/H Outbd
159W10184-1	-901	*	Tube Assy, Overwing - R/N Inbd
159W10185-1	-901	*	A Frame Assy, L/H Outbd
159W10186-1 **	NA	NA	A Frame Assy, L/H Inbd
159W10187-1 **	NA	NA	A Frame Assy, R/H Outbd
159W10188-1	-901	*	A frame assy, R/H Inbd

* All other part numbers
** Visual inspection only for non-tubular "A" Frame Assemblies

(2) The following test equipment will be required:

- Nortec NDT-131 Portable Ultrasonic Testing Machine (or equivalent).
- Nortec contoured E-M-10MNZ 3/8 inch concave transducer one each R = 0.437 inch, R = 0.625 inch, R = 0.875 inch and R = 1.062 inch.
- A suitable sound couplant.
- Reference standards (closed end and drilled/open end types) containing negligible and non-negligible corrosion approved for use in this test.
- American optics high resolution, flexible, industrial fiberscope FS 4201, or equivalent.

NOTE: For instruments and equipment rented from Gulfstream Aerospace, if the calibrations for the unit differ from those listed below, detailed settings matching the instrument to the reference standard will be provided and packaged with each rental unit.

(3) Record S/N, date, total hours and number of landings for each engine truss/tube assembly inspected.

(4) On aircraft remove/open the following:

- Main landing gear wheel well doors.
- Nacelle/Wing fillet panels.
- All nacelle access panels.
- If aircraft is on jacks, the main landing gear must be down.

- Engine gearbox (as required).
- Engine tailpipe(s) (as required).

SPECIAL NOTE FOR OPEN END TUBES ONLY

Corrosion on the tube inside surface is detectable visually through the use of a flexible fiber optics scope. The inside surface of the open end tubes is painted white and the corrosion is detectable by the observance of rust on the white background. It will probably be necessary to flush out the tubes with solvent before this inspection can be performed.

If no evidence of corrosion is observed visually, the tube may be considered to be without corrosion. if corrosion is found visually, the corrosion must be measured by the following ultrasonic method to determine if it is negligible or non-negligible.

(5) Select transducer to match engine mount tube diameter to be tested.

R = 0.437 inch for 7/8 inch diameter tube.
R = 0.625 inch for 1 1/4 inch diameter tube.
R = 0.875 inch for 1 3/4 inch and 1 7/8 inch diameter tube.
R = 1.062 inch for 2 inch and 2 1/8 inch diameter tube.

(6) Select reference standard to match mount type (closed end or drilled/open end). The closed end reference standard has an unpainted interior (ID) surface; the drilled/open end reference standard has a painted interior (ID) surface.

(7) Connect transducer cables to instrument.

(8) Turn delay calibration fully clockwise.

(9) Set range calibration to 4.42.

(10) Set range to 0.25.

(11) Set delay to 1.

NOTE: Initial pulse should be visible on the left side of CRT.

(12) Set reject to MAXIMUM.

(13) Turn gate OFF.

(14) Set filter on (—).

(15) Set DB gain to 100 DB's.

(16) Set frequency to 10MNZ.

(17) Set transmission to DUAL.

(18) Set dampening to MINIMUM.

(19) Using a suitable sound couplant, position transducer on the negligible corrosion area of reference standard and note carefully the response on the CRT. This response is an example of that produced by a mount tube with negligible corrosion.

(20) Position transducer on non-negligible corrosion area of reference standard and note carefully the diminished response. Any mount tube producing a response equal to or more diminished that this reference standard response is considered to have non-negligible corrosion.

(21) Above settings are intended only to aid an operator of advanced skills in calibrating the instrument. Electronic and mechanical variations in transducers, cables, instruments and power supply will require changes in listed settings.

(22) Principle of test is to monitor loss of sound energy on irregular pitted ID of tubes.

(23) Before proceeding to test tubes of different diameter Steps (5) thru (20) above must be repeated.

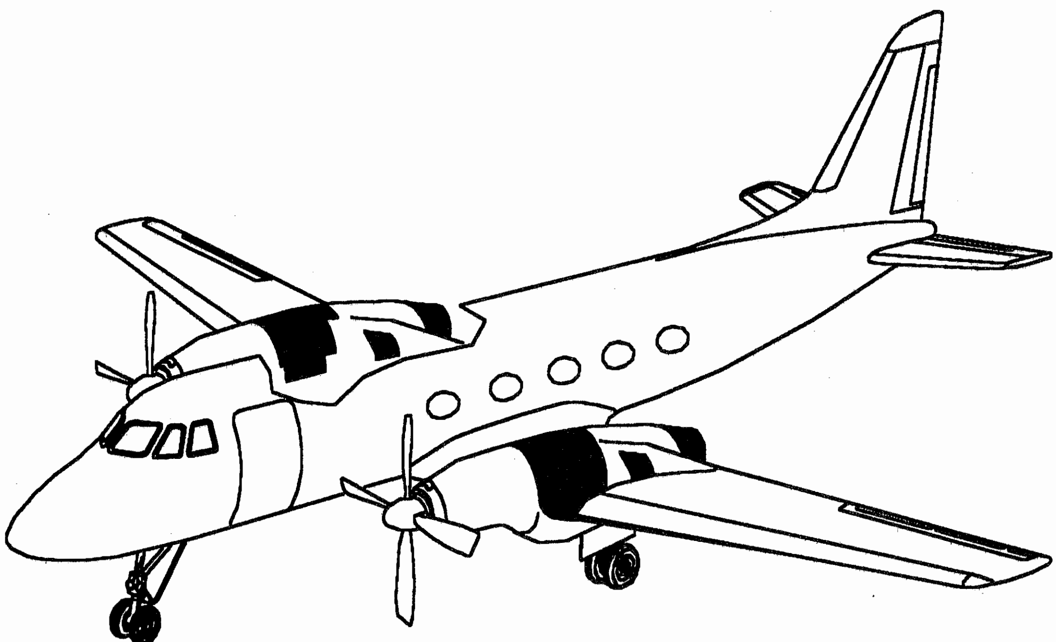
(24) Inspect for internal mount tube corrosion in all areas accessible to direct probe application and all areas within 1 inch of the tube end fittings and within 1 inch of all tube penetrations.

NOTE: Begin testing on aft end of lower tube accessible from inside of wheel well. If this portion of test is satisfactory, perform this portion of the test on each of the four Zee Truss assemblies before proceeding with the rest of the test.

To ensure no corrosion is present at the Jo-Bolts which attach the shelf to the lower tube of the truss assembly, a borescope inspection of this area is required. This will required flushing, inspecting and retreating the tube per CB 241C. This can be accomplished on the closed end tubes by drilling a 0.375 inch hole at the lower end of tube at the drain hole location described by Figure 8 of CB 241C. For maintenance convenience, borescope inspection may be delayed until all other areas of the mount have been tested and accepted for further service.

(25) For a mount tube subassembly to be considered to have negligible corrosion, all areas must be inspected.

(26) Failure of any one area is a failure of the entire mount subassembly.



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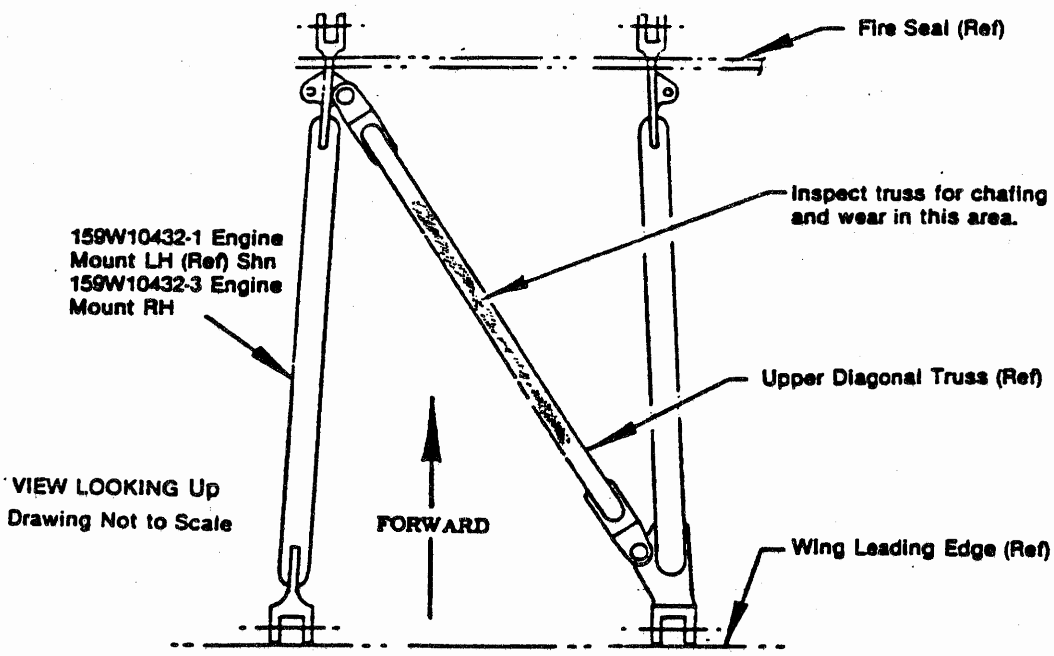
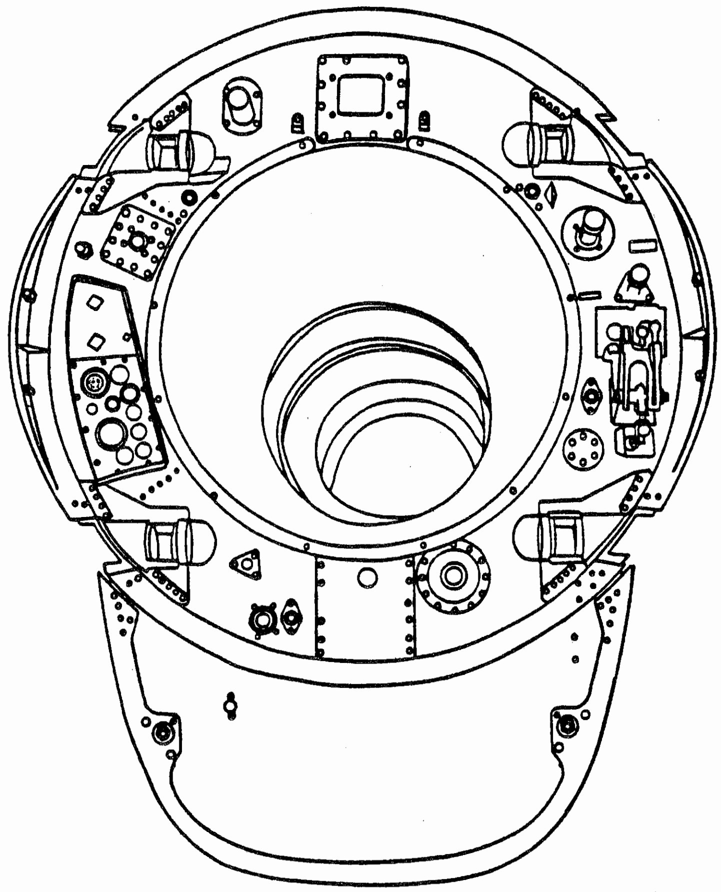
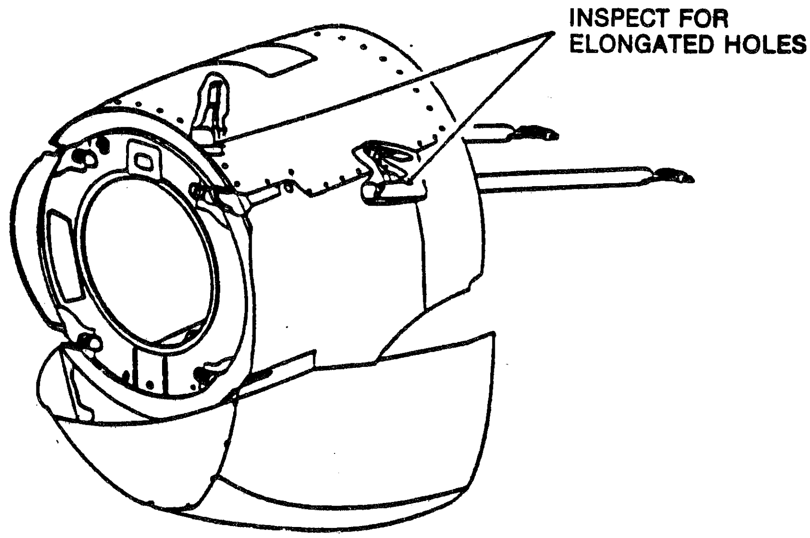
Engine Nacelle Inspection Area
Figure 201.



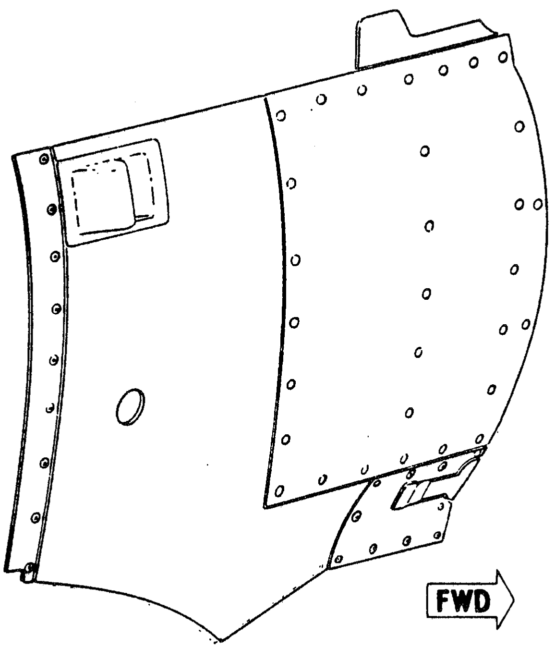
Inspection of Upper Diagonal Truss for Chafing
Figure 202.



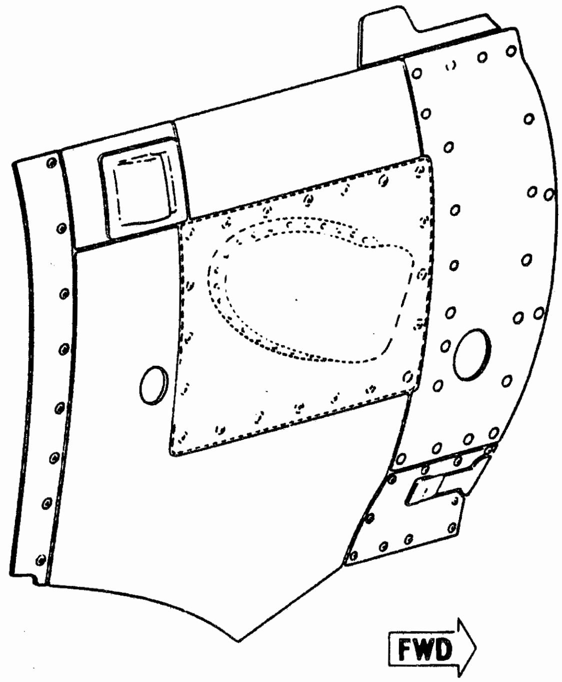
Engine Nacelle Firewall
Figure 203.



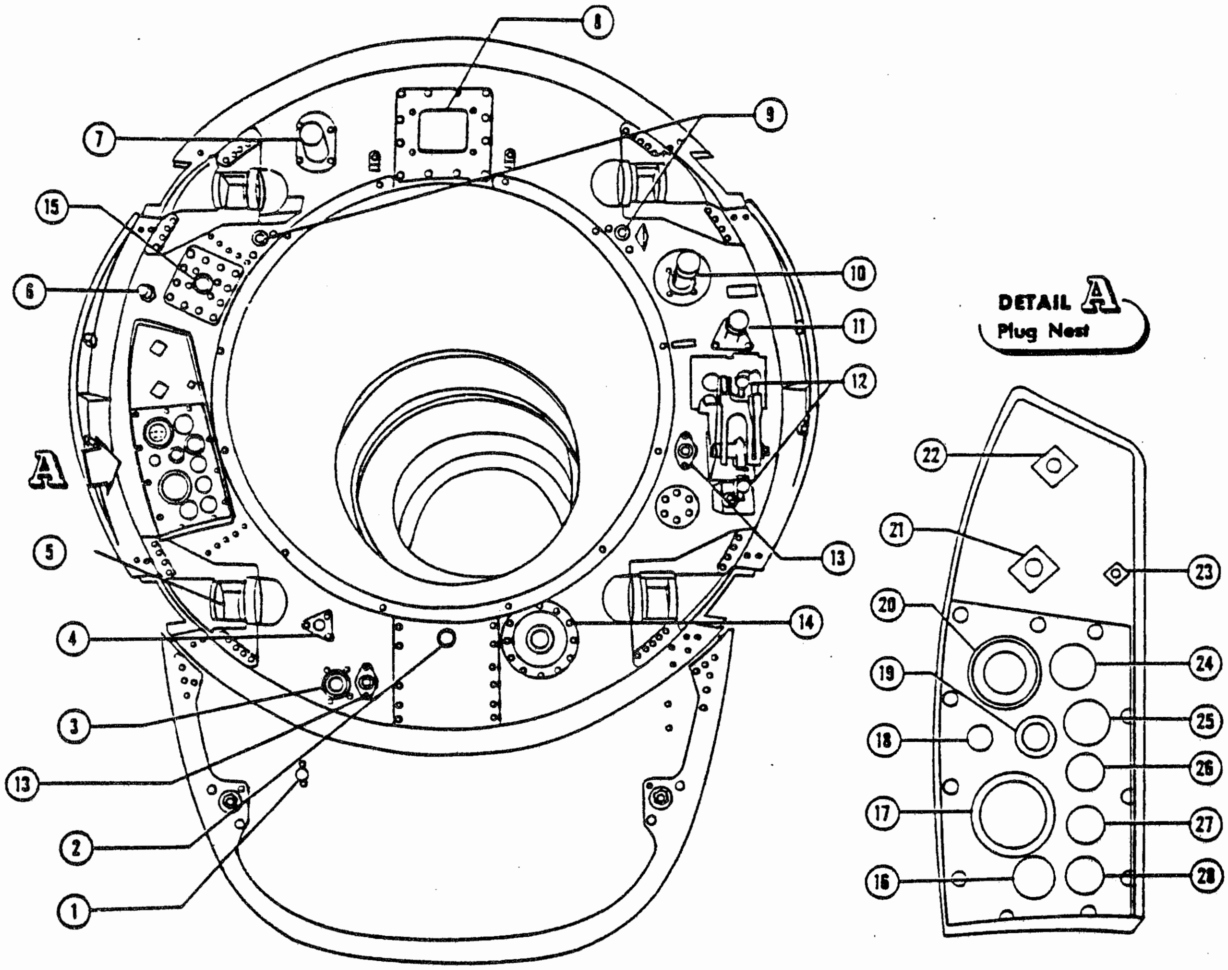
Engine Nacelle Inspection
Figure 204.



Left Nacelle Inboard
Figure 205.



Right Nacelle Outboard
Figure 206.



1. Engine Cowling Drain
2. Thermocouple
3. Main Engine Drain Line
4. Heat Shield Drain Line
5. Engine Mount (4)
6. De-icer
7. Engine Breather
8. Accessory Gearbox Drive Shaft
9. Fire Detection
10. Water/Methanol
11. Fire Extinguisher
12. Engine Controls
13. High Tension Ignition Lead
14. Main Fuel Line
15. Differential Pressure and Low Pressure Warning Switch Connector
16. 3rd Oil Line Connector
17. Air Intake De-icing and Propeller De-icing Connector
18. Propeller Hub Switch Connector (1 Lead)
19. Propeller Hub Switch Connector (2 Leads)
20. Port Harness, Oil Pressure Warning Light Switch, Hot Air Gate Valve, Oil Pressure Transmitter, Torque Pressure Transmitter, Oil Inlet Temperature Lead and Propeller Hub Switch (1 Lead) Connector
21. Positive Terminal Post-Starter Cable Connection
22. Positive Terminal Post-Feathering Pump Cable Connection
23. Negative Terminal Post-Starter Cable and Feathering Pump Cable Connection
24. Auto Feathering Switch Connector
25. Pulse Generator Connector
26. Pitch Coarsening Solenoid Connector
27. Flight Safety Lock Indicator Switch Connector
28. Increase Pressure Solenoid Connector

Firewall Connections
Figure 207.

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STABILIZERS

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CHAPTER 55

STABILIZERS

STABILIZERS — DESCRIPTION / OPERATION

1. General

The stabilizers consist of the vertical stabilizer and the left and right horizontal stabilizers. The vertical stabilizer is bolted to the fuselage frames at Fuselage Stations 635, 669, 690 and 708. The stabilizers are bolted to the fuselage frames at Fuselage Stations 669, 690 and 708.

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VERTICAL STABILIZERS — DESCRIPTION / OPERATION

1. Vertical Stabilizers

The vertical stabilizer consists of aluminum alloy beams, ribs and stringers covered with a stressed aluminum-alloy skin. The leading edge of the vertical stabilizer contains a flush-mounted de-icer boot and the vertical stabilizer cap houses the anti-collision light. The right hand side of the vertical stabilizer contains vortex generators.

Vortex generators were installed by ASC 41 to improve the aircraft single engine performance capabilities. If the vortex generators are missing or removed for any reason, the minimum control speed of the aircraft will be affected. Therefore, the generator(s) should be replaced at the operators earliest opportunity.

Vertical stabilizer attachment to the fuselage structure is made through a torque box assembly on the frames at Fuselage Stations 635, 669 and 690, from vertical stabilizer fittings mounted on the aft side of these frames. Attachment to fuselage structure is also made at the rudder support torque box by bolts which also pass through angle brackets at Fuselage Station 708. The vertical stabilizer skin is attached to the fuselage skin by a drag angle. A formed aluminum fairing is attached to the vertical stabilizer skin, fuselage skin and the air inlet duct fairing.

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VERTICAL STABILIZERS — MAINTENANCE PRACTICES

1. **Vertical Stabilizer — Removal / Installation**

   A. Removal

   (1) Remove rudder, see Rudder - Removal/Installation, Section 27-3-0.

   (2) Remove fin fairing (See Figure 201).

   (3) Remove drag angles.

   (4) Disconnect fin de-icer tubes.

   (5) Disconnect top anti-collision light leads on tail terminal strip at Fuselage Station 650.

   (6) Disconnect trim tab cables at turnbuckles.

   (7) Disconnect upper end of vertical gust lock pushrod and rudder control pushrod at Fuselage Station 669.

   (8) Remove bolts holding upper bracket to rudder torque tube.

   (9) Remove torque tube retainer ring.

   (10) Remove rudder torque tube assembly.

   (11) Attached fin hoisting sling (See Figure 202).

   (12) Remove attaching bolts at Fuselage Stations 635, 669, 690 and 708. (See Figure 201).

   (13) Remove vertical fin.

   B. Installation

   (1) Position fin attaching fitting on aft side of frames at Fuselage Stations 635, 669 and 690, and fin rudder support overhang between brackets on frame at Fuselage Station 708.

   (2) Replace attaching bolts at Fuselage Stations 635, 669, 690 and 708, (See Figure 201).

   (3) Remove fin hoisting sling.

   (4) Replace rudder torque tube assembly.

   (5) Replace retainer ring.

   (6) Replace bolts holding upper bracket to rudder torque tube.

   (7) Connect upper end of vertical gust lock pushrod and rudder pushrod.

   (8) Connect trim tab cables at turnbuckles.

   (9) Connect top anti-collision light leads on tail terminal strip at Fuselage Station 650.

   (10) Connect fin de-icer tubes.

   (11) Replace drag angles and fin fairing.

   (12) Install rudder, see Rudder - Removal/Installation, Section 27-3-0.

2. **Vertical Stabilizer / Fuselage Fitting — X-Ray Inspection**

   WARNING: USE NORMAL X-RAY SAFETY PRECAUTIONS DURING INSPECTION.

   NOTE: No formal procedure is given for X-ray due to varying techniques of X-ray technicians. However. full coverage of each fitting should be stressed. Ideally, one 900 and one 450 from both sides of fitting should be sufficient, but due to accessibility of fittings this may not be possible.

   A. X-ray following attach fittings: (See Figure 203)

   (1) Forward fitting

   (2) Mid fitting

   (3) Aft fitting

3. Vertical Stabilizer Hinge Support Fitting — Inspection

A. Remove rudder, see Rudder - Removal/Installation, Section 27-3-0.

WARNING: USE CLEANING SOLVENT IN A WELL VENTILATED AREA. AVOID PROLONGED INHALATION OF FUMES AND CONTACT WITH SKIN. KEEP SOLVENT AWAY FROM OPEN FLAMES.

B. Thoroughly clean areas to be inspected with solvent or equivalent.

C. Visually inspect fitting(s) for evidence of corrosion, cracks, wear and loose or missing fasteners.

D. Using a ball gage and micrometer, check bolt hole(s) for wear. Hole size should be 0.2500 to 0.2520 inch.

E. Carefully inspect control surface fittings and mating fitting for evidence of corrosion, wear, or chafing; check for signs of cracks, especially at areas adjacent to bearing and bolt attachment points.

F. Carefully check fittings for irregular surfaces under primer and for evidence of granular corrosion.

G. Inspect bearings for security of mounting, evidence of rust, and for roughness when rotated under load.

H. Replace worn, chafed, corroded, deeply scratched, or loose fittings.

I. Replace loose, rusted, or rough bearings.

NOTE: While rudder is removed, open rear vertical beam access panels and inspect interior structure of vertical fin for cracks, corrosion, cleanliness and general condition.

J. Replace rear vertical beam access panels.

K. Install rudder, see Rudder - Removal/Installation, Section 27-3-0.

4. Vertical Stabilizer / Rudder (Exterior) — Inspection

Inspect the following for:

A. Rudder, rudder trim tab, rudder spring tab, spring tab linkage and control cables for security of attachments.

B. Rudder left and right hand skin for evidence of corrosion, structural failure and freedom of operation.

C. Rudder static wicks for condition and security.

D. Navigation light for condition and security.

E. Exterior of dorsal fin and vertical stabilizer for condition of skin, evidence of structural failure and corrosion.

F. Ram air inlet duct for freedom from obstruction.

G. De-icer boots for condition and security.

H. Top anticollision light for condition and security.

I. Loose or working rivets in root rib area external skin. If loose or working rivets are found, drill out the working rivets and replace with next size up protruding head blind rivet.

J. External markings for legibility.

K. Trim tab hinges for cracks and freedom of movement. Vortex generators for condition and security

L. APU inlet duct for condition and freedom of Obstruction.

M. APU door for condition, security and proper operation.

5. Vertical Stabilizer / Rudder (Interior) — Inspection

A. Open access panels (shaded) on Figure 204

B. Inspect rudder seal for condition and security.

C. Inspect rudder and rudder spring/tab, trim tab actuator, counterweights and bearings for condition, security, evidence of corrosion and signs of failure.

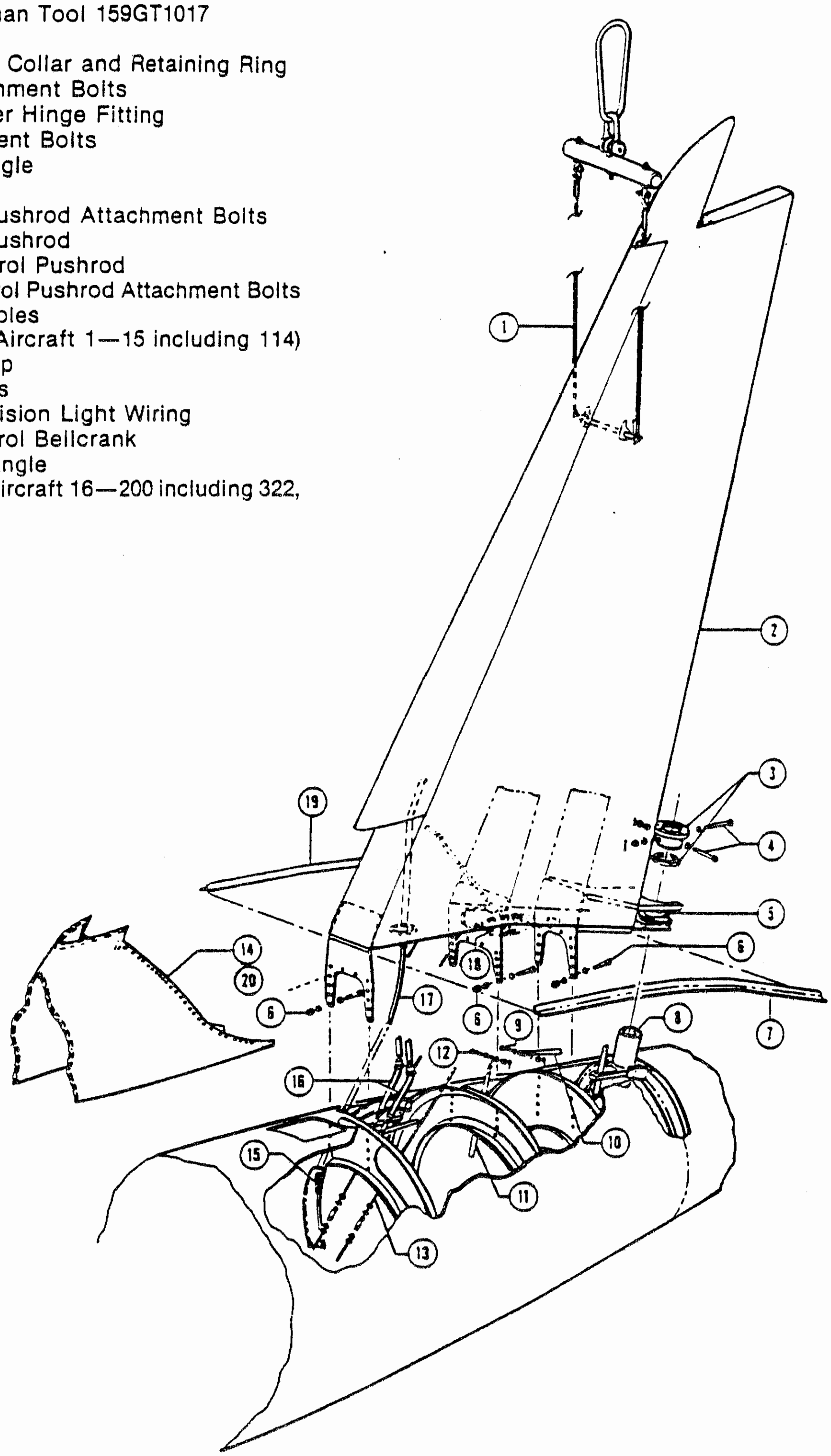
D. Inspect interior of vertical stabilizer through inspection holes for:

(1) Condition and security of structural members, adjacent skin and evidence of corrosion.
(2) Condition and security of flight control installation including cables, pulleys and fairleads.

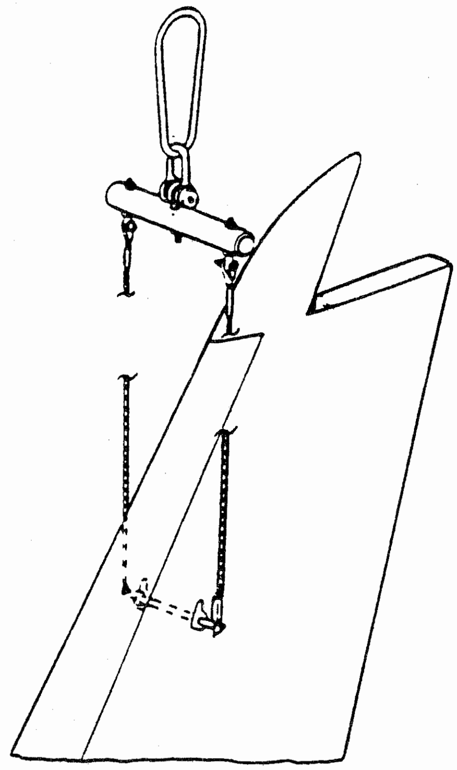
(3) Condition and security of electrical wires, connections and electrical plus.
(4) Condition and security of de-icer lines and connections.

E. Inspect rudder gust lock support brackets for cracks, (See Figure 205). If cracks are found, drawing 159RDCS141 is available for approved repair. The bracket may be replaced using blind rivets through the fin root rib web.

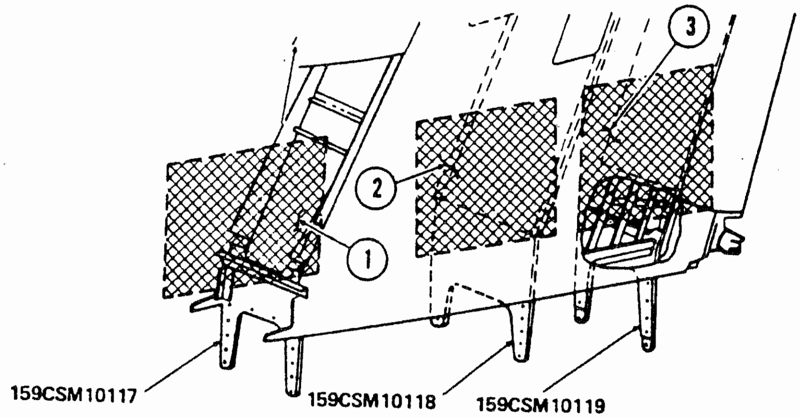
1. Sling-Grumman Tool 159GT1017
2. Fin
3. Torque Tube Collar and Retaining Ring
4. Collar Attachment Bolts
5. Lower Rudder Hinge Fitting
6. Fin Attachment Bolts
7. Left Drag Angle
8. Torque Tube
9. Gust Lock Pushrod Attachment Bolts
10. Gust Lock Pushrod
11. Rubber Control Pushrod
12. Rubber Control Pushrod Attachment Bolts
13. Trim Tab Cables
14. Fin Fairing (Aircraft 1—15 including 114)
15. Terminal Strip
16. De-icer Tubes
17. Top Anti-collision Light Wiring
18. Rudder Control Bellcrank
19. Right Drag Angle
20. Fin Fairing (Aircraft 16—200 including 322, 333)



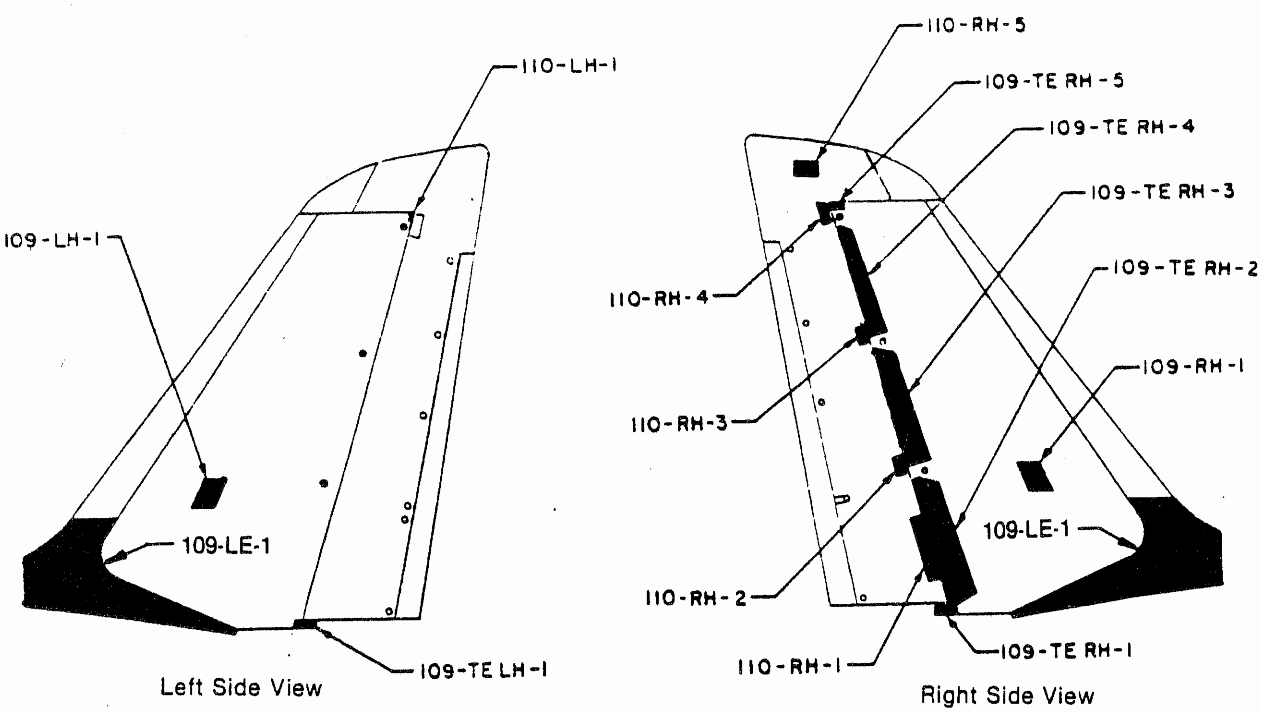
Vertical Stabilizer Installation
Figure 201.



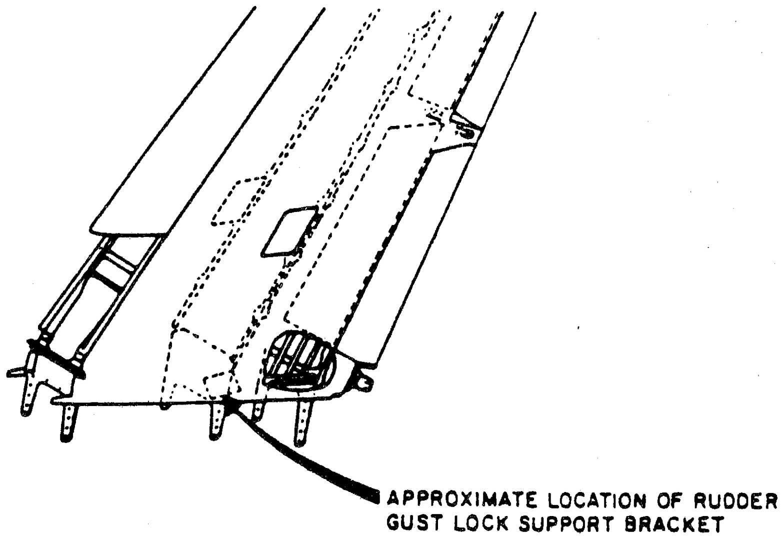
Vertical Stabilizer Hoisting Sling
Figure 202.



Vertical Stabilizer Fuselage fitting
Figure 203.



Vertical Stabilizer Access panel Locations
Figure 204.



Rudder Gust Lock Support Bracket Locations
Figure 205.

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HORIZONTAL STABILIZERS — DESCRIPTION / OPERATION

1. Horizontal Stabilizers

The stabilizers consist of aluminum alloy beams, ribs and stringers with a stressed skin covering. Each stabilizer is equipped with a flush type de-icer boot and tip cap. Hinged doors at the trailing edge of the lower skin are provided for access to the elevator seals, trim tab control cables and lower elevator stop adjustment bolt. A hinged door at the inboard end of the upper trailing edge provides access to the upper elevator stop adjustment bolt.

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HORIZONTAL STABILIZERS — MAINTENANCE PRACTICES

1. **Horizontal Stabilizer — Removal / Installation**

   A. Removal

   (1) Remove elevator, see Elevator - Removal/Installation, Section 27-1-0.

   (2) Remove drag angles at root of stabilizer. (See Figure 201).

   (3) Disconnect de-icer tubes.

   (4) Disconnect trim tab cables at turnbuckles.

   (5) Attach stabilizer hoisting sling (See Figure 202).

   (6) Remove attaching bolts at Fuselage Stations 669, 690 and 708, (See Figure 201).

   (7) Remove stabilizer.

   B. Installation

   NOTE: If original elevator is to be installed, bearing inspection may be preformed at this time, see Elevator Bearing/Fitting - Inspection,

   Section 27-1-0.

   (1) Position stabilizer at fuselage fittings and replace attaching bolts, Figure 201).

   (2) Remove sling.

   (3) Connect de-icer tubes.

   (4) Replace drag angles.

   (5) Install elevator, see Elevator - Removal/Installation, Section 27-1-0.

2. **Horizontal Stabilizer / Elevator Exterior — Inspection**

   Inspect the following for:

   A. Elevator and elevator trim tabs for security of attachment, condition of upper and lower skins, evidence of corrosion, structural failure and freedom of operation.

   B. Elevator and elevator trim tab attachment fittings and bearings for condition, security, signs of failure and evidence of corrosion.

   C. Elevator and elevator trim tab linkage, control rods, cables and mechanisms for condition, security and freedom of operation (as viewed with panels installed).

   D. Elevator static discharge wicks for condition and security.

   E. De-icer boots for condition and security.

   F. Exterior of horizontal stabilizers for condition of upper and lower skins, evidence of corrosion and structural failure.

   G. Root rib area external skins for loose or working rivets. If loose or working rivets are found, drill out the working rivets and replace with next size up protruding head blind rivet.

   H. External markings for legibility.

   I. Inspect elevator trim tab horn fittings for excessive play. Relative movement between the stabilizer hinge fittings and the tab hinge fittings should barely be detectable. If excess play is detected, disassembly should follow to determine cause.

   NOTE: Inspect for repair bushing security, the hole in the bushing(s) must be line reamed with corresponding hole in the opposite lug to 0.2495 - 0.2505 inch diameter.

   J. Inspect elevator trim tab fittings for relative motion to the tab honeycomb tapered section. Fitting should be solid, reject fittings if any relative motion is found.

3. **Horizontal Stabilizer / Elevator Interior — Inspection**

A. Open access panels (shaded) on Figure 203 and inspect interior for:

B. Condition, security and evidence of corrosion of seal curtain, structural members, counterweights and adjacent skin.

C. Elevator and elevator trim tab bearings for condition, security and signs of failure.

D. Clearance between trailing edge of elevator and leading edge of elevator trim tab. The clearance should be at least 0.060 inch measured at any point across the tab, top or bottom, and through its entire range of travel. If the clearance is less than 0.060 inch, bend leading edge of tab to obtain the correct clearance.

E. Elevator for excessive free play. Move control column to elevator down position and hold in this position. Have someone move elevators by holding rear beam of elevator. Do not attempt to move elevator by holding trim tab. Maximum allowable free play is 1/8-inch trailing edge displacement. A resulting abnormal movement or sound is an indication of wear. Move control column to elevator up position and repeat free play check. Contact Gulfstream Aerospace Technical Services for repair procedures, if required.

NOTE: Two areas of known wear are the stop fittings on the left and right side of elevator torque tube at vertical attachment bolts and in the 1/4 inch bolts holding the crank to the tube in the 159C10111 crank assembly at Fuselage Station 686.

F. Elevator and elevator trim tab linkage, control rods, cables and mechanism for condition, security and freedom of operation.

G. Elevator trim tab actuators for condition, security and signs of failure.

H. Interior of horizontal stabilizers through inspection holes for:

(1) Condition, security and evidence of corrosion of structural members and adjacent skin.

(2) Condition and security of flight control installation including cables, pulleys and fairleads.

(3) Condition, security and evidence of corrosion of deicing lines and connectors.

(4) Presence of foreign objects, loose hardware and accumulation of dirt and/or fluid.

I. Inspect for presence of foreign objects then close access panels.

4. **Horizontal Stabilizer Attach Fitting — X-Ray Inspection**

WARNING: USE NORMAL X-RAY SAFETY PRECAUTIONS DURING INSPECTION.

NOTE: No formal procedure is given for X-ray due to varying techniques of X-ray technicians. However, full coverage of each fitting should be stressed. Ideally one 900 and one 450 from both sides of fitting should be sufficient, but due to accessibility of fittings this may not be possible.

A. X-ray following attach fittings. (See Figure 204)

(1) Left forward fitting.

(2) Left mid fitting.

(3) Left aft fitting.

(4) Right forward fitting.

(5) Right mid fitting.

(6) Right aft fitting.

5. Elevator Hinge Fitting — Inspection

A. Remove elevator, see Elevator - Removal/Installation, Section 27-1-0.

B. Inspect control surface fittings and mating fitting for evidence of corrosion, wear, or chafing. Check for cracks adjacent to bearing and bolt attachment points and loose or missing fasteners.

C. Using a ball gage and micrometer, check the bolt hole(s) for wear. Hole size should be 0.2500 - 0.2520 inch.

D. Inspect rear beam web and rib attachment angle as follows:

(1) Remove seal attachment.

(2) Open covers on 3 1/2 inch diameter access holes in rear beam web.

NOTE: Use inspection mirror and flashlight.

(3) Inspect for cracks or damage to beam web, rib, rib attaching angle and fasteners.

(4) Inspect external area of rear beam, cap angles, overhang panels (top and bottom) and supporting bracketry.

E. Inspect fittings for irregular surfaces under primer and evidence of corrosion.

F. Inspect bearings for security, corrosion, and roughness when rotated under load.

G. Replace worn, chafed, corroded, deeply scratched, or loose fittings.

H. Replace defective bearings.

I. Install elevator, see Elevator - Removal/Installation, Section 27-1-0.

6. Elevator Stop Fitting — Removal / Installation

NOTE: Replacement of these fittings requires Gulfstream Aerospace Engineering supervision, therefore this procedure is listed for premature removal purposes only.

A. Removal.

(1) Remove elevator, see Elevator - Removal/Installation, Section 27-1-0.

(2) Remove hinge fitting/stop fitting.

B. Installation.

(1) Install hinge fitting/stop fitting.

(2) Install elevator, see Elevator - Removal/Installation, Section 27-1-0.

7. Elevator Stop Fitting — Inspection

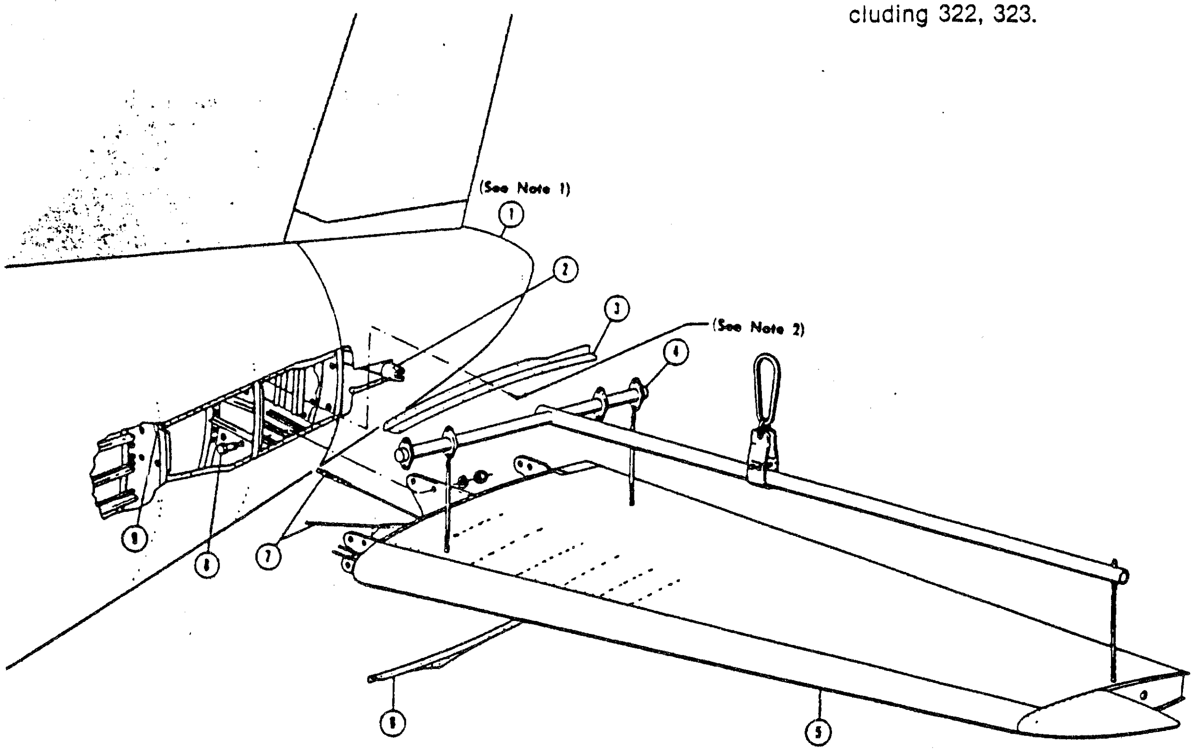
A. Remove elevator(s), see Elevator - Removal/Installation, see Section 27-1-0.

B. Visually inspect stop fitting for evidence of corrosion, cracks, wear, loose or missing fasteners and proper installation.

C. Inspect condition of stop bolts. Stop bolts should be tight and locking nuts should be locked in place.

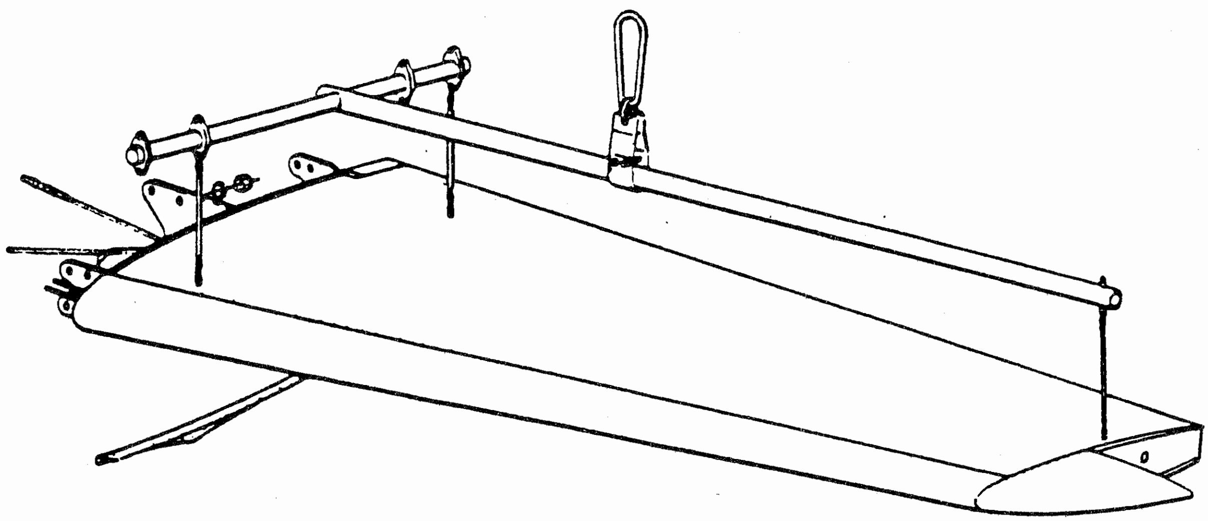
D. Install elevator, see Elevator - Removal/Installation, Section 27-1-0.

NOTE:
1. Tail Cone is removable on Aircraft 8—200 Including 322, 323.
2. Screws will be replaced by rivets on Aircraft 10—200 Including 322, 323.

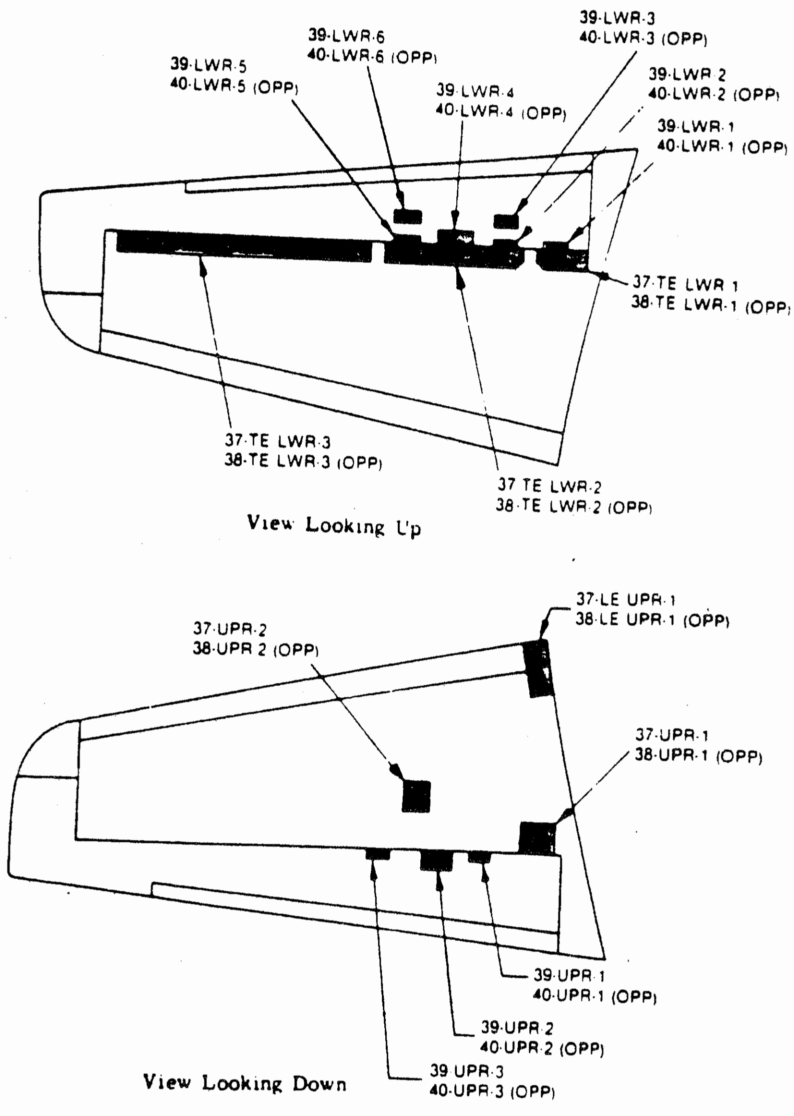


1. Aft Fuselage
2. Elevator Torque Tube
3. Upper Drag Angle
4. Sling - Grumman Tool 159GT1018
5. Stabilizer
6. Lower Drag Angle
7. Trim Tab Cables
8. Attaching Bolts
9. De-icer Tubes

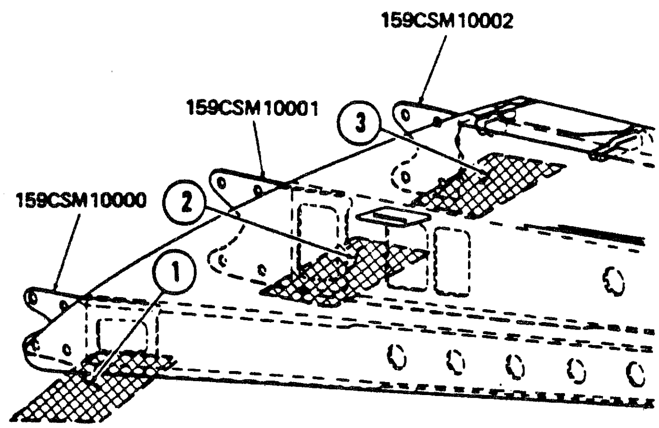
Horizonal Stabilizer Installation
Figure 201.



Horizontal Stabilizer Hoisting Sling
Figure 202.



Horizontal Stabilizer / Elevator Interior Inspection Panel Location
Figure 203.



Horizontal Stabilizer Attach fitting Location
Figure 204.

LIST OF EFFECTIVE PAGES

CHAPTER 56

WINDOW

* Pages revised by Revision 43
** Pages added by Revision 43
*** Pages deleted by Revision 43

TABLE OF CONTENTS

CHAPTER 56

WINDOW

WINDOWS – GENERAL

1. **Cockpit Windows**

CAUTION: OPERATORS MUST USE CAUTION TO GUARD AGAINST SCRATCHING THE ACRYLIC PORTIONS OF THE WINDOWS. HARD, SHARP ITEMS SUCH AS LOGBOOKS, CHECKLISTS, APPROACH PLATE HOLDERS, AND INSTRUMENT HOODS, WHETHER MADE OF METAL OR NOT, SHOULD NOT BE PLACED ON THE GLARESHIELD. SIMILARLY, CARE MUST BE TAKEN WITH SUNSCREENS AND OXYGEN MASKS OR OTHER ITEMS COMMONLY HUNG NEAR ACRYLIC PANELS.

The cockpit windows are made of several layers of tempered glass and vinyl plastic incorporated into a failsafe design to withstand cabin pressure. The front windshields and direct vision windows are also designed to withstand bird impact. The windows are heated to provide deicing. (For windshield heating requirements, refer to Chapter 30.)

2. **Cabin Windows**

The passenger cabin is provided with five windows on the left side of the cabin and five on the right side. Two windows on each side, over the wing area, may be opened from inside or outside to provide for emergency exits.

A fixed lavatory window is located on the aft end of the fuselage, on the right side. Its construction is identical with that of the cabin windows.

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COCKPIT WINDOWS — DESCRIPTION / OPERATION

1. General

The cockpit windows are made of several layers of tempered glass and vinyl plastic incorporated into a failsafe design to withstand cabin pressure. (See Figure 1) The front windshields and direct vision windows are also designed to withstand bird impact.

Windshield Panels — The left and right front windshield panels consist of three tempered glass panes, two layers of vinyl plastic and one layer of PPG 112 interlayer as shown in Figure 1. Panels are electrically heated by means of a conductive (NESA) coating on the inboard surface of the outboard glass sheet. This provides structural strength for bird impact and serves as a means for de-icing. (See Figure 2 for installation.)

Direct Vision (DV) Windows — The left and right direct vision windows consist of three tempered glass panes, two layers of vinyl plastic, and one layer of PPG 112 interlayer as shown in Figure 1. Panels are electrically heated by means of a conductive (NESA) coating on the inboard surface of the outboard glass sheet. This provides structural-strength for bird impact and serves as a means for deicing. An interior handle arrangement, consisting of three locks, unlocks the window and permits its sliding aft in a track arrangement. This provides visibility if the electrical heating system fails. The three locks consist of a center handle which operates three rollers, an upper handle which operates a pin and a lower handle which operates a sliding latch. For ease of operation when closing the DV window, the center handle should be locked first, the upper handle second and the lower handle last. When opening the window, reverse the procedure.

The 112 interlayer is prone to sulfuric acid attacks, which can appear as cracking or crazing. This condition was confined to the 0.030 inch thick interlayer, on the edge seal around the periphery, where the glass contacts the metal retainer. The crazing is a cosmetic problem only and doesn't affect the structural integrity of the window.

Side Windows — The left and right fixed side windows consist of two tempered glass layers. Two PPG 112 interlayers and one layer of vinyl plastic as shown in Figure 2. The panels are electrically heated by means of a conductive (NESA) coating on the outboard side of the inboard glass layer. The windows are not heated for bird impact but for deicing and crew comfort. (See Figure 1).

2. Cockpit Windows — Service Defects

The following information is presented to assist in identification and evaluation of defects that may develop in service. Proper consideration of defects will tend to reduce the number of panel failures in flight and help prevent unnecessary panel replacement. The most common types of defects are as follows:

NOTE: See aircraft flight manual for aircraft operating limitations with windshield defects.

A. Scratches — A scratch is a line-type defect in the surface of the glass and can be classified as hairline, light and heavy. Most scratches can be avoided by proper handling and care of windows. When performing maintenance on aircraft near the windows, they should be suitably protected. Windows should be cleaned with a mild detergent and water, using a clean, soft, cotton cloth, followed by drying with a clean, soft, cotton cloth. Cleaning solvents should not be used on the windows. Hairline scratches are usually caused by improper cleaning, for example, using a coarse cloth to dry the glass. Hairline scratches may be less noticeable by cleaning and waxing the glass. Light scratches are defined as less than 0.010 inch deep and heavy scratches are 0.010 deep or more. Heavy scratches are likely to have edge chips along the length of the scratch. Scratches should be inspected periodically to determine if the damage is progressive: for example, watch for crack development. A windshield with scratches will normally be replaced because of impaired vision before the windshield strength is affected. A light scratch up to 0.003 inch deep may be polished out using cerium oxide polishing powder mixed with water or oil. An all wool felt polishing wheel about 4 inches in diameter by 3/4 wide should be used. The scratch being polished should be blended with the surrounding glass in order to prevent distortion in the panel. The glass also should not be allowed to overheat as this will cause distortion in the glass. Relatively wide strokes with the polishing wheel (drill motor speed not more than 2700 rpm) should be used. When viewing the windshield panel for distortions, look through (not at) the glass. A grid of 1/2 inch spaced lines on a white background will help in evaluating distortion.

B. Delaminations – Delamination is a separation of the glass from the vinyl laminate and is indicated by bubble-type separation. Delamination may be caused by laminating stresses (manufacturing) by installation stress, or by aircraft operational stresses. Delamination from these stresses tend to relieve such stresses with consequent less danger of failure of the individual glass piles. The strength of the windshield in bending or in tension is not affected and a panel would be removed for this reason before an unsafe condition caused loss of windshield strength. Delaminations have characteristics that tend to divide them into types. The characteristics are clear or cloudy smooth-edge or rough-edge and paring-medium or island.

- Clear or Cloudy Delamination – Delamination is normally apt to be clear. Cloudy delamination, which is usually progressive, will result if moisture penetrates the vinyl in the delaminated area.
- Smooth-edge or Rough-edge Delamination – Smooth-edge delamination advances with a smooth boundary. It does not have a rough or jagged area. Rough-edge delamination is characterized by its irregular, jagged boundary and may develop long, finger-like projections. These projections may pull chips out of the glass surface.
- Parting-Medium or Island Delamination – The parting-medium is a thin film of brownish plastic material applied in a narrow strip between the glass panes and the vinyl ply around the edge of the panel. The purpose of the parting-medium is to provide a more flexible bond along the edges of the panel assembly. A low degree of adhesion is intended in the area due to the different expansion and contraction rates of the glass and vinyl. Parting-medium delamination develops in this area. Island delamination does not extend to the edge of the glass pane and may occur after a large delamination has formed and then closed up along the edge of the panel.

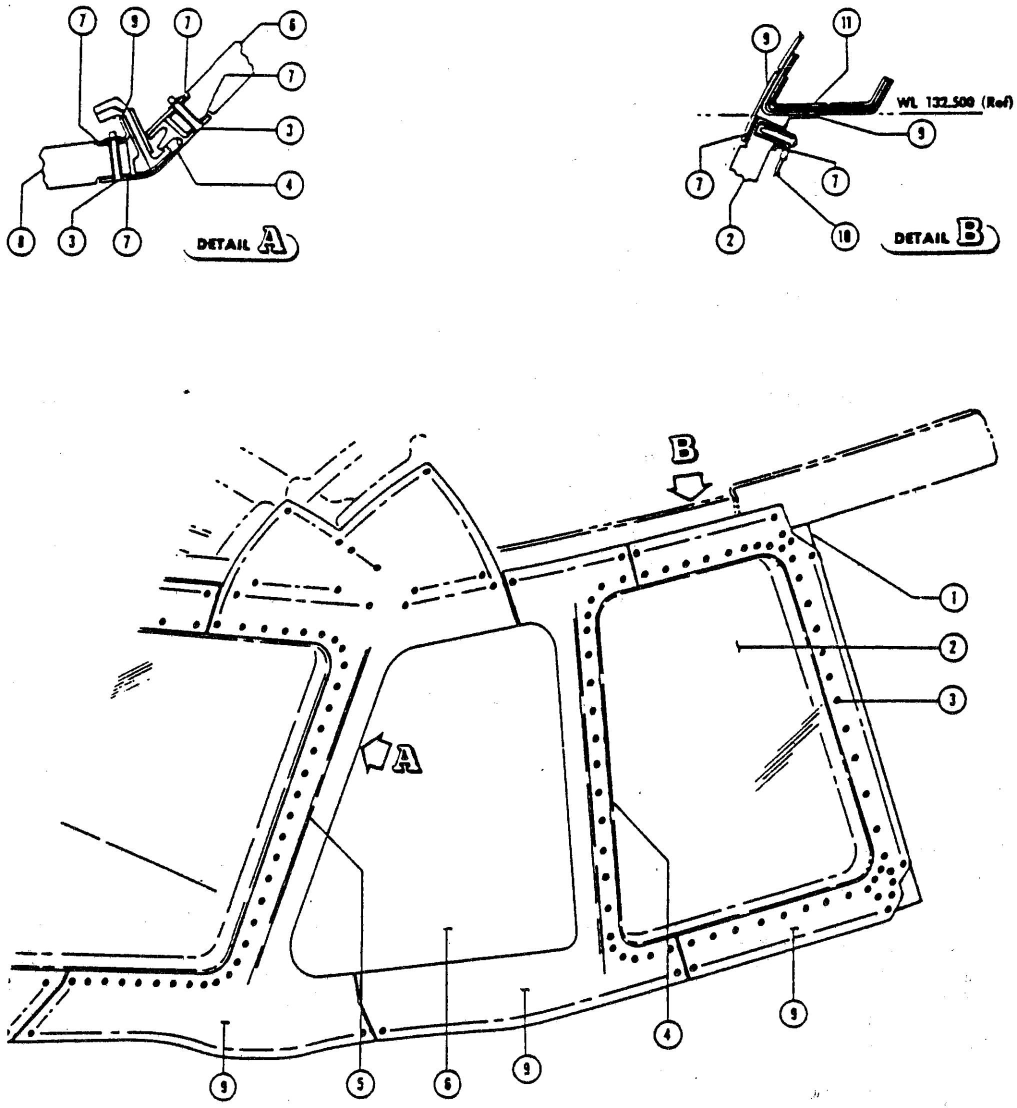
C. Chips – Chips are classified as inner-surface chips and outer-surface chips as follows:.

- Inner-surface Chips – Inner-surface chips occur within the panel at the vinyl and glass bonding surface. These chips or flakes of glass are pulled from the glass pane by the vinyl ply. Inner-surface chips are usually associated with rough-edge delamination. The inner-surface chips are critical because they may result in cracking of the glass or arcing of the NESA heating film. Windshields with inner-surface chips should be replaced at the first opportune time.
- Outer-surface Chips – Chipped areas or nicks in the outer two surfaces of the panel assembly are defined as outer-surface chips. Occasionally, a piece of glass may flake off at the edge of the glass pane: however, more often outer-surface chips are caused by the glass being struck by a foreign object. A windshield with a chip flaked off, leaving a smooth defect in the glass, should be inspected periodically to determine if the damage is progressive. Windshields that have deep, jagged chips should be replaced at the first opportune time.

D. Cracks – Small cracks in either of the outer glass panes are critical because it cannot be predicted when the crack will extend across the glass. A windshield with a crack should be replaced at the first opportune time. If the fully tempered center glass pane is cracked, it will break into cubic fragments all over and the windshield should be replaced before the next flight.

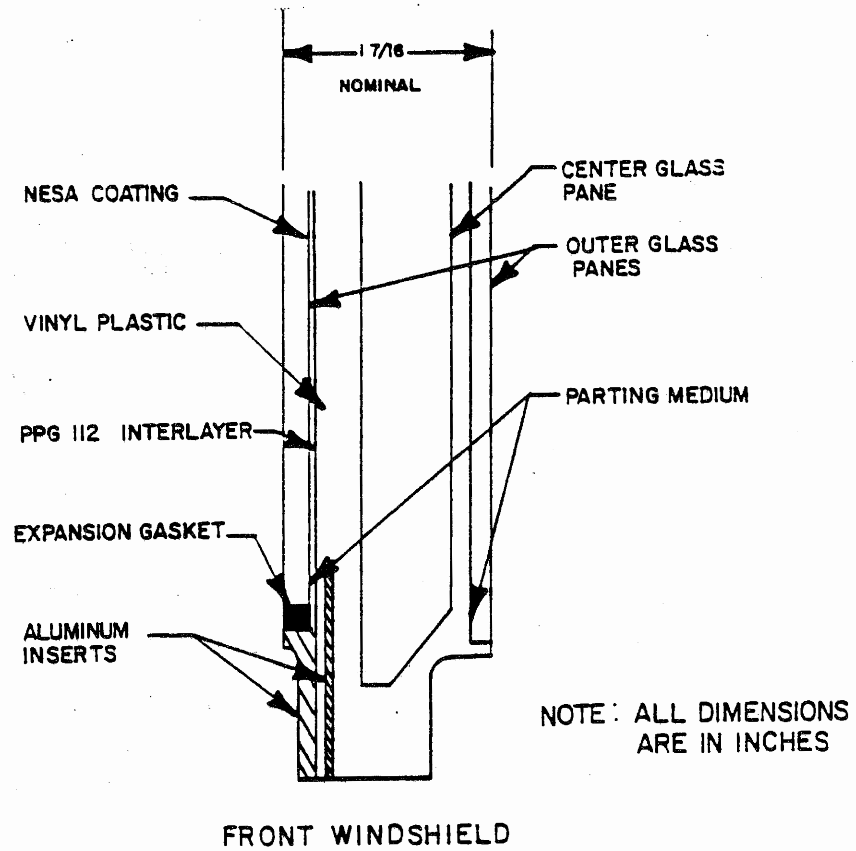
E. Bubbles – Bubbles occasionally occur in the vinyl and are usually indicative of overheating of the windshield. A few scattered bubbles are acceptable. However, if numerous bubbles develop, the panel would normally be replaced because of impaired vision. The windshield heat system should be checked out thoroughly when replacing a windshield with bubbles.

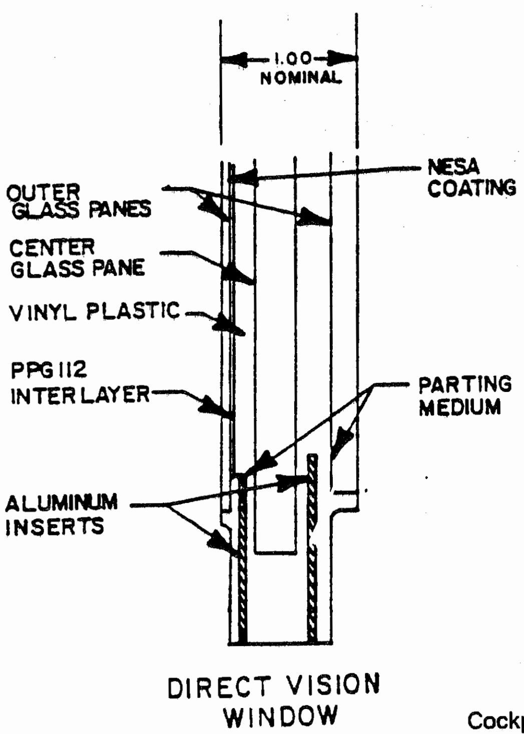
F. Arcing – Arcing is an electrical fault and appears as a dark or burnt spot in the panel. Arcing is usually associated with delamination and the NESA coating has been broken. In most instances of delamination, however, arcing does not occur. Windshields that have an arcing fault should be replaced at the first opportune time.

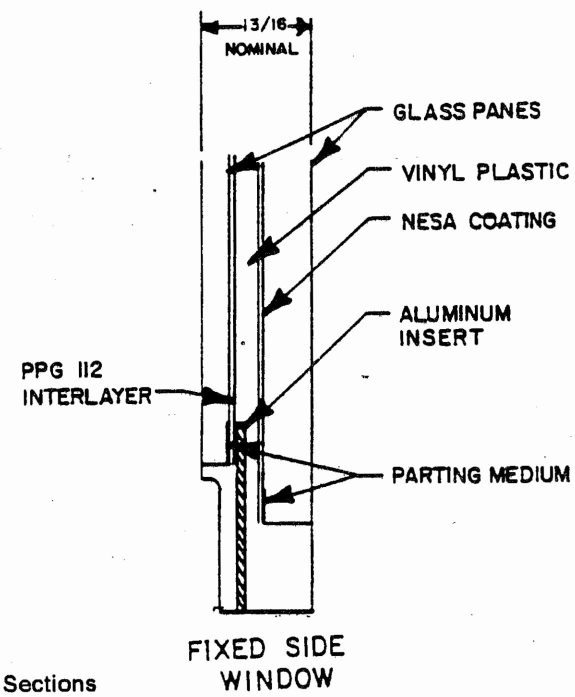


1. Frame Assembly
2. Fixed Panel
3. Window Attachment Screws
4. Post No. 3 Assembly
5. Post No. 2 Assembly
6. Direct Vision Window
7. Gasket
8. Windshield
9. Retainer
10. Post No. 1 Assembly
11. Terminal Block & De-icing System Electrical Lead
12. Cover and Rail (Upper)

Windshield Installation
Figure 1.







Cockpit Windows — Sections
Figure 2.

COCKPIT WINDOWS — MAINTENANCE PRACTICES

1. **Cleaning Procedures**

CAUTION: DO NOT USE BRUSHES DURING THIS PROCEDURE. A CLEAN, SOFT, LINT-FREE CLOTH SHOULD BE USED. FREQUENTLY CHANGE AREA OF CLOTH DURING CLEANING.

A. Windshield Cleaning Procedures

(1) Remove all excessive amounts of dirt and other substances from the glass surface (flush with a clean water rinse).

(2) Use either a mild soap and water, a 50/50 solution of isopropanol and water, ammonia and water, or cleaners such as Windex. Do not use any abrasive materials other than pumice or any strong acids or bases during the process.

(3) For stubborn areas, clean panel with Permatex 403D or mineral spirits. Follow with washing as per Step above.

(4) Rinse thoroughly and dry. Do not apply wax.

B. DV/Side Window Cleaning Procedures

(1) Remove all excessive amounts of dirt and other substances from the plastic surface with clean water.

(2) Use either a mild soap and water, a 50/50 solution of isopropanol and water, or aliphatic naptha type 2. Do not use any abrasive materials, strong acids or bases, methanol, or methyl ethyl ketone. When wiping the surface, use a soft cloth or sponge and use a straight rubbing motion.

(3) For stubborn areas, clean panel with Permatex 403D or mineral spirits. Follow with washing as per Step above.

(4) Rinse thoroughly and dry.

(5) After cleaning, plastic surfaces may be polished by applying a thin coat of hard polishing wax. Rub lightly with a soft cloth, using a circular motion.

2. **Windshield — Removal / Installation**

A. Removal

NOTE: Before removal of old windshield, inspect new windshield to ensure that it is correct side, free from delaminations, blemishes or damage. Check electrical circuits for proper resistance values. If resistance is different from present windshield, change transformer rectifier taps for that windshield to correct setting.

(1) Remove windshield wiper blade arm and link for windshield to be replaced.

(2) Punch screw pattern into cardboard or heavy wrapping paper so as not to get screw lengths mixed up.

CAUTION: IF AN AIR DRIVEN SCREW GUN IS USED TO REMOVE SCREWS, EXERCISE CAUTION AS SCREW HEADS MAY BE DAMAGED. EXCESSIVE PRESSURE APPLIED TO SCREWS WILL CAUSE BLIND FASTENER INSERTS TO BE DISLODGED. A SCREWDRIVER IS RECOMMENDED TO BREAK SCREWS LOOSE.

NOTE: Gulfstream Aerospace has windshield installation kits available which contain screw pattern, installation hardware, and liquid shim installation materials.

(3) Remove outer retainer screws.

(4) Cut sealant free of outer retainers and remove retainers.

(5) Clean outer retainers thoroughly.

(6) Check position of pressure seal springs, both top and bottom, at outer retainer joints (butt ends) so as to position correctly on reinstallation.

(7) Remove center post interior fairing and disconnect electrical leads.

(8) Remove window.

B. Installation (Liquid Shim Procedure)

CAUTION: PLACE PROTECTIVE COVER ON OUTSIDE OF OPPOSITE WINDSHIELD GLASS.

(1) If form-In-place gasket (soft gasket) is found on inner retainer angles, thoroughly clean inner retainers removing all traces of gasket.

(2) If liquid shim is found on inner retainer angles, clean shim thoroughly and inspect shim for deterioration, cracks, chipping, voids and adhesion to retainers. If general condition of shim is good, damaged areas may be repaired by following procedure in Step (17) (a) below.

(3) Inspect existing terminal clearance holes in center post retainer angle for proper clearance. (See Figure 201).

(4) Following rework will be required on five terminal (dual sensor) windshield installation due to thickness of glass. (See Figure 201 and Figure 202).

(a) Inspect center post retainer angle for addition of fifth terminal clearance hole. Add fifth clearance hole if it does not exist. See Figure 201 for location and dimensions.

(b) Remove center post interior fairing attach clip if a five terminal (dual sensor) windshield is to be installed. (This clip will interfere with added power terminal), see Figure 201.

NOTE: Center mount holes in center post fairing will no longer be required. Fill holes with ProSeal No. 890 or equivalent filler. Dress and smooth surface, paint as required.

(c) Remove glareshield P/N 159F10156-11.

(d) Remove one clip (P/N 159F10155-25 for left side rework or 159F10155-26 for right side rework) by drilling out four attachment rivets, see Figure 202.

(e) The 159F10155-13, 14, 17 and 18 angles must be trimmed approximately 3/8 to 1/2 inch to provide adequate clearance between glass and structure. (159F10155-13 and 17 for left side rework or 159F10155-14 and 18 for right side rework), see Figure 202.

(f) Carefully place new windshield into cavity to determine if sufficient clearance has been provided between 159F10155-13 and 17 or 14 and 18 angles and glass on inboard side. If clearance is inadequate, additional trimming of goggles will be required.

(g) Remove rubber bumper from glareshield and lay glareshield in place behind windshield. (Place new windshield in cavity if it was removed for additional trim rework in Step (4)(e) above.)

(h) In order to provide clearance between glareshield and inboard side of glass, determine amount of material to be removed from leading edge of glareshield, allowing additional clearance for rubber bumper. Remove and trim glareshield as required.

(i) Rebond rubber bumper on glareshield leading edge. (Use EC1357 or equivalent rubber cement.)

(j) Remove new windshield and install clip 159F10155-911 with four MS20470 AD4 rivets. See Figure 202. Alodine bare metal and coat with primer. Open areas may be closed with suitable filler as desired for cosmetic appearance.

(k) Install reworked glareshield and paint all reworked areas to match cockpit interior.

(5) Replace stake on type connectors on power and sensor leads with poke home connectors. Use P/N 1841-1-5616 on large terminals and P/N 1841-1-5620 on small terminals. Wire splicing may be required to accommodate No. 2 sensor.

(6) Install poke home connectors in windshield junction terminal position. Cover connectors with shrink tubing insulation.

(7) Inspect windshield retainer angles to determine any out of plane flatness conditions as follows:

NOTE: To adequately perform this inspection, a flat aluminum plate is required. Inspection plate and alignment pins can be manufactured or may be obtained from Gulfstream Aerospace

Service Representative. See Figure 203 for inspection plate and alignment pins manufacturing instructions.

(a) Install two alignment pins in each of four inner retainers. (See Figure 203 for pin locations.)

(b) Install flat aluminum inspection plate using alignment pins as a guide. Ensure that plate has bottomed against inner retainers.

(c) Insert feeler gage between plate and inner retainer. Inspect entire retainer plane for flatness and record findings. If a gap of 0.020 inch or less exists, proceed with instructions in Step (22). If gap is greater than 0.020 inch, liquid shimming of window inner retainers will be required. Proceed with next Step.

(8) Sand retainer angles with light abrasive using No. 180 to 320 grit silicone carbide or aluminum oxide abrasive paper.

(9) Wipe retainers with a clean wiper saturated with clean solvent. Wipe dry with a clean wiper.

(10) Using 1/4 inch wide masking tape, mask over 3/16 inch bolt holes in windshield retainer angles.

(11) Apply a liberal amount of release agent to under side and sides of inspection plate and eight alignment pins.

(12) Trowel heavy bead of EPON 934, Aerobond 2143 or equivalent liquid shim on retainer angles around entire periphery of windshield opening.

NOTE: Approximately 300 grams of liquid shim is required to adequately shim one windshield cavity. See Table I, for preparation and cure data of liquid shim.

(13) Install eight alignment pins in windshield retainer as referenced in Figure 203.

CAUTION: EVERY ONE HALF HOUR BREAK ALIGNMENT PINS FREE TO PREVENT PINS FROM BEING WELDED IN ATTACHMENT HOLES.

(14) Carefully place inspection plate into window cavity. Apply a steady even pressure to center of plate until excessive shim material has been forced out and maximum thickness of shim, in flat areas where no gap existed during flat plane inspection in Step (7), is 0.030 inch. Remove excessive shim material from around window retainers.

NOTE: Do not rock plate to force out excessive shim material or uneven, out of plane shim will result. MAXIMUM thickness of shim in flat areas must not exceed 0.030 inch or difficulty will be encountered when installing outer retainer in Step (28) below.

(15) When shim has cured to a pliable state, carefully remove inspection plate.

NOTE: Less time is required to trim shim in a pliable state because it can be cut with a sharp tool. When allowed to cure to its hardened state, shim may crack, break or chip during trimming procedure. If shim has been allowed to cure to hardened state, trim carefully with a file or suitable rotary tool. Avoid using percussion methods or tools. Normal liquid shim mixture will cure to a pliable state in approximately three hours.

(16) Trim remaining excess hard shim from window opening with caution, and remove shim material from window terminal clearance holes.

(17) Clean inspection plate and install plate to check shimmed retainer plane for a flat, even surface as in Step (7) above, except use following procedure if second application of liquid shim is necessary.

(a) A second application of liquid shim may be required to fill voids, shim damaged during removal of inspection plate or trimming process and for repair of old shim.

NOTE: To prevent exceeding 0.030 inch shim thickness in flat plane areas, second application need only be applied to areas where voids exist, shim material is missing or repairs are necessary provided a flat, even plane is end result on all retainer angles after rework.

(18) Carefully remove inspection plate and install old window into cavity.

CAUTION: SET DRILL STOP AT A POSITION WHICH WILL PREVENT DRILL FROM PASSING THROUGH RETAINER INTO PLATE NUTS.

NOTE: Leave alignment pins in place.

(19) Using a drill stop and a No. 11 drill (0.190 to 0.196 inch), drill through liquid shim to open screw holes in retainer angles.

(20) Remove old windshield and clean debris from window cavity, retainers and screw holes.

(21) Perform final inspection of shim to ensure it has not been damaged during drilling procedure.

(22) On inboard side of windshield apply a thin coat of Vultax 2-5-23A or suitable adhesive and position inner gasket 159CE10005-15 or 159CE10005-911 with AN960C10L washers to inner side of windshield.

CAUTION: DO NOT DISLODGE INNER SUPPORT BUSHINGS IF INSTALLED.

NOTE: 159CE10005-911 gasket and AN960C10L washers are used on liquid shim configuration.

(23) Ensure that all bushings are installed in windshield retaining screw holes.

(24) Install windshield using alignment pins as a guide. Caution should be used at installation so as not to damage shim on retainers.

(25) Position outer retainer with a few screws to check clearance between retainer and aluminum edge insert on windshield. If no clearance exists provide approximately 1/16 inch clearance by filing outer retainer.

NOTE: Clean retainer of all filing debris.

(26) Install Presstite 590.5 tape neatly in same area as original location. See Figure 204 for tape locations.

(27) Install pressure spring seals at top and bottom of windshield at locations noted during old wind. shield removal.

(28) Install retainers paying particular attention to proper screw length.

NOTE: Install retaining screws in center post area first.

(29) Using a screw driver adaptor and torque wrench, torque windshield retainer screws to 20 - 25 inch-pounds.

NOTE: When sufficient screws have been installed to hold windshield in place, remove alignment pins. After screws have been torqued, do not retorque screws because windshield cold flow could be induced.

(30) Apply MIL-S-8802 TYPE II (B1/4 or B1/2) sealant between structural skin and retainers and inside or retainer and aluminum edge insert on window, but DO NOT ALLOW SEALANT TO OVERLAP INSERT ON TO WINDSHIELD SEAL OR GLASS.

(31) Connect electrical leads inside (on dual sensor windshields use sensor position one terminal first. Sensor position two is to be used if sensor one fails). See Figure 201 for terminal designations.

(32) Install center post interior fairing.

(33) Ensure windshield transformer taps have been changed to correct settings.

(34) Check out windshield heating system for proper operation when sealant is dry.

(35) Install windshield wiper arm and link. Check wiper for correct tension.

TYPE	CLASS	TRADE NAME	BASE MATE-RIAL	PARTS BY WEIGHT	HARD-ENER	PARTS BY WEIGHT	CURE TEMP & CURE TIME HRS-MINIMUM	BARCOL INDENTER TYPE 935 HARDNESS UNIT	MAXIMUM WORKING TIME (MINUTES)
Type II Rigid	Class B (Tro-welled) or (Extruded)	Epon 934	934A	100	934B	33	Room Temp. 16 200°F ±10°F 1 150°F ±10°F 2	70-90	35
		Aerobond 2143*	2143A	100	2143B	40	Room Temp. 16 200°F ±10°F 1 150°F ±10°F 2	70-90	35

(1) Thoroughly stir base material. Weigh out required amount using a clean mixing plate or cup. Using a second spatula, thoroughly mix hardener. Weigh required amount into base material. Using a third spatula, thoroughly mix components by stirring and folding.

(2) Addition of 2% weight of base resin (MAXIMUM) of Cab-O-Sil thickening agent is permissible.

(3) * or equivalent liquid shim.

3. **DV Window — Removal / Installation**

NOTE: Before removing old window, check replacement for damage, blemishes, and that it is correct side. Check electrical circuits for proper resistance values. If resistance values are different from old window, change transformer taps for that window to correct setting, see Windshield Heat Power Unit - Removal/Installation, Section 30-3-0.

A. Removal

(1) Disconnect electrical leads.

(2) Remove retainer locks from aft end of slide track.

(3) Slide window and frame out of track.

B. Installation

(1) Position window and frame in track and slide into position.

(2) Replace retainer locks on slide track.

(3) Connect electrical leads.

(4) Adjust studs at base of window to minimize window rattling and vibration when either window is closed.

(5) With window closed, direct a stream of water against outside of window to check for leaks; adjust window if leaks are detected.

(6) Perform Windshield Heat - Operational Test, see Section 30-3-0.

4. **Windshield / DV Window — Resistance Check**

NOTE: This procedure applies only to aircraft having windshields P/N 159SCCE100-11/12 and DV windows P/N 159SCCE105-5/-6 manufactured in 1979 and 1980.

A. Check windshield / DV window part number and serial number.

NOTE: Part number and serial number of windshield are located along bottom, lower portion of window, approximately 2 to 3 inches from center post. Part numbers must be read from outside. Use of a mirror is recommended.

Part number and serial number of DV windows are located center of top edge, where sensor wires enter window.

Windshield P/N 159SCCE100-11/-12 and DV windows P/N 159SCCE105-5/-6 bearing manufacture serial numbers beginning with 9.H-1 through 0-H-12 are affected windows. No further action is required if part numbers/serial numbers are not same as above.

B. If affected serial numbered windows are installed, proceed as follows:

(1) Remove electrical power from aircraft.

(2) Disconnect leads and check windshield/DV window to ensure resistances within tolerance of table below.

NOTE: Resistance code is located adjacent to serial number of front window and DV window.

WINDOW	WINDOW RESISTANCE	WINDOW CODE
Front	94.5 - 103.5 OHMS 85.5 - 94.5 OHMS 76.5 - 85.5 OHMS	R1 or Z R2 or Y R3 or X
DV	62.0 - 69.0 OHMS 56.0 - 63.0 OHMS 51.0 - 57.0 OHMS	R7 or Z R2 or Y R3 or X

NOTE: If resistances are within tolerances, windows may resume normal service until next recurring inspection.

If resistance does not exceed 10% of specified range maximums, transformer tap adjustments may be made and windows may resume normal service until next recurring inspection.

If resistance exceeds 10% of specified range maximums, flight operations must be conducted observing limitations In Flight Manual applicable to window heat inoperative until affected windshield/DV window can be replaced.

(3) Reconnect leads.

5. **Cockpit Side Window — Removal / Installation**

CAUTION: IF WIRES ARE ATTACHED, ENSURE THEY ARE LABELED WHEN REMOVED FOR PROPER INSTALLATION.

A. Removal (See Figure 205)

(1) Before removal of old window, check new one to ensure it is correct side, free from any damage or blemishes and check electrical circuits for proper resistance values. Panel resistance, as measured at power lead connections on window (two red, outer larger diameter terminals) must fall within range of 62.9 to 85.1 ohms.

NOTE: If side window heat option is installed, the two smaller diameter terminals are sensor connections and are used on left hand side window only.

(2) Punch screw pattern into heavy cardboard so as not to get screw lengths mixed up.

(3) If installed, remove acoustic fence.

(4) Disconnect electrical leads from thermal switch on window, if side windows are not heated wires will be stowed.

(5) Disconnect electrical power leads.

(6) Remove retainer screws from outside.

CAUTION: IF AIR GUN IS USED, EXERCISE CAUTION AS SCREW HEADS COULD BE DAMAGED.

(7) Remove retainer plates after cutting sealant free with knife.

(8) Clean thoroughly before installation.

(9) Lift out window.

B. Installation

CAUTION: APPLICATION OF WINDOW HEATING POWER TO A SIDE WINDOW PANEL WHICH HAS A PANEL RESISTANCE VALUE BELOW 62.9 OHMS MAY CAUSE EXCESSIVE CURRENT FLOW AND DAMAGE WINDOW.

(1) Check that window panel resistance was measured and noted. (See Step A. (1) above).

(2) Clean cavity area thoroughly and place new window into position.

(3) Install tape neatly in same areas as originally located after pulling retainer plates.

CAUTION: CORRECT SIZE SCREWS ARE IMPORTANT, AS LONG SCREWS WILL BOTTOM AND PROBABLY LOOSEN BLIND NUT FASTENERS.

(4) Ensure that gaskets are in position, then install retainers with proper size screws that have been dipped in sealant.

(5) Connect electrical power leads to two outer (larger diameter) red terminals in window, if side windows are not heated wires will be stowed.

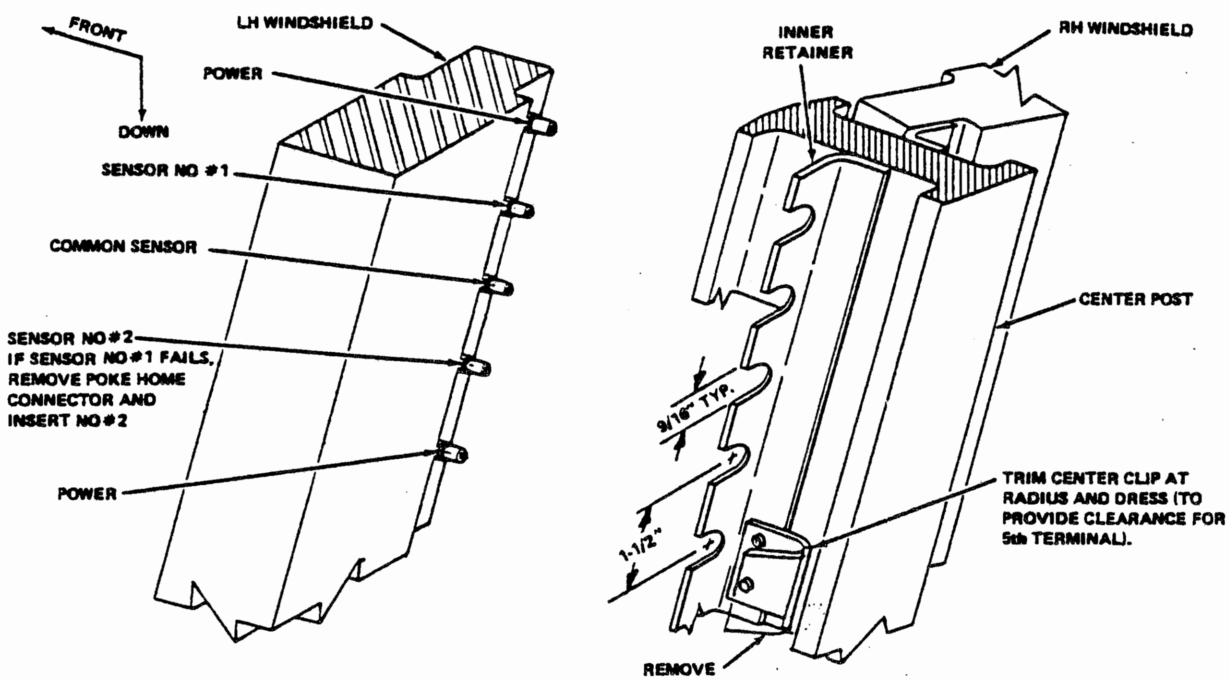
NOTE: If side window heat option is installed, the two smaller diameter terminals are sensor connections and are used on left hand side window only.

(6) Connect electrical leads to thermal switch on window.

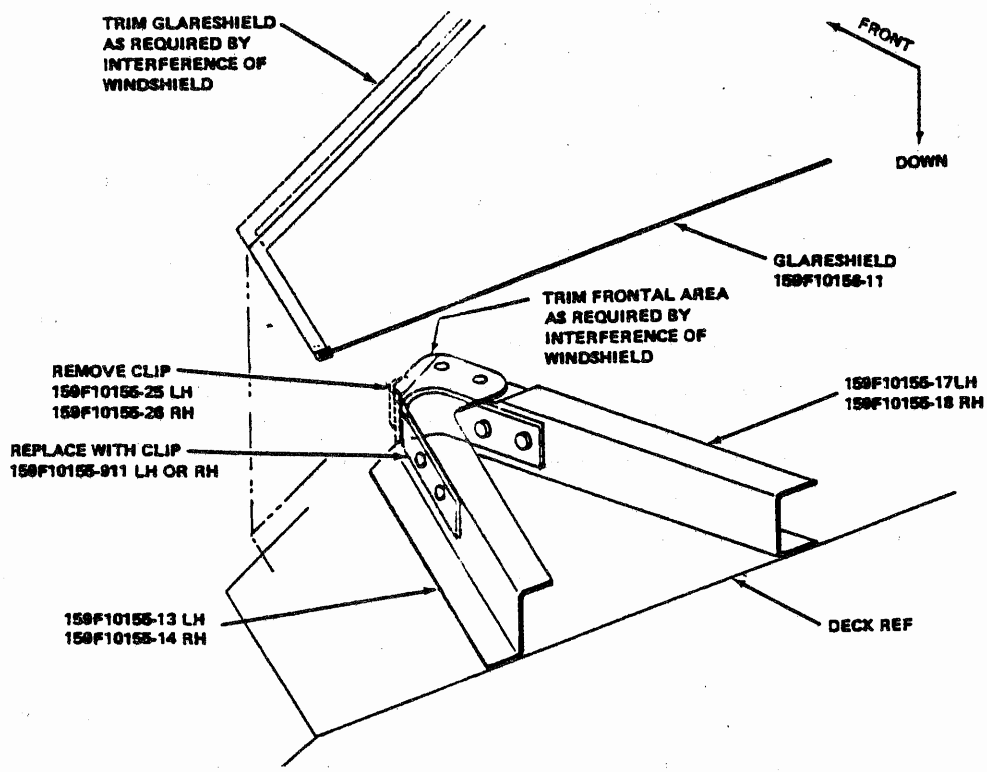
(7) Apply sealant between structural skin and retainers and allow to dry for 10 hours.

(8) Install acoustic fence, if applicable.

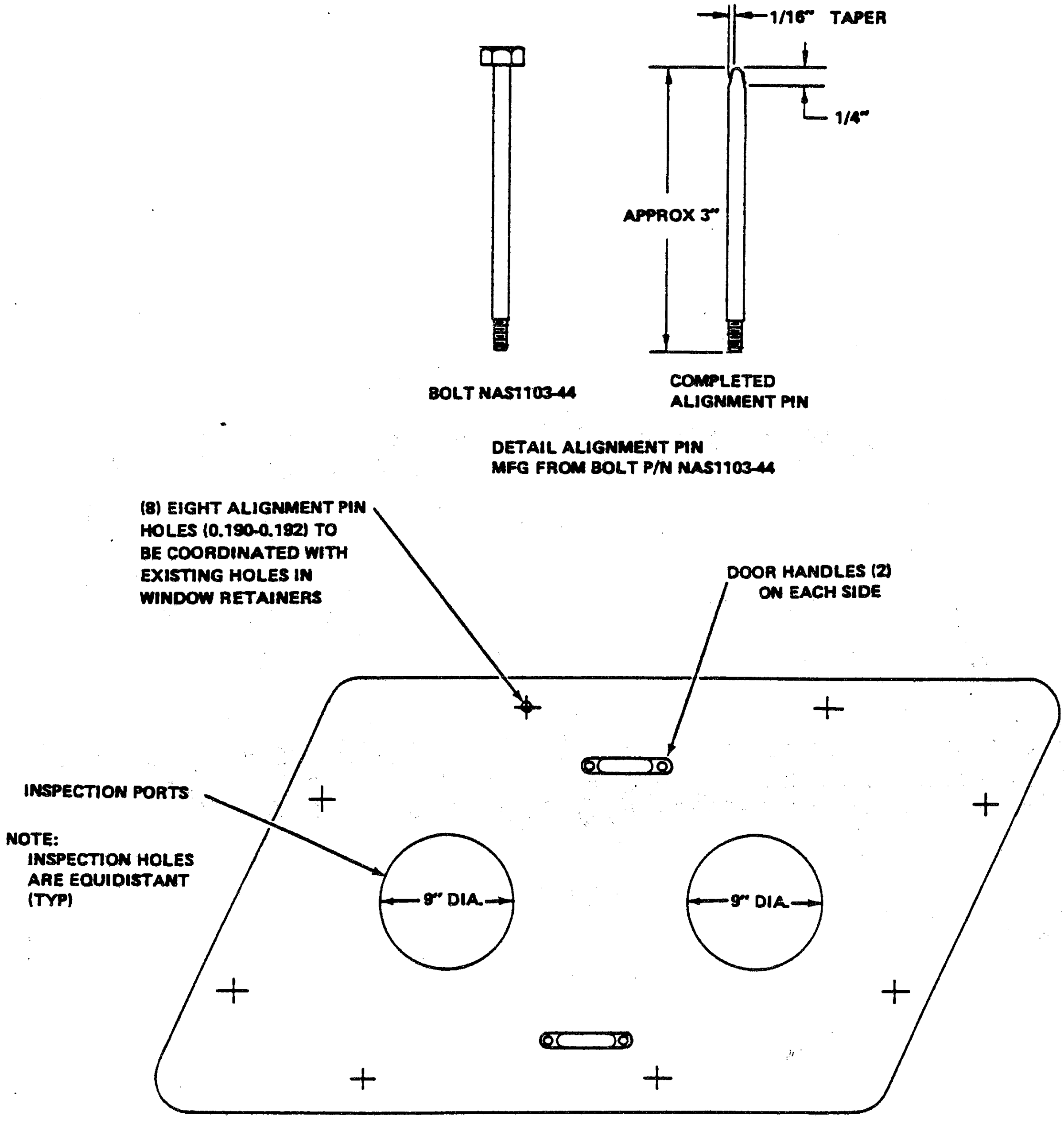
(9) Perform Windshield Heat - Operational Test, see Section 30-3-0.



Windshield Center Post and Inner Retainer
Figure 201.

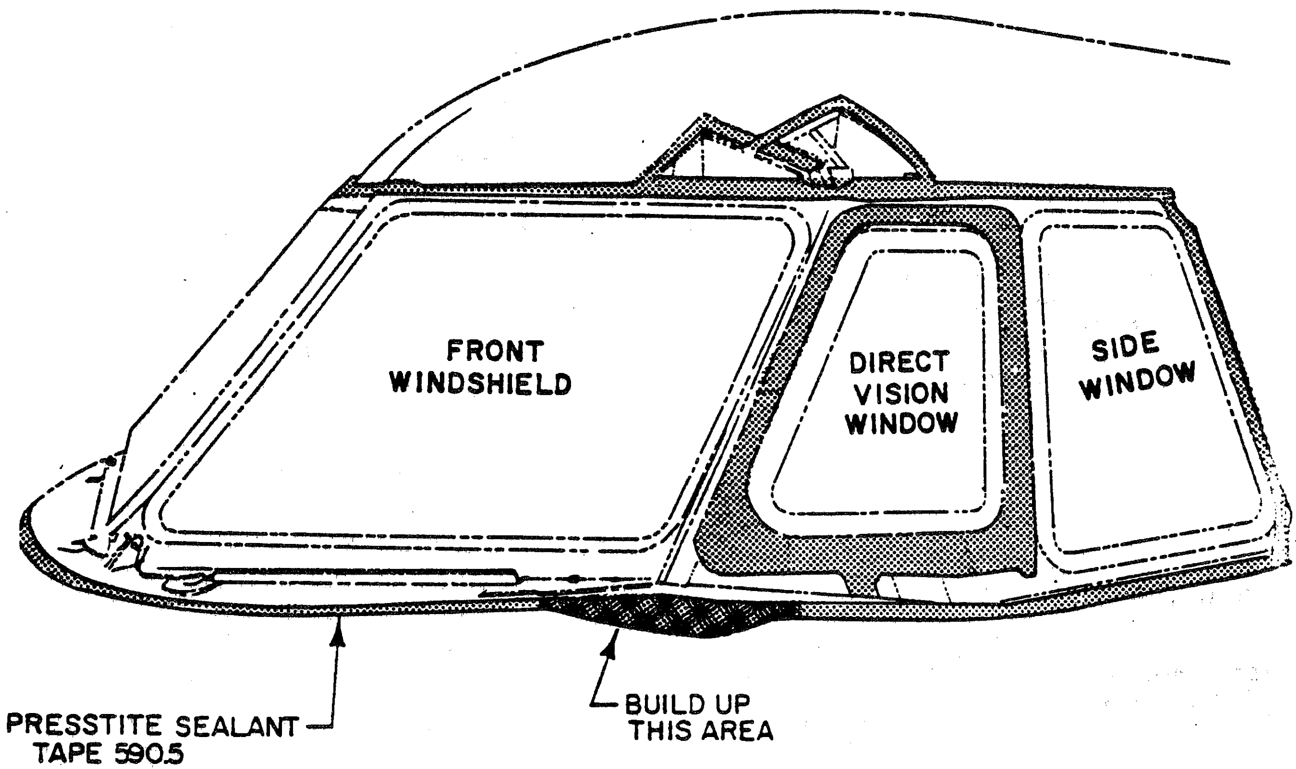


Glareshield and Supports
Figure 202.

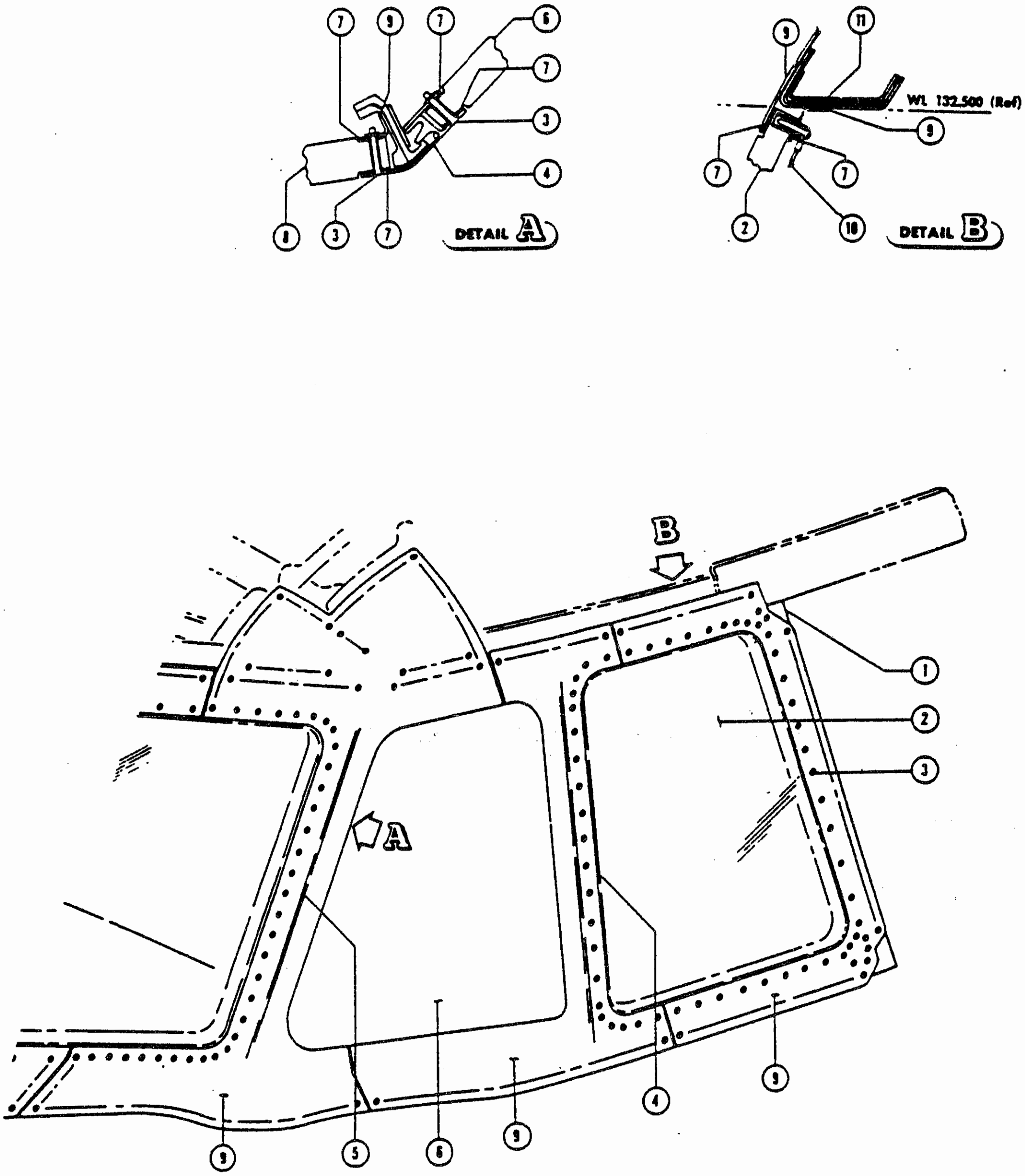


SAME CONFIGURATION AS WINDSHIELD TO BE MADE FROM ALUMINUM CAST TOOL AND JIG PLATE
THICKNESS: 1/2" FLATNESS: WITHIN 0.005 ON BOTH SIDES WITHIN 2" OF ALL EDGES.

Inspection Plate
L/H Shown, R/H Opposite
Figure 203.



Window Outer Retainer Presstite Sealant Tape Location
Figure 204.



1. Frame Assembly
2. Fixed Panel
3. Window Attachment Screws
4. Post No. 3 Assembly
5. Post No. 2 Assembly
6. Direct Vision Window
7. Gasket
8. Windshield
9. Retainer
10. Post No. 1 Assembly
11. Terminal Block & De-icing System Electrical Lead
12. Cover and Rail (Upper)

DV and Cockpit Side Window — Installation
Figure 205.

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CABIN WINDOWS — DESCRIPTION / OPERATION

1. **Cabin Windows — Emergency Escape**

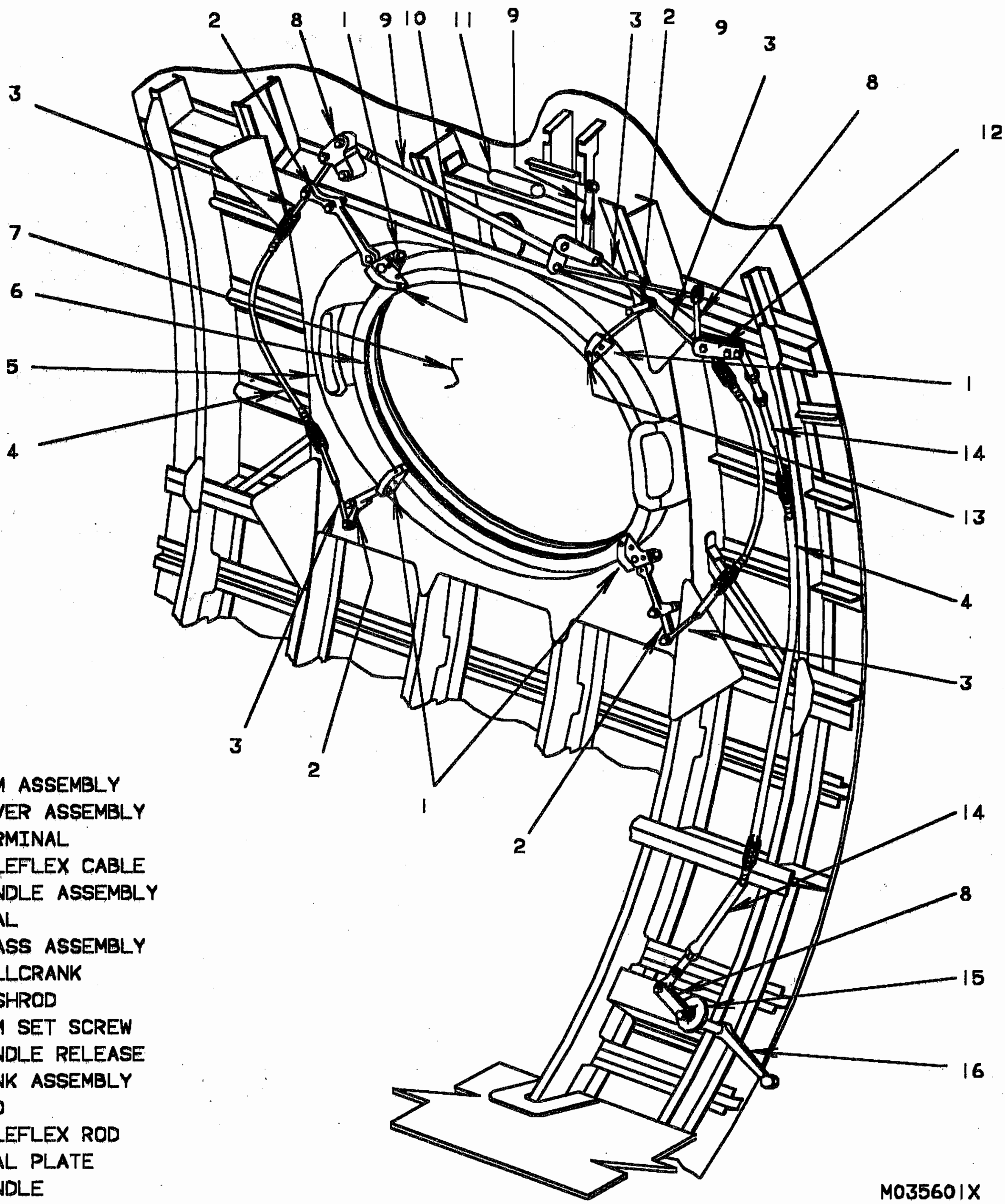
There are four windows, two on each side of the fuselage, located above the wing which may be used as emergency exits (See Figure 1). The windows may be removed from either inside or outside of the aircraft. They are constructed of multiple layers of plexiglas. The nominal thickness of each window is 1.387 inches. The window size is 20 3/8 by 27 1/4 inches. Two inside handles on each window permit pulling the window inward after the inside emergency release handle is pulled. An outside emergency release handle for each window, in a spring loaded hinged access door in the wing fillet, permits emergency release of the window from the outside. The outside emergency handle access door is appropriately placarded. The inside and outside handles are painted red, and actuate a system of teleflex cables, push rods and bellcranks.

2. **Cabin Windows — Fixed**

Six fixed type cabin windows are provided, three on each side of the fuselage. The windows are constructed of multiple layers of plexiglas. They are elliptical in shape, with a nominal thickness of 1.387 inches. The window size is 20 3/8 by 27 1/4 inches. The windows are water-tight and mounted so that they may be replaced without damaging the aircraft skin.

There are four panels separated by three desiccated air spaces that are vented together. The outside panel is the structural panel, and will withstand loads created by aircraft pressurization. The windows are defogged through the use of a desiccator system. Refer to Chapter 30-5-0 for information concerning the cabin defog system.

A fixed lavatory window is located on the aft end of the fuselage, on the right side. Its construction is identical with that of the cabin windows.



1 CAM ASSEMBLY
2 LEVER ASSEMBLY
3 TERMINAL
4 TELEFLEX CABLE
5 HANDLE ASSEMBLY
6 SEAL
7 GLASS ASSEMBLY
8 BELLCRANK
9 PUSHROD
10 CAM SET SCREW
11 HANDLE RELEASE
12 LINK ASSEMBLY
13 PAD
14 TELEFLEX ROD
15 SEAL PLATE
16 HANDLE

M035601X

Emergency Escape Window — Typical
Figure 1.

CABIN WINDOWS – MAINTENANCE PRACTICES

1. **Cabin Emergency Escape Window Gasket — Replacement**

CAUTION: DUE TO NEW DESIGN OF GULFSTREAM I CABIN WINDOW SYSTEM, IT IS IMPORTANT THAT CABIN WINDOWS INCORPORATED IN GULFSTREAMS 1 - 178, 322 AND 323 **ARE NOT INTERCHANGED** WITH WINDOWS IN 179 - 200. ALTHOUGH THESE WINDOWS ARE IDENTICAL IN SIZE, THEY ARE CONSTRUCTED TO CARRY LOADS ON DIFFERENT PANES. AIRCRAFT 1 - 178 ARE VENTED TO ATMOSPHERE AND CARRY LOAD ON INNER PANE. AIRCRAFT 179 - 200 ARE VENTED TO CABIN PRESSURE AND CARRY LOAD ON OUTER PANEL. WRONG APPLICATION OF THESE WINDOWS WOULD BE DANGEROUS. FOLLOWING ARE CABIN WINDOW PART NUMBERS AND EFFECTIVITY.

GULFSTREAMS 1 - 172, 322 AND 323	P/N 159CE10034-3
GULFSTREAMS 173 - 178	P/N 159CE10034-7
GULFSTREAMS 179 - 200	P/N 159CE10035-7

NOTE: The 159CE10035-7 is the preferred alternate for the 159CE10034-3 and -7 windows.

A. Remove window from installed position, remove cabin interior trim from window.

CAUTION: WHEN SCRAPING THE OLD WINDOW GASKET FROM THE WINDOW FORGING, TAKE CARE NOT TO SCRATCH THE FORGING SURFACE.

B. Remove old gasket from the window with a plexiglas or micarta scrapper until the surface is smooth.

C. Using a clean cloth saturated with PS-661, Federal Specification VV-G-109 or Aliphatic Naptha, Federal Specification TT-N-95, remove all loose material.

D. Cover the outboard side of the skin around the window opening with masking tape.

E. Completely cover the outboard surface of the glass with masking tape.

F. Using a hand operated pressure gun, apply a 3/8" diameter bead of sealant MIL-S-8802, Class B-1/2 or B-2 to window forging.

NOTE: A properly fitted window will have a gasket of approximately 1/16" thick.

CAUTION: APPLY THE PARTING AGENT TO THE SEAL AT LEAST 1/2 INCH WIDER THAN THE MATING SURFACE. PARTING AGENT MUST BE APPLIED CAREFULLY, TO ENSURE COMPLETE COVERAGE. IF THESE PRECAUTIONS ARE NOT STRICTLY OBSERVED, IT MAY BE DIFFICULT TO REMOVE WINDOW IN THE EVENT OF AN EMERGENCY.

G. Apply two coats of polyvinyl alcohol solution (which acts as a parting agent) to seal mating surface of glass assembly. Allow the solution to air dry. If polyvinyl solution is not available, use vaseline as a parting agent.

NOTE: Maximum step permissible between window and adjacent structure is $\pm$ 1/16 inch.

H. Set window assembly in place while sealant is still fluid, and lock emergency release, adjust setscrews to hold window firmly in place while maintaining permissible step.

I. Sealant will be extruded outside of the window onto the masking tape installed in Step (4) above. Smooth extruded sealant flush with masking tape.

J. Remove excess sealant. Before sealant cures, clean windows with material referred to in Step (3) above.

K. Remove masking tape from skin around window.

L. Remove masking tape from window assembly when sealant is no longer tacky.

NOTE: The drying (tack free) time for sealant MIL-S-8802 Class B-1/2 or B-2 under controlled temperature and relative humidity conditions (77 $\pm$ 2°F, and 50 $\pm$ 5%) is 10 to 24 hours.

Prior to removing newly installed windows, it is recommended that they be indexed by placing four pieces of masking tape,equally spaced, across window frame and window opening frame.

Apply a mark on tape and carefully cut tape between window and frame. When installing window align these marks.

M. Accomplish the following operations at each escape window after sealant has cured.

(1) Release retention mechanism by pulling emergency release handle.

(2) Grasp both handles and gently pull the window away from the frame without disturbing the gasket.

(3) Trim any excess sealant that could hinder window removal.

(4) Install cabin interior trim to window assembly.

(5) Fit window into position, align index marks and latch by returning emergency release handle to locked position.

N. Check each window for leakage by directing a stream of water from a hose on the outside of the window. Inspect windows on the inside for evidence of leakage. If leakage is found, reseal and water check again.

O. Connect vent hose.

P. Install any removed upholstery.

2. Emergency Escape Window — Rigging

A. Retract four lever assemblies and install window.

NOTE: Emergency escape windows are interchangeable and need not be returned to the same opening from which they were removed. However, resealing will be required if this is done.

B. Push T-handle above window until detented position is reached.

C. Attempt to rock each lever assembly.

D. Inspect setscrew on all lever assemblies (four places each window) for presence of Loctite "Screw Lock".

E. If there is clearance between a setscrew and the rubber pad or if no Loctite "Screw Lock" is evident, proceed as follows:

(1) Remove all four setscrews from that window. Clean and degrease setscrew threads with Locquic Primer T #47-56.

(2) Apply Loctite "Screw Lock" to all threads of setscrews.

(3) Adjust setscrews so that window is firmly in place and such that the mechanism may be operated to release the window. Pull T-handle to check mechanism.

NOTE: Difficulty in pulling T-handle indicates that the setscrews are too tight. Back them off 1/2 turn at a time until window is released by a normal pull of the T-handle.

(4) Return T-handle to locked position.

3. Surface and Edge Repair

A. General

Surface defects such as nicks and scratches are structurally acceptable without rework if it cannot be detected by a fingernail (defined as 0.003 inches or less). Windows with surface defects which are structurally acceptable may be objectionable from a visibility standpoint. The quality of visibility that is acceptable will be determined by the individual owner.

Surface defects which can be detected with a fingernail (defined as greater than 0.003 inches in depth) are not structurally acceptable without rework. To establish the rework status of the panel, it must be removed from its retainer to determine its unworked thickness. Before returning the reworked panel to service, it must meet the following minimums;

Window Panel	Thickness Minimum	Thickness Maximum	Minimum After Rework
Outer	0.330	0.370	0.320
Inner	0.330	0.370	0.320

The preferred method of using all abrasive materials is wet with water. This method provides a cooling effect to the surface being reworked and generally produces a higher quality finish than dry abrasives. The use of a power rotating sander is acceptable, however, it is recommended that if one is used, it be pneumatically powered and fitted with a water dispensing device.

B. Shallow Scratches (0.003 to 0.010 inches deep)

After the surface has been cleaned, polish the area with light pressure in a rotary motion. Start with 400 to 600 grit paper and finish with finer grades 2400, 4000 and 8000 of micro mesh. The selection of grit for initial sanding will vary with a depth of the scratches and experience of the personnel.

C. Deep Scratches (0.010 to 0.20 inches deep)

Scratches of this depth should be removed by reducing the thickness of the entire panel. This will eliminate the possibility of distortion. Deep scratches may be removed using abrasives and procedures called out in Step B. above, provided the effect on visual quality is acceptable. All reworked areas must not exceed the minimums listed in Step A.

D. Edge Defects

A small amount of edge chips or crazing is not uncommon in acrylic window material. It is permissible to rework the minor defects by chamfering with a fine hand file and hand sanding until a surface finish of 125 RMS or better is obtained, except for the critical radius C_1. Polish by hand for a smooth transition with adjacent material. See Figure 201 for allowable limits for rework. Any rework requiring material removed beyond the limits set in this procedure should be presented to Gulfstream Aerospace Technical Operations Department.

E. Polishing

When using a rotating buffer fitted with a felt pad, apply a generous amount of polishing agent (Novus #1 or #2 or equivalent per Federal Standard P-P560, Type 1) to the pad and window panel. Work the panel using a rotary motion. Do not allow the wheel to rotate in excess of 1300 RPM using a five inch diameter or smaller wheel, nor use excessive pressure as this causes heat build-up resulting in burns in the acrylic window. Final polishing may be accomplished by hand. After polishing, inspect each panel to ensure minimum thickness per Step A. above.

4. **Cabin Windows — Inspection**

Inspect entire visible surface of the outer window panel for any evidence of in-plane crazing or cracking including the panel's rebate radius.

The visual appearance of in-plane crazing looks like water pressed between two panes of glass, interspersed with "Spider Web Like" craze lines. In-place cracking appears the same except no craze lines will be visible, only a solid line defining the inner edge of the crack. (See Figure 202)

If in-plane crazing or cracking is visible in the outer panel, the panel should be replaced before further flight.

5. **Emergency Window Release — Operational Test**

NOTE: Four windows, two on each side of fuselage located above wing may be used as emergency exits. Window release operation should be checked from both outside and inside for proper operation.

CAUTION: TO PREVENT DAMAGE TO THE WINDOW AND EQUIPMENT, POSITION A MAN INSIDE THE CABIN TO ASSIST IN REMOVING EMERGENCY WINDOWS.

A. Procedure (Internal)

(1) Prior to window removal place four pieces of masking tape equally spaced across window frame and fuselage frame.

(2) Using a felt tip pen, or equivalent, and a straight edge, apply a mark on the tape to extend over the window frame and fuselage frame.

(3) Carefully cut tape at fuselage frame so as not to damage the window seal.

(4) Pull (red) emergency release handle.

(5) Pull window inward by two handles attached to each window.

(6) Replace window in opening, align marks on tape and lock release handle into detent.

(7) Remove tape strips.

B. Procedure (External)

(1) Prior to window removal place four pieces of masking tape equally spaced across window frame and fuselage frame.

(2) Using a felt tip pen, or equivalent, and a straight edge, apply a mark on the tape to extend over the window framė and fuselage frame.

(3) Carefully cut tape at fuselage frame so as not to damage the window seal.

(4) Open spring-loaded hinged access door in wing fillet.

(5) Lift (red) emergency release handle.

(6) Push window inward.

(7) Replace window in opening, align marks on tape and lock inside release handle into detent.

(8) Lubricate mechanism with dry film lubricant per MIL-L-60326 Type I or approved equivalent.

(9) Close external hinged access door.

(10) Remove tape strips.

(11) Remove any upholstery that may get wet during following Step.

(12) Check each window for leakage by directing a stream of water from a hose on the outside of the window. Inspect windows on the inside for evidence of leakage. If leakage is found, reseal and water check again.

(13) Connect vent hose.

(14) Install any removed upholstery.

NOTE: If release does not function properly, re-rig per Emergency Escape Window — Rigging, this Section.

6. Fixed Cabin Window — Removal / Installation

CAUTION: DUE TO NEW DESIGN OF GULFSTREAM I CABIN WINDOW SYSTEM, IT IS IMPORTANT THAT CABIN WINDOWS IN AIRCRAFT 1 - 178, 322 AND 323 **ARE NOT INTERCHANGED** WITH WINDOWS IN AIRCRAFT 179 - 200. ALTHOUGH THESE WINDOWS ARE IDENTICAL IN SIZE, THEY ARE CONSTRUCTED TO CARRY LOADS ON DIFFERENT PANES. AIRCRAFT 1 - 178, 322 AND 323 ARE VENTED TO ATMOSPHERE AND CARRY LOAD ON INNER PANE. AIRCRAFT 179 - 200 ARE VENTED TO CABIN PRESSURE AND CARRY LOAD ON OUTER PANEL. WRONG APPLICATION OF THESE WINDOWS WOULD BE DANGEROUS. FOLLOWING ARE CABIN WINDOW PART NUMBERS AND EFFECTIVITY.

AIRCRAFT 1 - 172, 322 AND 323	P/N 159CE10034-1
AIRCRAFT 173 - 178	P/N 159CE10034-5
AIRCRAFT 179 - 200	P/N 159CE10035-5

NOTE: The 159CE10035-5 is the preferred alternate for the 159CE10034-1 and -5 windows.

A. Removal

(1) Remove all necessary upholstery and interior trim to gain access to window.

(2) Disconnect vent hose from window vent tube.

(3) Loosen set screws holding window in place.

(4) Remove four retaining clips and remove window.

B. Installation

CAUTION: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAME.

WHEN SCRAPING THE OLD WINDOW GASKET FROM THE WINDOW FORGING, TAKE CARE NOT TO SCRATCH THE FORGING SURFACE.

(1) Remove old gasket from fuselage opening with a plexiglas or micarta scrapper.

(2) Using clean cloth saturated with Aliphatic Naptha, remove all loose material.

(3) Tape fuselage on outside of window opening.

(4) Cover outboard surface of window with masking tape.

(5) Using a hand operated pressure gun, apply a 3/8 diameter bead of sealant on metal window frame.

NOTE: A properly fitted window will have a gasket approximately 1/16 inch thick.

(6) Apply parting agent to fuselage window opening.

NOTE: Maximum step permissible between window and adjacent structure is $\pm$ 1/16 inch.

(7) Install window and tighten lock tabs evenly, while maintaining permissible step.

(8) Remove masking tape from skin around window.

(9) Remove masking tape from window assembly when sealant is no longer tacky.

NOTE: The drying time for sealant under controlled temperature and relative humidity conditions (77 $\pm$ 2°F and 50 $\pm$ 5%) is 10 to 24 hours.

(10) Remove excess sealant. Before sealant cures, clean windows with Aliphatic Naptha. Remove all loose material.

(11) Remove any upholstery that may get wet during following Step.

(12) Check each window for leakage by directing a stream of water from a hose on the outside of the window. Inspect windows on the inside for evidence of leakage. If leakage is found, reseal and water check again.

(13) Connect vent hose.

(14) Fit interior trim to window assembly and install any removed upholstery.

7. Cabin Windows — Cleaning

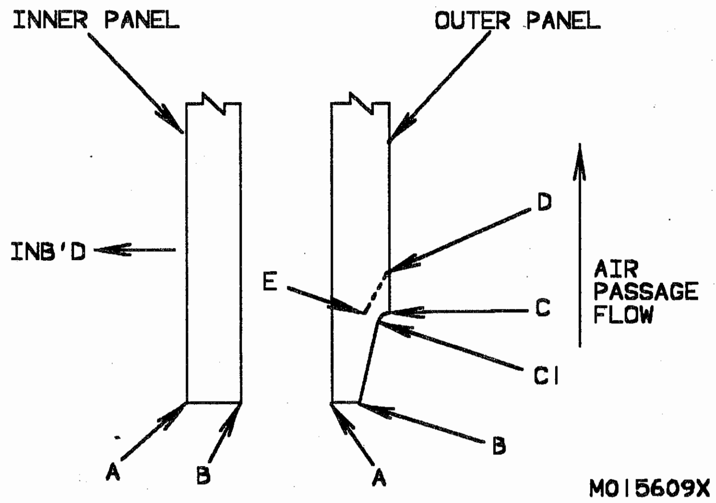
A. Remove all excessive amounts of dirt and other substances from the plastic surface with clean water.

B. Use either a mild soap and water, a 50/50 solution of isopropanol and water, or aliphatic naptha type 2. Do not use any abrasive materials, strong acids or bases, methanol, or methyl ethyl ketone. When wiping the surface, use a soft cloth or sponge and use a straight rubbing motion.

C. For stubborn areas, clean panel with Permatex 403D or mineral spirits. Follow with washing as per Step above.

D. Rinse thoroughly and dry.

E. After cleaning, plastic surfaces may be polished by applying a thin coat of hard polishing wax. Rub lightly with a soft cloth, using a circular motion.



A = 0.090" X 45 CHAMFER MAX. WITH MIN. 0.25" RUN OUT AT EACH END

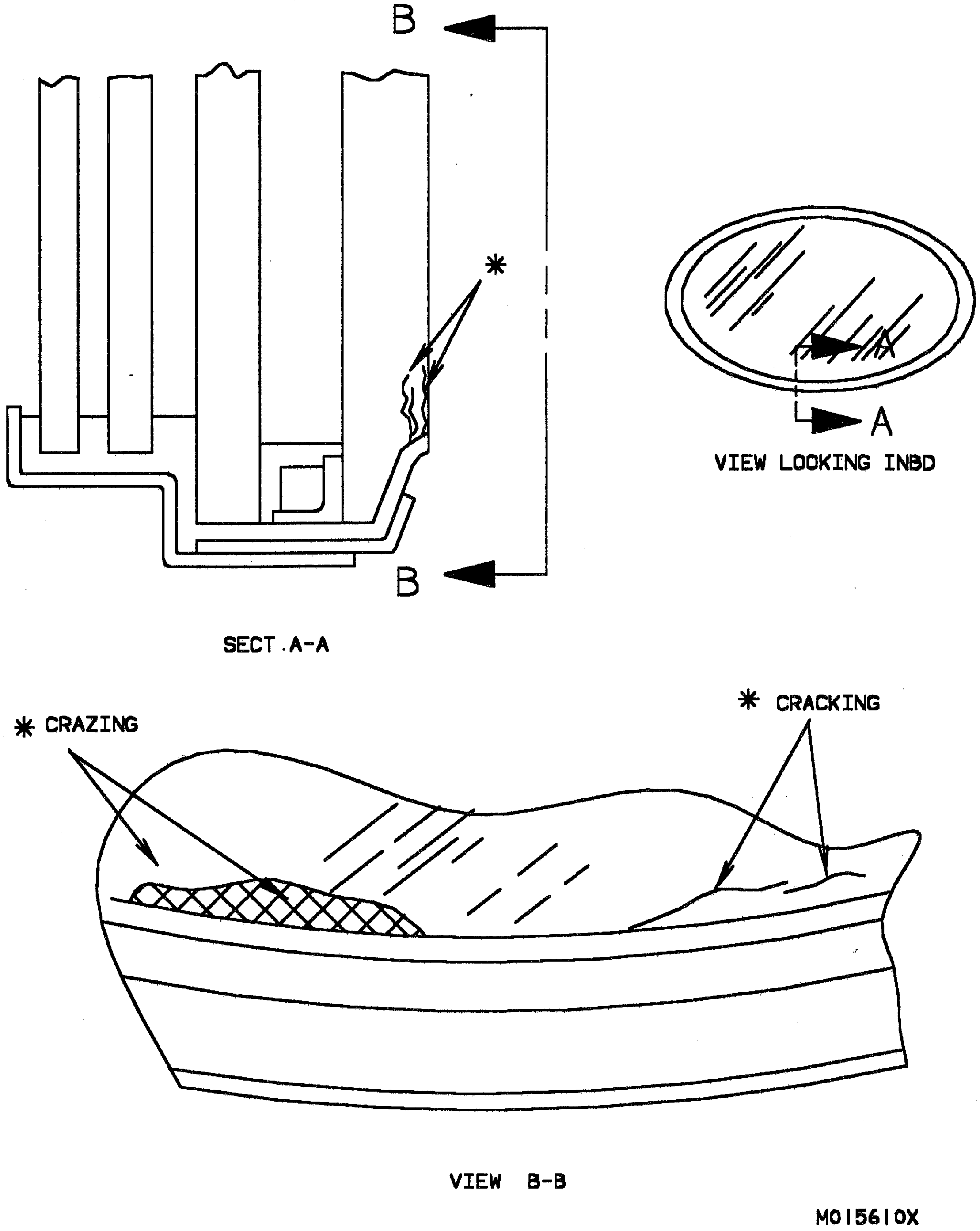
B = 0.063 X 45 CHAMFER MAX. WITH MIN. 0.25"RUN OUT AT EACH END

BED = IS REWORKABLE ON AN INDIVIDUAL BASIS. CONSULT GULFSTREAM AEROSPACE

C = 0.063 X 45 CHAMFER MAX. WITH MIN. 1.0 IN RUN OUT

C_1 = CRITICAL RADIUS BLEND OUT ANY MARKS UNTIL IT CAN NO LONGER BE FELT WITH A FINGERNAIL. MAINTAIN AS LARGE A RADIUS AS POSSIBLE.

Allowable Limits for Rework
Figure 201.



Cabin Window Inspection
Figure 202.

LIST OF EFFECTIVE PAGES

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DOORS

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*52	i	May 8/92
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*52-1-0	1	May 8/92
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*52-5-0	1	May 8/92
*52-5-0	2	May 8/92
*52-5-0	3	May 8/92
*52-5-0	4	May 8/92
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*52-5-0	202	May 8/92
*52-6-0	1	May 8/92
*52-6-0	2	May 8/92
*52-7-0	1	May 8/92
*52-7-0	2	May 8/92

* Pages revised by Revision 46
** Pages added by Revision 46
*** Pages deleted by Revision 46

LIST OF EFFECTIVE PAGES

CHAPTER 57

WINGS

SECTION	PAGE	DATE
*57	i	September 11/92
*57	ii	September 11/92
*57	1	September 11/92
*57	2	September 11/92
*57-0	1	September 11/92
57-0	2	February 28/92
*57-0	201	September 11/92
*57-0	202	September 11/92
*57-0	203	September 11/92
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*57-0	205	September 11/92
*57-0	206	September 11/92
*57-0	207	September 11/92
*57-0	208	September 11/92
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***57-0	218	February 28/92

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CHAPTER 57

WINGS

WINGS — DESCRIPTION / OPERATION

1. Description

The wing is a full cantilever structure consisting of a stub center section which is integral with the fuselage, inner and outer panels, leading edge structures, trailing edge overhangs and tip caps. The basic structure of the wing consists of front and rear beams, ribs and upper and lower skins. The leading edge structures are equipped with flush type de-icer boots. Main splice joints are located at Wing Stations 40.95 and 310.93 for the box beam structure and at Wing Station 455.34 for the wing tip caps. The flap track fittings, attached to the rear beam and overhand structure, are located at Wing Stations 66.50, 133.27, 205.25 and 274.75. Aileron hinge fittings are at Wing Stations 314.312, 364.37, 402.62 and 440.87. See to Chapter 6 for station locations.

Center Section.

The center section box beam is within the cabin pressure vessel. Access plates installed in the lower cover are sealed to maintain fuselage pressurization. The front and rear beams of the center section intersect the Fuselage Station 288 1/2 and 353, respectively. Fittings riveted to the lower end of the frames at these stations are used to bolt the center section to the fuselage.

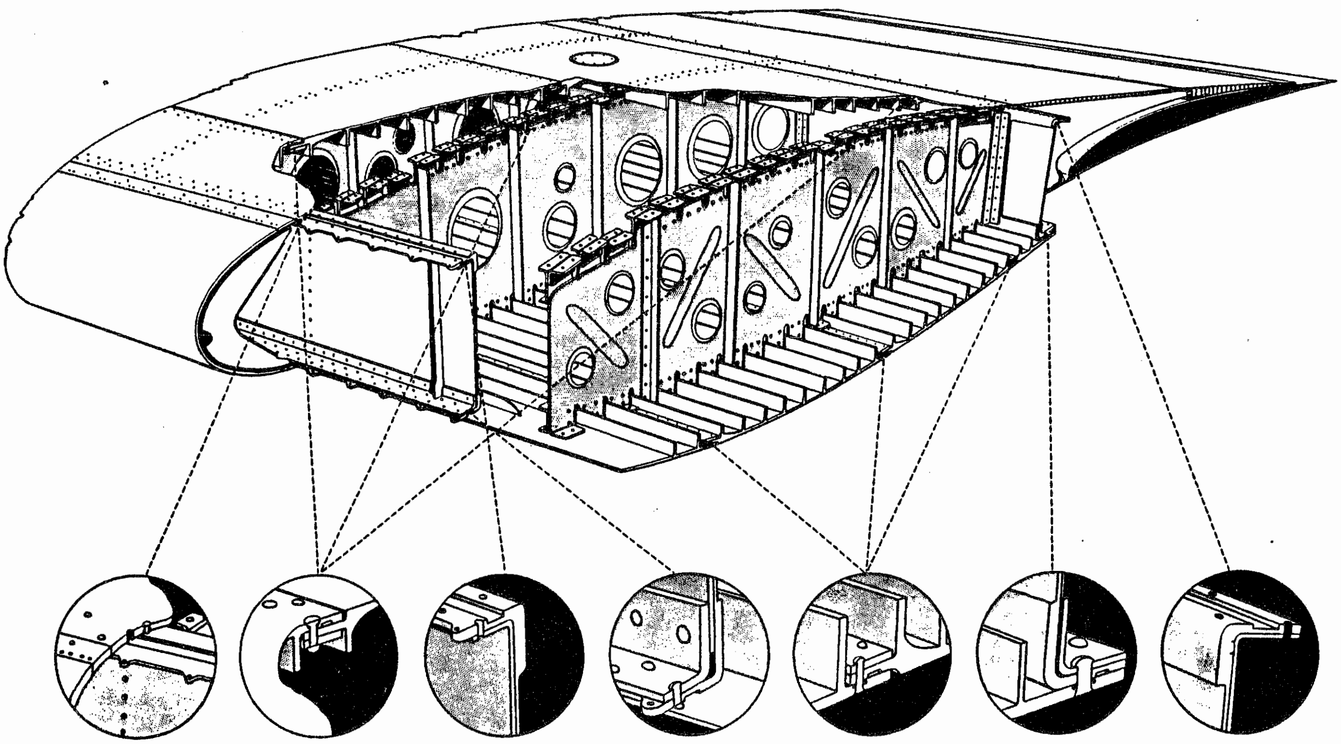
Inner Panel (See Figure 1)

The inner panel extends from Wing Station 40 to 310.93 and consists essentially of ribs, beams and upper and lower covers. The inner panels form the integral fuel tanks and utilize machined skin covers on the upper and lower surfaces. Each cover is composed of four integrally stiffened planks. The interior of the tank is coated with EC-1635 type Buna-N-Rubber for corrosion prevention. Between the wing beams and the plank skin, a single continuous channel is chemically milled and provides a gap which is filled by non-curing Thiocol rubber sealant. Rivets in tank area are sealed with O-rings. Adjacent to channel, there are injection holes placed every six inches to permit the injection of Thiocol sealant under pressure. Stressed access plates are installed in the upper cover of each tank for inspection of internal structure and for removal and installation of fuel system components. The main landing gear and nacelle structure are supported by fittings attached to the inner panel structure by leak proof fasteners. See Chapter 28 for replacement of sealing fasteners. The leading edges, equipped with flush mounted de-icer boots, are attached to the inner panel by screws and platenuts.

Outer panel.

The outer panel extends from Wing Station 310.93 to 455.34 and consists of ribs, beams, and upper and lower covers. The outer panels carry the water/methanol tanks between Wing Station 313.250 and 348.5. On aircraft having ASC 125 the water/methanol tanks are removed from the outer panels and relocated as integral tanks in the wing fillet. See Chapter 28-0 for details on aircraft having ASC 125. Seven bladder type fuel cells extend from Wing Station 313.250 to the outer tip of each wing in place of the water/methanol tanks. Stressed access plates are installed in the upper and lower covers of each tank to permit inspection of the internal structure, water/methanol tanks (or in the case of aircraft having ASC 125 - fuel tanks), and control surface linkage. The leading edge, equipped with a flush-mounted de-icer boot, is attached to the outer panel structure with screws and platenuts. At Wing Station 455.343, a tip is installed which contains a navigation light. Access to navigation light is through a removable plexiglas cover. (For location of tanks, See Chapter 28.)

Integral fuel tanks design permits any leak to be repaired from the outside with minimum effort



NON - CURING THIOCOL RUBBER INJECTED UNDER PRESSURE INTO CHANNEL SEAL GROOVE

Typical Wing Section Showing Integral Fuel Tank
Figure 1.

WINGS — MAINTENANCE PRACTICES

1. **Lower Center Section — Inspection**

   A. Open access panels shaded in Figure 201.

   B. Inspect lower wing section for corrosion and fluid accumulation.

   C. Inspect removed fairings for corrosion or any foreign substance adhering to fairing surfaces.

   D. Inspect area for presents of foreign objects, security of all attachments.

   E. Close access panels.

2. **Wing (Under Heat Shield) — Inspection**

   A. Remove tailpipe, tailpipe heat shield, see Tailpipe Forward and Aft — Removal/Installation, see Chapter 78.

   B. Remove access cover 25-UPR-4 or 26-UPR-4.

   C. Inspect wing structure inside wing for evidence of overheating, structural damage, corrosion and cleanliness.

   D. Inspect for presence of foreign objects then close access cover.

   E. Install tailpipe, tailpipe heat shield, see Tailpipe Forward and Aft — Removal/Installation, see Chapter 78.

3. **Inboard Wing Beam / Wing Panel — Inspection**

   A. Remove leading edges shaded in Figure 202

   B. Inspect the inboard section of the forward beams for:

      (1) Condition, evidence of corrosion, structural failure and evidence of fuel leaks.

      (2) Condition, security, and evidence of leakage of hydraulic, pneumatic and water/methanol installations, including components, tubing, hoses and connections.

      (3) Condition and security of flight and engine control installations including cables, pulleys, bellcranks, rods and fairleads.

      (4) Condition, security and evidence of overheating of electrical installations including wiring, conduits, plugs and connectors.

      (5) Inspect all cables, paying particular attention to aileron primary control cables at wing stub section.

      (6) Condition, security and evidence of corrosion of forward wing attachment fittings.

      (7) Condition, security and evidence of structural failure and corrosion of the leading edges including de-icer boots lines and connections.

      (8) Cracks in front beam caps, upper and lower, in nacelle area. Pay particular attention to areas adjacent to machined post that supports engine mounting A-frames on front beam caps. If cracks are detected, contact Gulfstream Aerospace Technical Operations for assistance.

      (9) Inspect for presence of foreign objects then install leading edges and outer wing access panels.

      (10) Perform De-icer Boot — Operational Test, see Chapter 30.

4. **Wing / Flap Track — Removal / Installation**

NOTE: Replacement of these fittings require Gulfstream Aerospace engineering supervision, therefore this procedure is for recording removal/installation only.

A. Removal

(1) Remove flap, see Chapter 27.

(2) Remove flap track.

B. Installation

(1) Install flap track.

(2) Install flap, see Chapter 27.

5. **Outboard Wing Beam / Wing Panel — Inspection**

A. Remove outer wing access panels shaded in Figure 203 and Figure 204, and inspect interior of outer wing panels for:

(1) Condition, security and evidence of corrosion of structural members and adjacent skin.

(2) Condition and security of flight control installations including cables, bellcranks, control rods and fairleads.

(3) Condition, security and evidence of overheating of electrical installations including components, wiring, conduits, plugs and connectors.

(4) Condition and security of water/methanol and fuel installations including components, tubing, hoses, fittings and connectors. Aircraft having ASC 125.

(5) Presence of foreign objects, loose hardware and accumulation of dirt and/or fluid.

(6) Inspect water/methanol pumps and transmitter for security and evidence of leakage. This inspection not required on Aircraft having ASC 125).

(7) Inspect for cracks in rear beam lower cap angle. If cracks are detected, contact Gulfstream Aerospace Technical Operations for assistance.

(8) Inspect fuel line (P/N 159P10091-7) for chafing. Pay particular attention to where it passes from lower nacelle to rear beam through elongated slot in the wing lower gap plate (P/N 159W10070-11/-12). If line is chafing. If cracks are detected, contact Gulfstream Aerospace Technical Operations for assistance.

(9) Inspect for presence of foreign objects then install leading edges and outer wing access panels.

(10) Perform De-icer Boot — Operational Test, see Chapter 30.

6. **Lower Wing Plank Corrosion — Inspection**

A. Remove nacelle fairings (P/N 1159WP10011-11/-12 and 159WP10012-11/-12).

B. Inspect wing structure for corrosion and condition of surface protective treatment.

NOTE: Corrosion with a cordwise dimension of not more than 2 inches, a spanwise dimension of not more than 3 inches and an average depth of not more than 0.010 inch may be blended out and the surface retreated with finish Number 76 (Alodine) and 2012 (Epoxy Primer).

C. After cleaning off corrosion and smoothing out rough surface (if required), inspect for signs of delamination. If delamination is suspected, an ultrasonic inspection should be performed.

NOTE: Corrosion that is more extensive than that defined above must be evaluated by a structural engineer. Contact Gulfstream Aerospace Technical Operations for assistance.

D. Inspect area for foreign objects, security of all attachments.

E. Install nacelle fairings.

NOTE: Addition of a 1/4 inch diameter notch in the rear end of each fairing will drain and ventilate the area and help prevent further corrosion.

7. **Wing / Fuselage Attach Fittings — Inspection**

A. Visual inspection of wing to fuselage attachment fittings (WM fittings).

(1) Gain access to WM fittings via access holes in lower aft intermediate plank in stub section and in the dihedral break rib at Fuselage Stations 288.5 and 353.0. (See Figure 205)

(2) Using a bright light and mirror, inspect eight WM fittings for cracks and loose or working fasteners. If cracks are suspected, use dye penetrant inspection for verification.

NOTE: During the following visual inspections, when sealant obscures or hinders performance of inspection, it must be removed in sufficient quantity so inspection can be performed. Restoration of removed sealant need only be in form of fillets around fasteners and along frame edge at skin surface. All areas of sealant removed must be corrosion protected. Reseal with sealant manufactured per MIL-S-8802 or MIL-S-7502. Apply sealant in a manner which will eliminate sealant removal on subsequent inspections.

B. Visual inspection of fuselage to wing attachment fittings (BM fittings) in cabin area. (See Figure 205)

(1) Remove interior furnishings and air conditioning ducts as require

(2) Remove floorboards as required to expose area bounded by Fuselage Stations 288.5 thru 354.0, below Stringer 15 and upper surface of wing skin.

(3) Visually inspect attachment beams, inboard and outboard flanges and web radii for cracks, working or missing fasteners from Stringer 15 down to attach fittings.

(4) Inspect frame end fittings for cracks in the radii and any indication of loose or working attachment bolts at Fuselage Stations 288.5 and 353.0.

(5) At Fuselage Stations 297 thru 344, inspect for cracks in frame radii and around attachment fasteners in frame and tee clips.

C. Visual inspection of IBM fittings outside of cabin area. (See Figure 205)

(1) Remove fairing 107-LH FIL-1, 107-RH FIL-1, 107-LH FIL-7 and 107-RH FIL-6.

(2) Inspect fuselage and wing attachment fittings, links and fuselage skin (four locations) for cracks and evidence of loose or working fasteners.

D. Visual inspection of attachment IBM fittings inside center wing section at forward and aft wing beams. (See Figure 205)

NOTE: Visual inspection of the eight IBM fittings from inside center wing section can be performed, with difficulty, through existing access holes in lower aft intermediate wing plank, in conjunction with access holes in dihedral break ribs located at outboard edge of wing center section.

(1) Remove access covers 27-CWIB-1 thru 27-CWIB-4 and 28-CWIB-1 thru 28-CWIB-4.

(2) Visually inspect eight IBM fittings located within wing center section for cracks or evidence of working fasteners. Particular attention is to be focused on side flanges and radii of fittings (areas 90° to beam face).

(3) Inspect area for presents of foreign objects, security of all attachments and install access covers.

8. **Wing Interior — Inspection**

A. Defuel aircraft below level of access covers.

B. Open access panels shaded in Figure 206.

C. Condition, security and evidence of corrosion of structural members and adjacent skin.

D. Condition, security and freedom of movement of flight control installations including cables, bellcranks, control rods and fairleads.

E. Condition, security and evidence of overheating of electrical installations.

F. Condition, security and evidence of leakage of fuel installations including components, tubing, hoses, fittings and connectors.

G. Presence of foreign objects, loose hardware and accumulation of debris.

H. Condition of O-ring under water/methanol filler cap.

I. Water/methanol noise filter and low pressure switch (located in wing gap band at wing station 307 to 313) for corrosion, connections for security and condition and electrical plug for safety wire (not applicable on aircraft having ASC 125.

J. Wing fuel tank interior structure, planks, fuel tanks, plumbing, probes and wiring for corrosion, cracks, contamination, deterioration, security of components and attaching hardware and clamps. Inspect for evidence of contamination. Look carefully for a brown to reddish brown material and if any is found, inspect remainder of tank for same condition. Particular attention should be given to area of fuel cap.

NOTE: If flakes of a large size or a large quantity of small flakes are found, tank should be cleaned as directed in Chapter 28.

K. Inspect condition of O-ring under fuel tank filler caps.

L. Check fuel tank vent valve for freedom of operation and relief hole for obstructions. (Vent valve hole was added to Aircraft 1-93, and 114, having ASC 154, Aircraft 94-200, 322 and 323.

M. Fuel pump mounting and probe wiring for condition and security.

N. Fuel ejector for general condition.

O. Outboard wing fuel cells (Aircraft having ASC 125 for condition, security of fuel connections, attaching hardware, O-ring under filler cap, and evidence of leakage from Wing Station 313 to 455.

P. Inspect for presence of foreign objects then close access panels.

9. Fuel Tank Contamination — Inspection

A. Defuel aircraft below level of inboard access covers.

B. Open upper inboard fuel tank access covers 25-UPR-1 AND 26-UPR-1.

C. Inspect fuel tank through access covers for foreign objects and/or fluid accumulations other than fuel.

D. Inspect for evidence of contamination. Look carefully for flakes of a brown to reddish brown material and if any is found, inspect remainder of tank for same condition. Particular attention should be given to area of fuel cap.

NOTE: If flakes of a large size or a large quantity of small flakes are found, the tank should be cleaned as directed in Chapter 28.

After cleaning the tank of all evidence of liner flakes, use every possible precaution to ensure adequate mixing of the fuel additives being used and inspect carefully at the next 150 hour/6 months inspection. If necessary, repeat the cleaning operation.

If a sizeable amount of the tank liner material is found at the third 150 hour/6 month inspection, notify Gulfstream Aerospace of the findings and the estimated total amount of liner flakes found so that corrective action can be recommended.

E. Close access covers.

10. Wing / Flap Exterior — Inspection

NOTE: Refer to skin and flight control corrosion inspection / repair procedures for the following inspection. (See Chapter 27)

A. Exterior of inner and outer wing panels, top and bottom, for cracks, distortions, punctures, loose and missing rivets, evidence of corrosion, structural failure, obstructed vents and evidence of fuel and water/methanol leaks. On Aircraft with ASC 239, pay particular attention to lower dihedral splice fittings and around inspection port installations.

B. Fuel and water/methanol caps for condition, security and proper operation. Inspect interior of tanks as visible through filler cap opening.

C. Inspect fuel boost pumps for external security and evidence of leakage.

D. Inspect wing rear beam for:

- Condition, evidence of corrosion, structural failure and fuel leakage.

- Condition, security and evidence of leakage of water/methanol, fuel, fire protection and air conditioning installations including components, tubing, ducting, fittings and connections.
- Condition, security and evidence of overheating of electrical installations including components, wiring, plugs and connectors.

E. Navigation light for security and condition.

F. Wing light covers for condition and security.

G. Inspect ailerons and aileron tabs for security of attachment, condition and evidence of corrosion of upper and lower skins, condition and security of static discharge wicks, evidence of structural failure and freedom of operation.

H. Aileron trim tab and aileron spring tab attachment fittings and bearings for condition, security and signs of failure.

I. Stall warning transducer for condition and security. (Left wing only).

J. Wing flap, upper and lower skins, for condition, evidence of corrosion, chafing and structural failure.

K. Flap carriage for condition and security of attachment.

L. Wing flap installations including flap tracks, torque tubes, jack screws, actuators and flap seals for condition, security and evidence of corrosion.

M. External markings for legibility.

11. Aileron — Inspection

A. Open access panels shaded in Figure 207 and inspect for the following:

B. Aileron seals for condition and security.

C. Aileron, aileron trim tab and spring tabs for evidence of corrosion, clearance and travel.

NOTE: Visually check clearance between tabs and main surface for aileron trim and aileron spring tabs. This check should be made for full range of tab travel in case of trim tab and full range of travel in case aileron spring tabs. Gap should be at least 0.060 inch all along , top and bottom through the whole range of travel. If the gap is less than this value bend trailing edge of aileron skin up on the upper surface or down on lower surface until the gap is at least 0.060 inch. No special equipment is necessary to do this.

D. Aileron, aileron trim tab and spring tab, trim tab actuator, tab linkages, mechanisms, control rods and cables for condition, security and freedom of operation.

E. Aileron and aileron spring tab attachment fittings and bearings for condition, security and corrosion .

F. Static discharge wicks for condition, security and corrosion.

G. Inspect for presence of foreign objects then close access panels.

12. Outer Wing Fwd Beam — Visual or X-Ray Corrosion Inspection

NOTE: This inspection will require outer fuel cells be defueled and removed on aircraft having ASC 125 or draining and removal of water/methanol tanks on aircraft not having ASC 125.

A. Remove leading edges and open access covers as shown on Figure 208

B. Wipe fuel cells or water/methanol tanks dry.

C. Visually inspect interior, or x-ray double skinned area of outer wing in water/methanol tank area for Aircraft not having ASC 125, and outer wing fuel cell area for Aircraft having ASC 125. (Visual inspection will require removal of cells and liners.)

NOTE: When minor corrosion is found, it should be treated in accordance with Maintenance Manual. If corrosion is more than 0.010 inches deep over an area greater than one inch in diameter is found, contact Gulfstream Aerospace for disposition. We will need to know size, maximum depth, and location of all main corroded areas.

Following steps cover x-ray procedure that may be used in place of a visual inspection.

D. Position films per Tables I, II, III and Figure 209, Figure 210 and Figure 211.

E. Position x-ray head according to Figure 209, Figure 210 and Figure 211.

F. Position identification numbers 1 - 36 (6 per cell) on bottom side of wing plank. Place 6 numbers starting at 4 inches from front and back of bottom wing plank in center of cell and maintain an equal distance between the numbers.

G. Position identification in center of cell for forward and aft beams.

H. Radiograph bottom wing planks.

(1) Sample Technique

Kilovoltage	-	55kV
Milliamperage	-	4mA
Time	-	2 1/4 minutes
SFD	-	42 inches
Film Type	-	Kodak M
Film Size	-	Per Table I

(2) Position x-ray beam on a spot 1/4 panel span forward and aft on forward side and in middle of cell inboard to outboard (except for cell No. 5) and radiograph per sample technique.

(3) Repeat Step (2) for aft side of cells (except for cell No. 5).

(4) For cell No. 5 position the x-ray on a spot 1/4 the distance inboard to outboard on inboard side and in the middle of the cell forward and aft and radiograph per sample technique.

(5) Repeat Step (2) for the outboard side of the cell No. 5.

I. Radiograph forward beam.

(1) Sample Technique

Kilovoltage	-	50kV
Milliamperage	-	4mA
Time	-	2 1/4 minutes
SFD	-	36 inches
Film Type	-	Kodak M
Film Size	-	Per Table II

(2) Position x-ray beam on a spot in center of forward beam for each beam and radiograph per sample technique.

J. Radiograph aft beam.

(1) Sample Technique

Kilovoltage	-	50kV
Milliamperage	-	4mA
Time	-	2 1/4 minutes
SFD	-	36 inches
Film Type	-	Kodak M
Film Size	-	Per Table III

(2) Position x-ray beam on a spot in center of aft beam for each cell and radiograph per sample technique.

K. Radiograph front beam x-ray.

(1) Remove water/methanol fitting, P/N 159P10134-1.

(2) Fold water/methanol tank aft to facilitate positioning of x-ray film.

(3) Use x-ray techniques referenced in inspection procedure. Steps H.(1)-(5), I.(1)-(2), J.(1)-(2) and Table II.

(4) Position film and x-ray head per Figure 210. Water/methanol fitting hole should be in center of film.

(5) If front beam at Wing Station 320 is cracked, contact Gulfstream Aerospace Service Engineering Department for repair procedures. Following completion of repair, proceed with action required.

L. Ensure all x-ray film, film identification and foreign material is removed from wings before closing.

M. Position water/methanol tanks and connect snap hangers.

N. Install water/methanol fitting, P/N 159P10134-1. Fill water/methanol tank and leak check system.

NOTE: Upon installation of water/methanol fitting, P/N 159P10134-1, special care should be taken to prevent preloading of front beam web. After installation of P/N 159P10011-7/-9 line connecting fitting to water/methanol pump, ensure that beam is not loaded (dimpled). If web is loaded, line should be loosened and repositioned until dimple is no longer present.

O. Inspect for presence of foreign objects then install access covers removed in Step A. above. Leak check wing.

P. Install leading edge sections removed in Step A. above.

TABLE I - BOTTOM WING PLANK

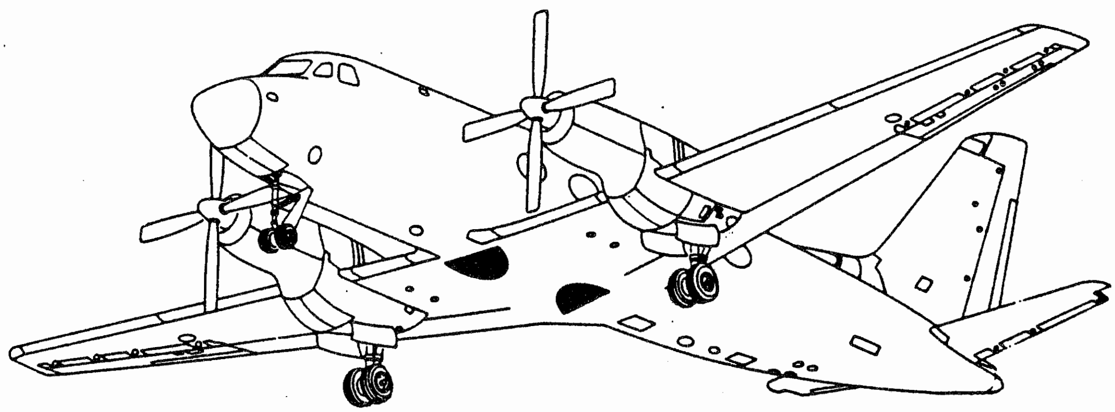
CELL NO.	FILM SIZE	NO. OF FILMS	NO. OF EXPOSURES	ORIENTATION OF FILM
1	7″ x 17″	6	2	Inboard to Outboard
2	7″ x 17″	6	2	Inboard to Outboard
3	7″ x 17″	5**	2	Inboard to Outboard
4	7″ x 17″	5**	2	Inboard to Outboard
5	7″ x 17″	4	2	Forward to Aft
6	7″ x 17″	4	2	Inboard to Outboard
7	7″ x 17″	4	2	Inboard to Outboard

* ONE HALF THE FILM USED FOR FORWARD EXPOSURE AND ONE HALF USED FOR AFT EXPOSURE.
** THREE FILMS ARE USED FOR THE FORWARD EXPOSURE AND TWO FILMS FOR THE AFT EXPOSURE.

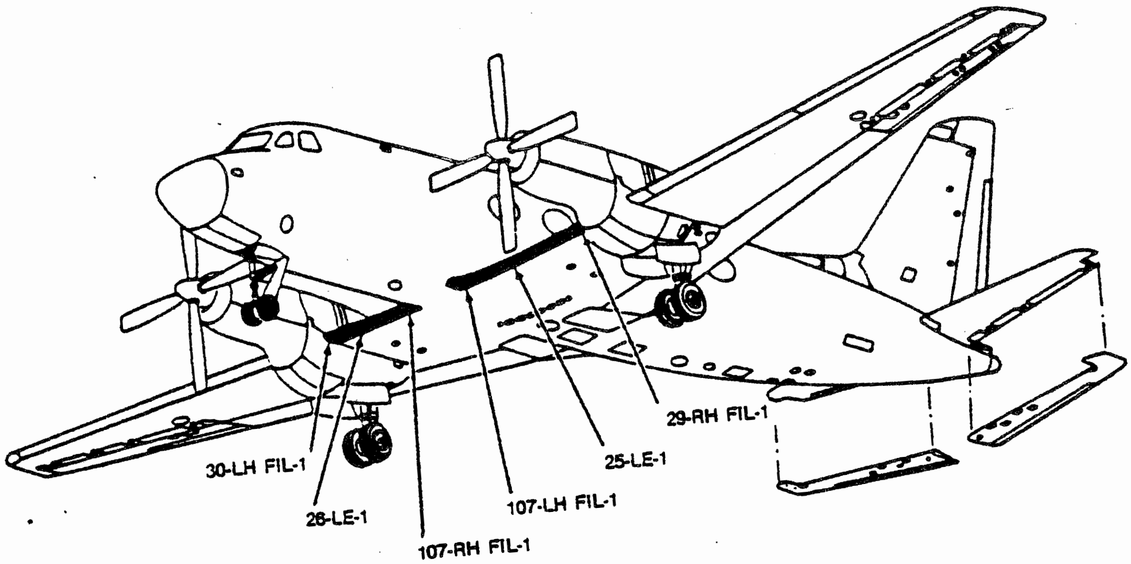
TABLE II - FORWARD BEAM

CELL NO.	FILM SIZE	NO. OF FILMS	NO. OF EXPOSURES
1	5″ x 7″	3	1
2	7″ x 17″	1	1
3	7″ x 17″	1	1
4	7″ x 17″	1	1
5	5″ x 7″	2	1
6	7″ x 17″	1	1
7	7″ x 17″	1	1

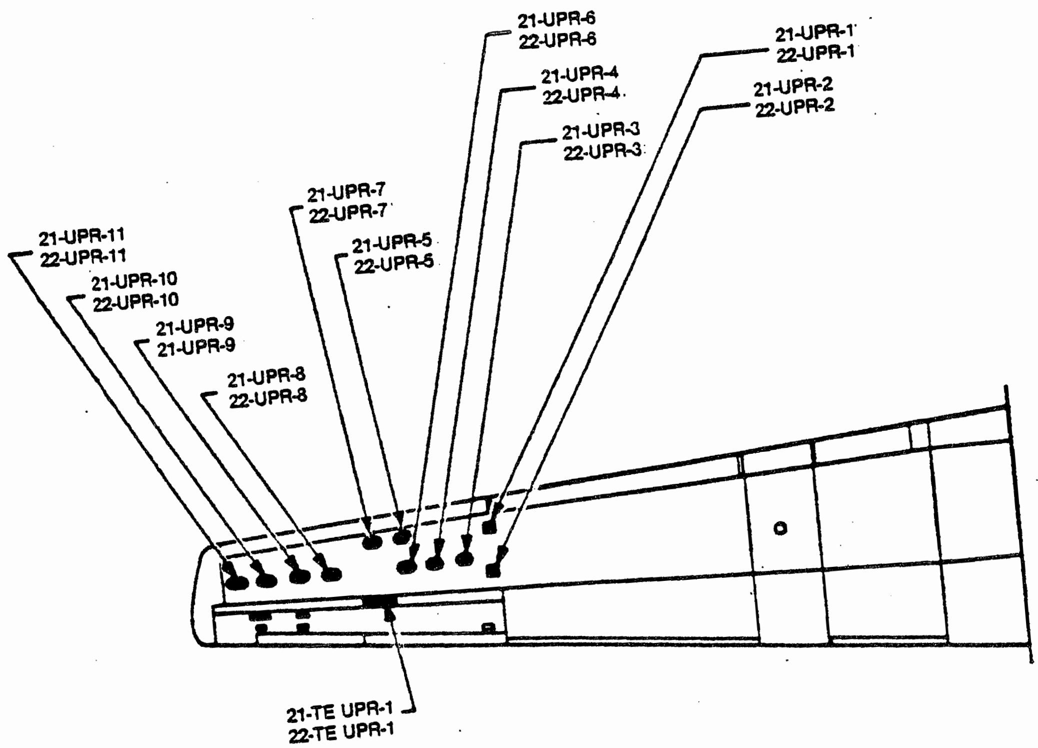
TABLE III - AFT BEAM			
CELL NO.	FILM SIZE	NO. OF FILMS	NO. OF EXPOSURES
1	7″ x 17″	1	1
2	3 1/2″ x 17″	1	1
3	7″ x 17″	1	1
4	3′ 1/2″ x 17″	1	1
5	3′1/2″ x 17″	1	1
6	3′ 1/2″ x 17″	1	1
7	3′ 1/2″ x 17″	1	1



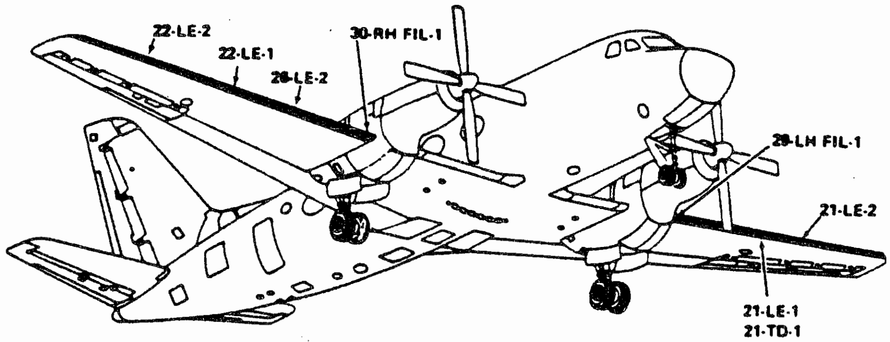
Center Wing Panels
Figure 201.



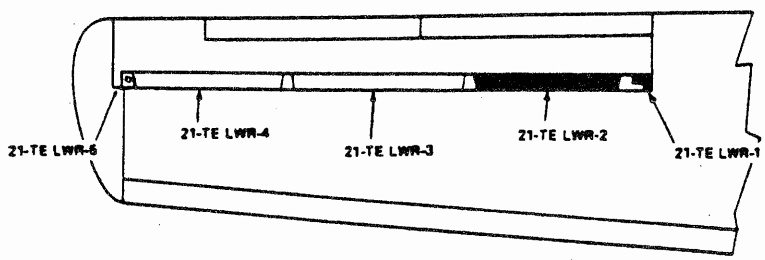
Inboard Wing Beam Inspection
Figure 202.



Wing Panel
Figure 203.

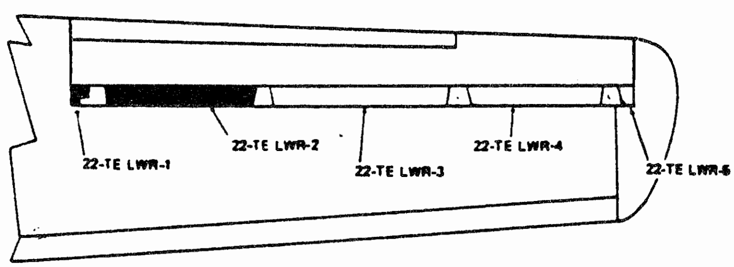


LEGEND
21 — LH OUTER WING PANEL
25 — LH INNER WING PANEL
LE — LEADING EDGE
TD — TRANSDUCER
22 — RH OUTER WING PANEL
26 — RH INNER WING PANEL
107 — FUSELAGE
FIL — FILLET, WING TO FUSELAGE
29 — LH NACELLE & ACCESSORY SECTION
30 — RH NACELLE & ACCESSORY SECTION



LEGEND
21 — LH OUTER WING PANEL
LWR — LOWER
TE — TRAILING EDGE

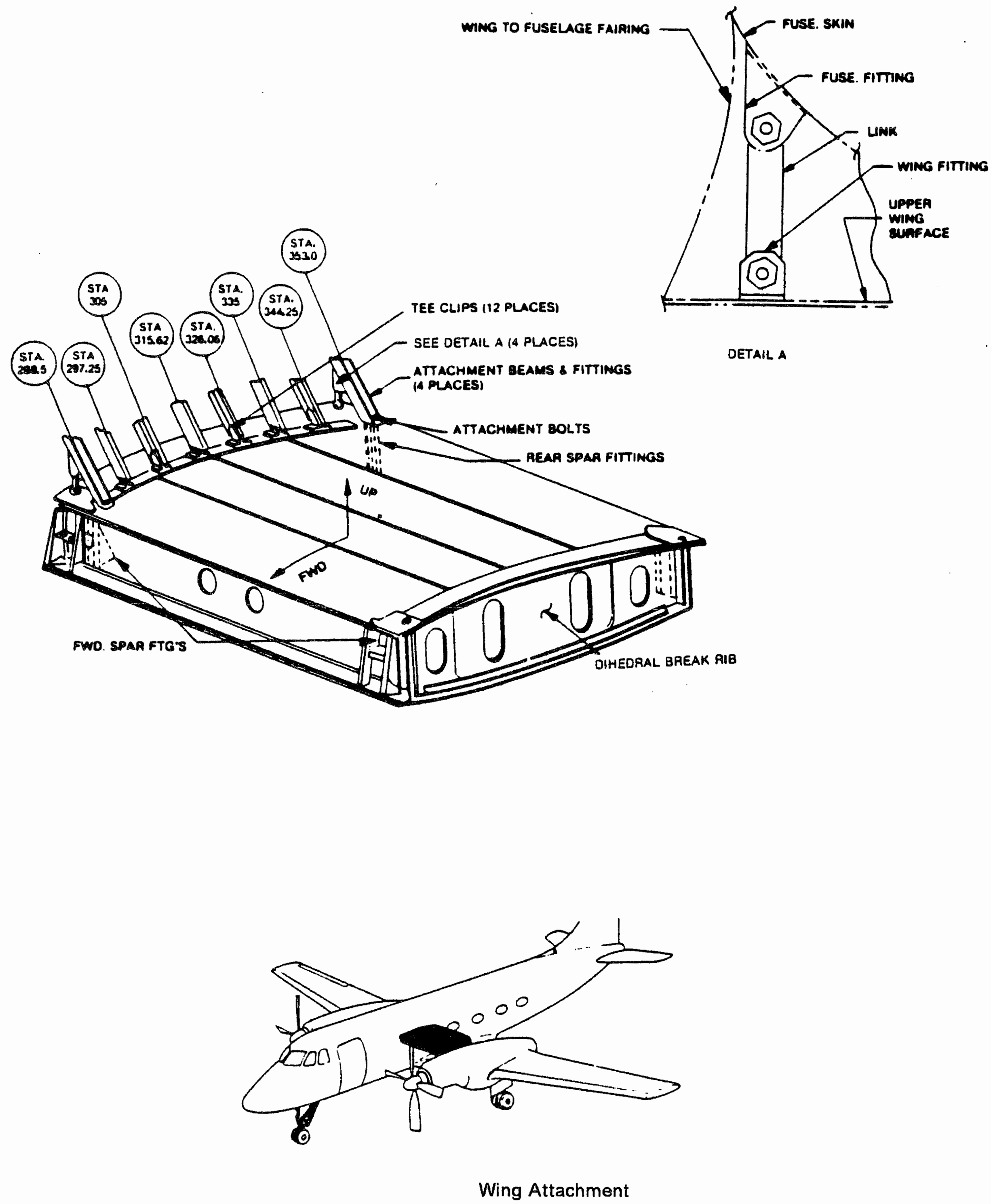
LEFT OUTER WING (VIEW LOOKING UP)



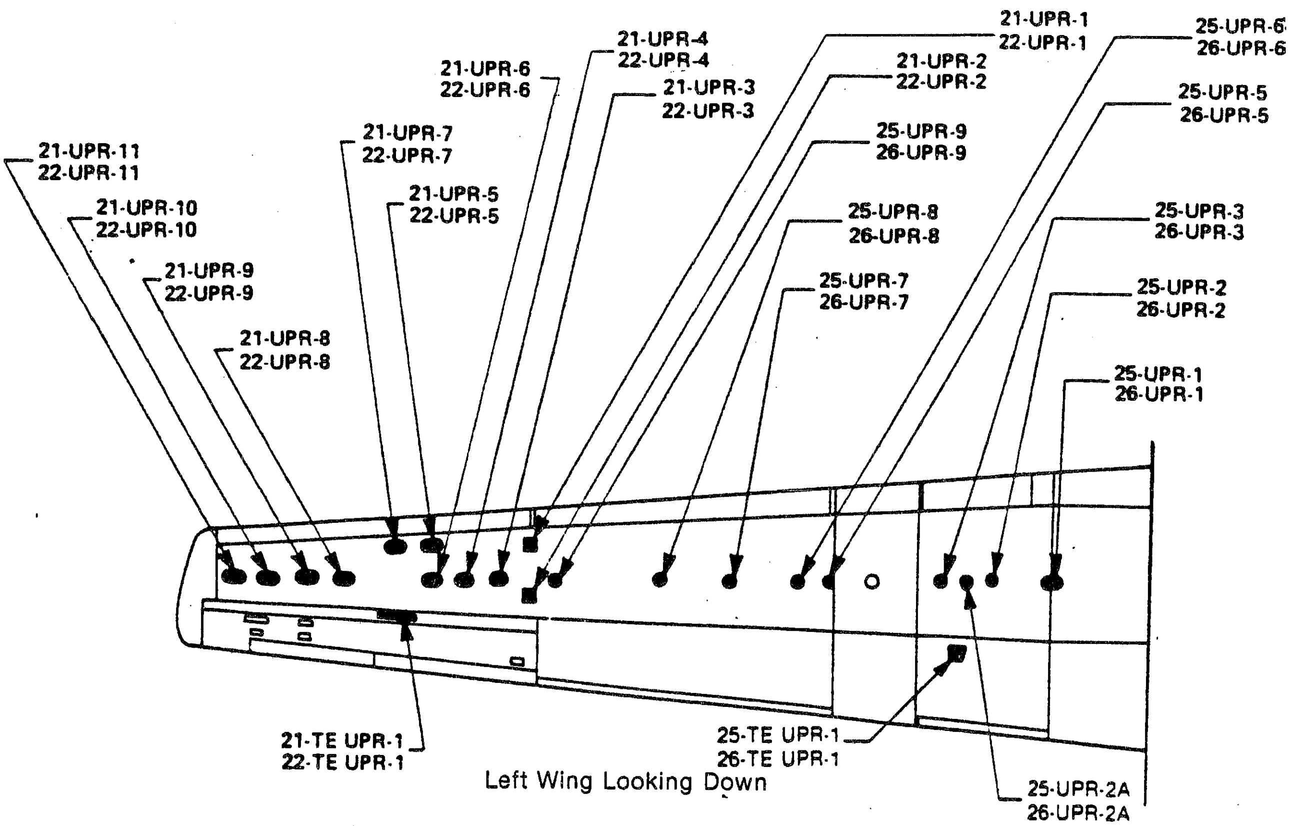
LEGEND
22-RH OUTER WING PANEL
LWR — LOWER
TE — TRAILING EDGE

RIGHT OUTER WING (VIEW LOOKING UP)

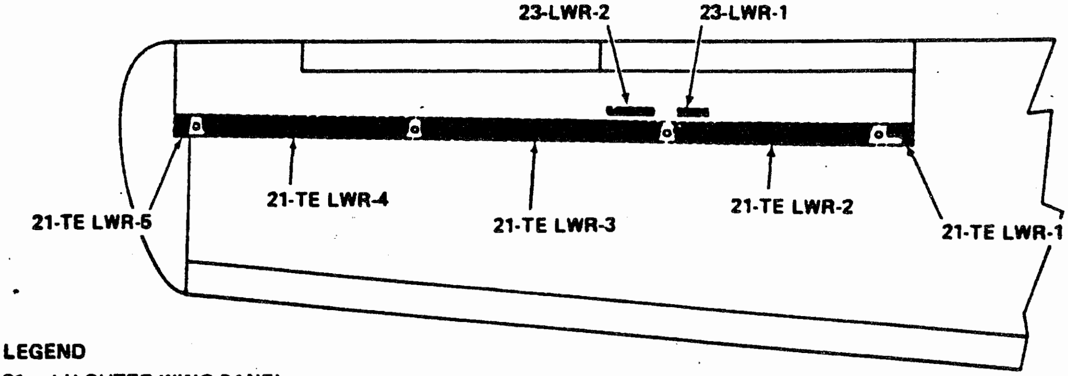
Wing Panel Inspection
Figure 204.



Wing Attachment
Figure 205.

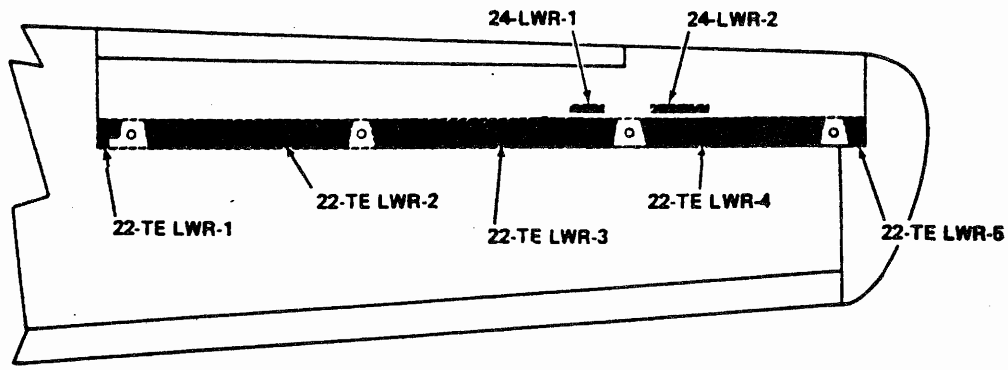


Upper Wing Panels
Figure 206.



LEGEND
21 – LH OUTER WING PANEL
23 – LH AILERON
LWR – LOWER
TE – TRAILING EDGE

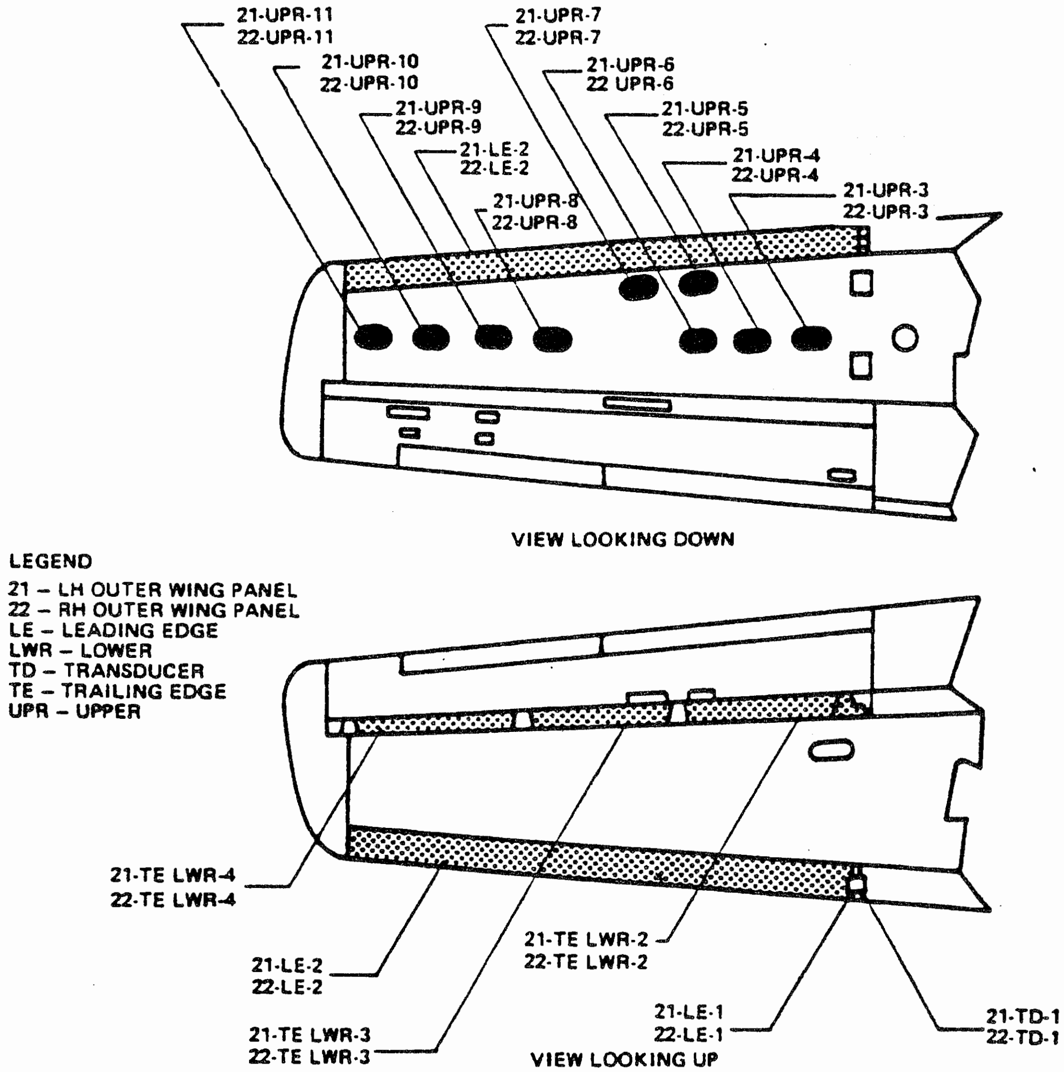
LEFT AILERON (VIEW LOOKING UP)



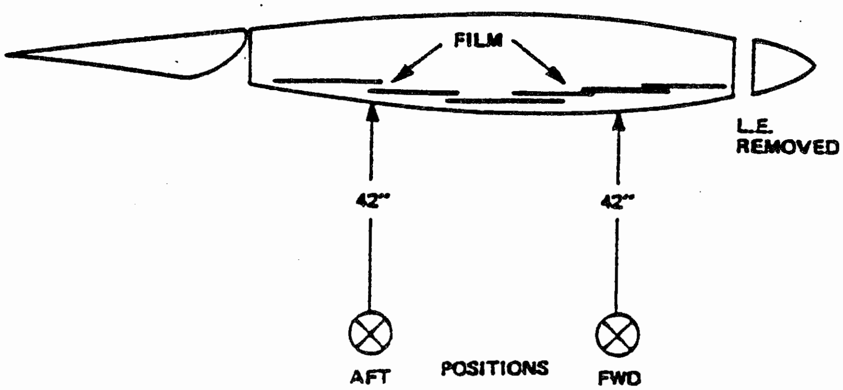
RIGHT AILERON (VIEW LOOKING UP)

LEGEND
22 – RH OUTER WING PANEL
24 – RH AILERON
LWR – LOWER
TE – TRAILING EDGE

Aileron
Figure 207.

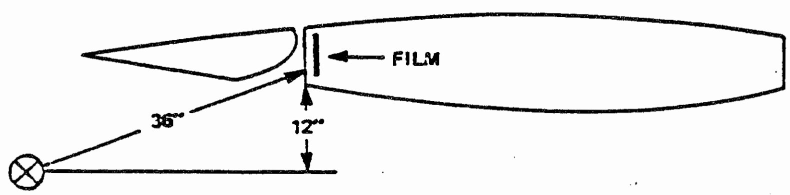


Wing Corrosion — Inspection
Figure 208.

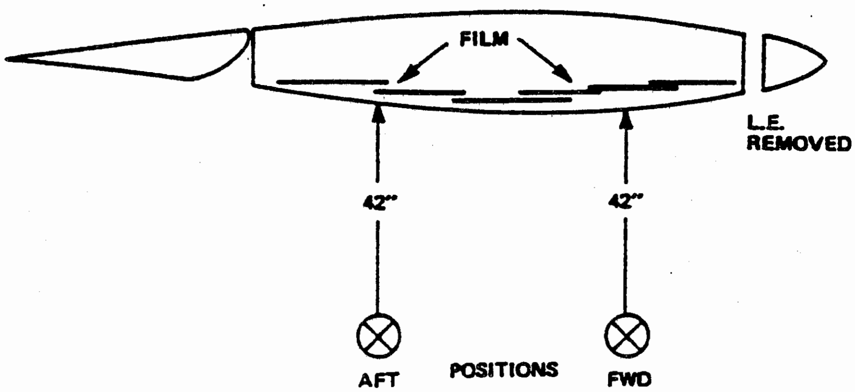


Bottom Wing Plank
Figure 209.

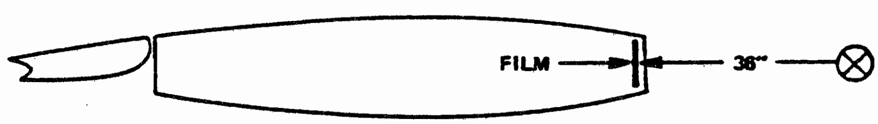
Forward Beam
Figure 210.



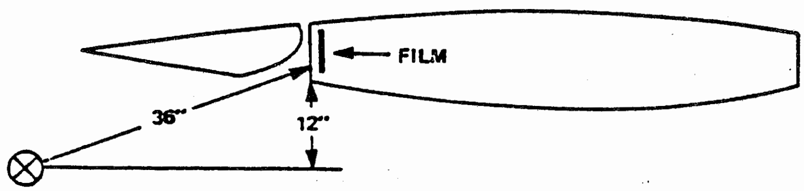
Aft Beam
Figure 211.



Bottom Wing Plank
Figure 209.



Forward Beam
Figure 210.



Aft Beam
Figure 211.

LIST OF EFFECTIVE PAGES

Chapter 61
PROPELLERS

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*Pages revised, added, or deleted by this revision.

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CHAPTER 61

INSTRUMENTS

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*61	ii	September 11/92	*61-0	211	September 11/92
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* Pages revised by Revision 48
** Pages added by Revision 48
*** Pages deleted by Revision 48

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CHAPTER 61

INSTRUMENTS

PROPELLERS — DESCRIPTION / OPERATION

1. Description

The aircraft is equipped with two Rotol, 11 foot 6 inch diameter, four bladed, hydraulically operated (engine oil type) full feathering, two stage lock, constant speed propellers with electrical and mechanical control. The blades are constructed of solid, anodized aluminum allow with leading edge rebated for the fitting of electric de-icing boots. The propeller system incorporates an external hydraulic governing system to maintain precise control during all operating conditions. The governing system provides the power required to change blade angle or pitch to compensate for variations in blade loading and to maintain a constant rpm. Continuous indication of propeller operation and warning of malfunction are provided by a system of indicator lights. Dual purpose controls (located on the pedestal) provide for the operation and the control of propeller blade angle, blade lock removal, plus fuel control functions. Auto feathering, manual feathering and propeller synchronizing systems are also incorporated.

There are ten propeller system warning lights, which are located at the center of the eyebrow panel. Eight of these lights incorporated individual press-to-test sockets and dimming bezels. The four center lights (two FLIGHT FINE PITCH and two CRUISE PITCH) use 12V bulbs. The remainder, including the two FEATHERING PUMP indicator lights, use 24V bulbs.

2. Operation

(See Figure 1 and Figure 2)

The propeller blades are controllable from 0° (flat pitch) to 85 1/2° (full feather). This movement is produced by engine oil hydraulic power, controlled by the governor and electrical valves. The propeller blade angle is a function of engine rpm, aircraft speed and throttle lever setting. To prevent an improper blade angle from being attained, two mechanical locks are provided within the propeller hub. These locks prevent uncontrolled blade fining-off, which in turn prevents dangerous overspeeding. The locks do not impede blade angle changes in a coarsening direction, thus, ability to feather is unrestricted. The locks are 34 1/2° known as the cruise lock, and at 20°, known as the flight fine pitch lock. (See Figure 3) The cruise lock prevents overspeed in the case of a malfunction when at cruise blade angles. The flight fine pitch lock prevents overspeed of the propeller when in blade angles below cruise, such as in takeoff, approach and certain climb segments. It also performs a second function in preventing the propeller from going into flat pitch until it is required for braking purposes after touchdown.

The throttle for each engine, as shown in Figure 1, serves the dual purpose of a normal throttle in governing fuel flows from the fuel control unit of the engine, plus an additional function as a propeller pitch control through a synchronized linkage system. Movement of the throttle is transmitted to the Propeller Control Unit (PCU) as well as to the Flow Control Unit (FCU). As the lever is moved forward, fuel flow increases, and the constant speed rpm of the propeller is increased accordingly. Normal engine turbine speeds range from idle, 6500 to 7500 rpm, to takeoff at 15,000 rpm. Propeller rpm is approximately at a ratio of 1:11 of turbine rpm.

The high-pressure fuel cock lever for each engine is a four position, dual function lever as shown in Figure 4. It controls the high-pressure fuel cock in the flow control unit, allowing fuel to flow or be shut off. It also functions as a manual feathering lever and provides a manual method of removing the 34 1/2° lock from its propeller. The FEATHER position is utilized during manual feathering of the propeller, not involving an auto feathering cycle. Fuel is off in this position. The FUEL OFF and FUEL ON positions refer merely to the fuel control functions of the high-pressure cock lever. In these two positions, the propeller governor is in its normal constant speed operation. (See Figure 5) The CRUISE LOCK OUT position actuates mechanical linkages to the propeller control unit to initiate manual removal of the 34 1/2° lock from its propeller in the event of failure of the automatic cruise lock removal devices. Fuel is on in this position. In cruise lock out, the propeller governor has control over blade angle and is in its normal constant speed operation.

The flight fine pitch lock selector lever is a single lever with a spring loaded detent and latch, located in the center of the pedestal between the throttles. This lever has only two positions, FLIGHT and GROUND INTERLOCK. It controls the removal of the flight fine pitch locks (20°) from both propellers. Movement of this lever from FLIGHT position to the GROUND INTERLOCK position will remove the 20° locks simultaneously from both propellers and allow fining off to flat pitch (ground fine). (See Figure 6) This lever actually moves cams in the

pedestal, which in turn actuate two switches controlling electrical energy to the 20° lock removal devices in each propeller control unit. There is no direct, mechanical way to remove these locks. The flight pitch lock selector lever is the only way to remove the 20° locks. It is under the direct manual control of the pilot, and requires two separate and distinct movements to place it in the GROUND INTERLOCK (lock removed) position. This will be discussed further and in detail under pedestal safety features.

The single gust lock lever is located forward of the flight fine pitch lock selector lever in the center of the pedestal. Besides locking the flight controls when engaged, the gust lock lever also activates, by mechanical action within the pedestal, certain safety devices to prevent accidental misuse of the other pedestal controls involved with the propeller.

All electrical wiring as shown in Figure 7 and Figure 8 is enclosed in flexible metallic conduit and relays and resistors are isolated from other electrical circuits. Because of the importance of this system, certain particularly vital parts of this system are kept separate from all other propeller wiring and are run in separate conduits. The following major components make up the propeller electrical system.

UNIT	NO PER A/C	LOCATION
Feather Pump Fuses (Current Limiters)	2	Inverter Fuse Panel
Feather Pump Relay	2	Inverter Relay Box
Feather Pump Indication Circuit Breaker	2	Inverter Relay Box (Circuit Breaker Section)
Propeller System Circuit Breakers (Except Feather Pump Indication)	10	Copilots Circuit Breaker Panel
Propeller Control Relay	2	Propeller Junction Box Floorboard No. 11
Isolation Relay	2	Propeller Junction Box Floorboard No. 11
Propeller Warning Light Resistor	4	Propeller Junction Box Floorboard No. 11
Propeller X Relay	1	Propeller Junction Box Floorboard No. 11
Propeller Z Relay	1	Propeller Junction Box Floorboard No. 11
Propeller System Warning Lights	10	Eyebrow Panel
Propeller Feather Switch	2	Eyebrow Panel
Propeller Synchronizing Switch	1	Eyebrow Panel
Synchronizing Circuit Disabling Switch (Landing Gear Handle)	1	Cockpit Floor Forward of Pedestal (Landing Gear Handle Linkage)
Alternator - Propeller Synchronizing	2	Left and Right Accessory Gearbox
Propeller Synchronizer (SYNCH-N) Circuit Breaker	1	Pilots Circuit Breaker
Propeller Synchronizer Relay	1	Right Nacelle Relay Panel
Propeller Synchronizer Corrector Motor	1	Right Nacelle (Inboard Side)
Propeller	2	Forward of Nacelle
Propeller Control Unit (PCU)	2	Mounted on Lower Right Forward Section of Engine
Flight Safety Stop (Cruise Lock) Withdrawal Solenoid	2	Propeller Control Unit
Flight Fine Pitch Lock Withdrawal Solenoid	2	Propeller Control Unit
Pitch Coarsening Solenoid	2	Propeller Control Unit
Flight Safety Stop Pressure Switch (Cruise Lock Out)	2	Propeller Control Unit
Hub Switch	3	Mounted in each Propeller
Auto Feathering Torque Sensing Switch	2	Mounted on their respective Engine, above the Water/Methanol Unit
Feather Pump	2	Respective Engine near the Propeller Control Unit

3. Automatic Cruise Lock Removal

Both 34 1/2° cruise locks are removed automatically when both propellers are below the 36 1/2° blade angle, 2° above the lock. (See Figure 9) This is accomplished by dual electrical hub switches within each propeller completing circuits to a solenoid valve in each propeller control unit. Dual switches are utilized for safety on each side. Since it is necessary for both propellers to pass through their respective 36 1/2° hub switches in order to automatically remove the cruise locks, one propeller alone running away and fining off through its hub switch due to a malfunction cannot remove its own lock. In this case, the malfunctioning propeller will be stopped at its 34 1/2° lock. This type of arrangement provides the safety of a lock for overspeed protection, and also enables both locks to be removed automatically when both propellers fine off normally due to retardation of power and airspeed. This system is completely immobilized by placing the flight safety lock switch on the eyebrow panel in the EMERGENCY position. This switch is just to the right of the eight amber propeller indication lights. It is kept in the NORMAL position unless it is desired to prevent automatic cruise lock removal.

4. Manual Cruise Lock Removal

The 34 1/2° cruise locks can also be removed manually by moving the appropriate high-pressure fuel cock lever to its CRUISE LOCK OUT position. (See Figure 10) This action initiates cruise lock removal for that propeller only by mechanical means through direct cable and pushrod connections to the individual propeller control unit. No electrical energy is necessary.

5. Automatic Coarsening

Auto coarsening takes place in the event of a propeller electrical, hydraulic or mechanical system malfunction at low blade angles wherein the propeller attempts to go below 18° in flight. A hub switch at 18° within the propeller will complete an electrical circuit to override the propeller governor and force the propeller above 18°. (See Figure 11) This will occur only when the flight fine pitch lock selector lever is in the FLIGHT position. Auto coarsening will not occur when the flight fine pitch lock selector lever is in the GROUND INTERLOCK position, since it is desirable that the propeller fine off to flat pitch at that time.

CAUTION: AT ANY TIME THE FLIGHT FINE PITCH LOCK SELECTOR LEVER IS IN THE FLIGHT POSITION, PROPELLER BLADE ANGLE CAN BE NO LESS THAN 18° BECAUSE OF AUTO COARSENING.

Auto coarsening is not available in emergency dc operation.

6. Auto Feathering

The auto feathering system feathers a propeller when all of the following conditions exist:

A. Both engine high-pressure fuel cock levers are in operative positions (FUEL ON or CRUISE LOCK OUT).

B. The opposite engines propeller is not in the process of auto feathering.

C. The affected engines throttle lever is above 11,500 rpm quadrant position (slightly above the 30° position of quadrant travel).

D. The power loss on that engine is sustained so that its torque pressure drops and stays below 50 pounds. The auto feathering system is self-recoverable and noncommitted. It has no time delay system associated with its institution, and will revert the propeller system to normal constant speed operation if the torque pressure recovers and goes above 50 pounds. There is no disarming switch in the cockpit, as this system is armed on each propeller at all times except when that engine throttle lever is below the 11,500 rpm (approximately 30°) quadrant position. The operative engine auto feathering system is also disarmed automatically when the opposite engine is inoperative or in the process of auto feathering (to prevent simultaneous auto feathering of both engines).

In order to feather any oil operated propeller, whether automatically or manually, two basic requirements must be accomplished, the oil must be ported to the correct side of the propeller operating piston to force the propeller toward increase pitch, and sufficient oil under pressure must be supplied to move the propeller to feather. (See Figure 12) The auto feathering system provides for the requirements mentioned above. These are accomplished by completion of certain electrical circuits, one of which places the governor in an overspeed condition, porting the oil to increase pitch, another starts the feathering pump to deliver sufficient oil under pressure from the engine oil tank to take the propeller to its desired pitch position.

CAUTION: THE FEATHER PUMP DOES NOT STOP AUTOMATICALLY, AND HAS A THREE MINUTE TIME LIMIT. THEREFORE, IT IS ESSENTIAL THAT THE AUTO FEATHERING BE FOLLOWED BY A MANUAL FEATHERING PROCEDURE WITHIN THREE MINUTES FROM THE INSTITUTION OF THE AUTO FEATHER. THIS ACTION DE-ENERGIZES THE PUMP.

Auto feathering is not available in emergency dc operation.

7. **Manual Feathering**

(See Figure 13)

To feather a propeller manually (without institution of an auto feather) two basic operations are necessary:

A. Place the high-pressure fuel cock lever for that engine in the FEATHER position. (This not only shuts the fuel off but ports the oil to the increase side of the operating piston mechanically.)

B. Push and hold in the feather pump button for that engine and hold it in until propeller rotation ceases. This supplies the oil to accomplish feathering.

NOTE: Slight rotation of the propeller may occur even if full feathered. This is normal due to the small internal resistance of the engine and a slight blade angle. This rotation is directly proportional to the airspeed. If rotation increases over a period of time, merely push the feather pump button for a few seconds to send a charge of oil to drive the propeller back to feather.

8. **Unfeathering**

Unfeathering in flight consists of:

A. Moving the high-pressure fuel cock lever to the CRUISE LOCK OUT position to release the governor to constant speed mode and remove the 34 1/2° lock, thus enabling the propeller to establish its minimum governing rpm for a light off. (See Figure 14)

B. Pushing the feather pump button for a few seconds starts the propeller toward decrease pitch.

Booster pumps must be on, and the airstart ignition utilized, to light off the engine. This is not directly concerned with the actual propeller movement. Actual step-by-step unfeathering procedures are covered in the Flight Manual, Emergency Procedures Section.

9. **Automatic Cruise Lock Removal with One Propeller Feathered**

The system is capable of automatic cruise lock removal with one propeller feathered. Since the prerequisite for automatic removal of the 34 1/2° cruise locks is that both propellers must be below their respective 36 1/2° hub switches, it is evident that with a propeller feathered (85 1/2° blade angle) it could not possibly be below its 36 1/2° hub switch. It could not arm its half of the automatic lock removal circuit and therefore, automatic lock removal could not take place. Since it is desirable to have automatic cruise lock removal on the operative propeller, even with the other propeller feathered, the 36 1/2° hub switch on each propeller has two segments. One completes the circuit at 36 1/2° blade angle and below, the other segment completes the same circuit at full feather.

With a fully feathered propeller, one-half of the cruise lock automatic removal circuit is completed. As the operative propeller passes through its 36 1/2° hub switch, it completes the second half of the cruise lock removal circuit and the operative propeller comes through in the normal manner. The feathered propeller cruise lock removal solenoid is also energized, but since the propeller is feathered, it is of no consequence, since no oil pressure is available.

10. **Pedestal Safety Features**

A. To prevent takeoff with the gust lock positioned to ON, simultaneous advancement of both throttle levers is limited to approximately 30° of quadrant travel (approximately 11,000 rpm). However, individual throttles can be moved to full power for engine checkout, with the opposite throttle moving aft automatically as the one is moved forward past the 30° quadrant position. This is accomplished by a mechanical whiffle tree mechanism built into the pedestal which becomes activated only when the gust lock is placed on.

B. To prevent application of the gust lock in flight, mechanical interlocks within the pedestal prevent any movement of the gust lock to the ON position with the flight fine pitch lock selector lever in the FLIGHT position.

C. To prevent inadvertent removal of the 20° locks, the flight fine pitch lock selector lever cannot be moved to GROUND INTERLOCK position until the throttles are below the 30° quadrant position.

WARNING: UNDER NO CIRCUMSTANCES SHOULD THE FLIGHT FINE PITCH LOCK SELECTOR LEVER EVER BE MOVED FROM THE FLIGHT POSITION UNTIL ALL THREE WHEELS ARE FIRMLY AND SOLIDLY ON THE GROUND. IF THIS WARNING IS DISREGARDED, AT THE AIRSPEEDS WHICH OCCUR ON APPROACH, MOVEMENT OF THIS LEVER TO THE GROUND INTERLOCK POSITION WILL ALLOW BOTH PROPELLERS TO FINE OFF TO FLAT PITCH WITH A RESULTANT LOSS OF PROPELLER THRUST. THIS IS THE EQUIVALENT OF REVERSING BOTH PROPELLERS IN THE AIR, WITH CONSEQUENT CATASTROPHIC RESULTS.

D. The 20° lock is automatically activated on takeoff by mechanical linkages within the pedestal which carry the flight fine pitch lock selector lever to the FLIGHT position when both throttles pass through the 30° quadrant position on application of takeoff power. This action activates the 20° lock protection for both propellers.

NOTE: This cannot be accomplished with the gust lock on. The gust lock mechanism (when in ON position) also prevents any possible movement of the flight fine pitch lock selector lever out of the GROUND INTERLOCK position with or without throttle movement.

E. The flight fine pitch lock selector lever incorporates a detented lock and latching mechanism in the FLIGHT position so as to require two separate motions before the lever can be moved rearward to GROUND INTERLOCK to remove the 20° locks. A small latch on the aft side of the lever must be moved forward and held there while the entire lever and latch is moved rearward to GROUND INTERLOCK. This can only be done when both throttles are retarded.

NOTE: The throttles will only carry the flight fine pitch lock selector lever to the FLIGHT position on power application. It must be moved to GROUND INTERLOCK by the above described double motion.

11. Propeller Indication Lights

(See Figure 15 and Figure 16)

A. Eight amber lights, located in the center of the eyebrow panel, give the crew propeller blade angle information. Each light is of the press-to-test type with individual dimming bezels. The obvious impression that the four lights on the right side refer to the right propeller and the four on the left refer to the left propeller is not correct. The outboard lights are directional, but the two groups of inboard lights are not. These eight lights supply explicit information to the crew, and if taken for their exact purpose, they will not be difficult to understand. In learning the action of these lights, be careful not to read things into them. The information which follows pertaining to these lights will be discussed according to their meaning when they are on. The eight lights, during normal cruise flight, are off. They come on in a descending sequence as airspeed drops and the aircraft descends and lands. As the aircraft rolls out and taxis, all eight lights are on. Before takeoff (at idle rpm's), all eight lights are on, going off in an upward sequence until cruise speed is reached, when they are all off.

B. The two top lights are called the CRUISE PITCH lights. The cruise pitch lights always come on together, never separately. Each light socket contains a 12V bulb.

CAUTION: DO NOT PUT 24V BULBS IN THESE LIGHT SOCKETS.

The cruise pitch lights come on when either propeller is at a blade angle of 36 1/2° or below, and also when either propeller is FULL FEATHERED. They are part of the automatic cruise lock removal system, if the FLIGHT SAFETY LOCK switch is positioned to EMERGENCY, they cannot come on since this entire circuit is made static.

WARNING: SINCE ILLUMINATION OF THE CRUISE PITCH LIGHTS CAN DENOTE ELECTRICAL INITIATION OF CRUISE LOCK REMOVAL, IT IS UNSAFE TO FLY FASTER THAN 250 KNOTS (TRUE AIR SPEED) WITH THOSE LIGHTS ON. IF HIGHER SPEEDS MUST BE USED, SET SAFETY LOCK SWITCH TO EMERGENCY POSITION.

See Flight Manual — Emergency Procedure Section.

C. The cruise lock out lights are the second two from the top. They are spaced outboard and below the cruise pitch lights. They are individually operated by each individual propeller cruise lock removal mechanism, therefore can be considered directional. The actual device causing these lights to come on is an oil pressure switch within each propeller control unit. This switch is activated only when the propeller hydraulic system is activated to initiate 34 1/2°lock removal. Without oil pressure, the light is off. Each light contains a 24V bulb.

CAUTION: DO NOT USE 12V BULBS IN THESE LIGHT SOCKETS.

These cruise lock out lights come on when automatic cruise lock removal is initiated by the 36 1/2° hub switches in both propellers as discussed previously. They also come on individually by moving the high-pressure fuel cock lever to the CRUISE LOCK OUT position, since this is the manual initiation of cruise lock removal. Both high-pressure fuel cock levers must be in the cruise lock out position to result in both lights coming on. One high-pressure fuel cock lever in the CRUISE LOCK OUT position will only turn on its own CRUISE LOCK OUT light.

NOTE: These lights indicate only that the propellers hydraulic system has been initiated to remove the 34 1/2° lock. They do not indicate that the lock is removed. The actual removal is indicated by observing the TGT and RPM indicators.

The cruise lock out lights will be inoperative in emergency dc operation.

D. The flight fine pitch lock lights are the third two from the top. They are grouped closely, in the same manner as the cruise pitch lights. These lights always come on together. They cannot come on separately. Each light socket contains a 12V bulb.

CAUTION: DO NOT USE 24V BULBS IN THESE LIGHT SOCKETS.

The flight fine pitch lock lights are lever position indicators. Both are on when the flight fine pitch lock lever is in the GROUND INTERLOCK position. These lights do not indicate that the 20° locks have been removed. They merely mean that the flight fine pitch lock lever is in GROUND INTERLOCK position to Initiate lock removal.

E. The below flight fine pitch lock lights are the bottom two and are spaced outboard. They are turned on individually by each propellers 18° hub switch (2° below the 20° lock). These lights come on when their individual propellers blade angle is at 18° or below. Since the propeller cannot get to the 18° hub switch on decreasing blade angles unless the 20° locks are removed, these bottom lights can be considered an absolute indication that the 20° lock is removed. Each socket contains a 24V bulb.

CAUTION: DO NOT USE 12V BULBS IN THESE LIGHT SOCKETS.

Since the 18° hub switches also control autocoarsening, these lights blink on and off warning the crew that auto coarsening is in progress during low speed approach conditions. On normal landing, these lights come on immediately after the flight fine pitch lock lights. The below flight fine pitch lock lights are inoperative in emergency dc operation.

12. Summary of Propeller Light Indications — Normal DC Power

A. Electrical Power ON - Engines Not Running: Blade Angles Zero

(1) All lights ON except two cruise lock out lights. (Without oil pressure, lights will be off).

B. Engine Starting

(1) Cruise lock out lights come on as each engine is started and oil pressure develops.

(2) With both engines running at idle all lights are on.

C. Takeoff

(1) With the gust lock in the OFF position, both throttle levers advanced to takeoff, flight fine pitch lock lever moves to FLIGHT position and the below flight fine pitch lock and flight fine pitch lock lights go off as propellers go above 18° of blade angle.

(2) The top four lights remain on.

D. Climb

(1) Cruise lock out and cruise pitch lights go off successively as blade angles increase.

NOTE: Avoid sustained operation when these lights are blinking. This indicates the propeller blade angles are cycling around 36 1/2° and all automatic mechanism are also cycling.

E. Cruise

(1) All lights are off.

WARNING: SINCE ILLUMINATION OF THE CRUISE PITCH LIGHTS CAN DENOTE ELECTRICAL INITIATION OF CRUISE LOCK REMOVAL, IT IS UNSAFE TO FLY FASTER THAN 250 KNOTS (TRUE AIR SPEED) WITH THOSE LIGHTS ON. IF HIGHER SPEEDS MUST BE USED, SET SAFETY LOCK SWITCH TO EMERGENCY POSITION.

Refer to Flight Manual - Emergency Procedures Section.

F. Institution of Auto feathering

(1) Feather pump light for feathering engine comes on, then cruise pitch lights come on as the propeller blade angle arrives at FULL FEATHER (if the top two lights are not already on).

G. Pressing the Feathering Pump Button

(1) Feather pump light for that engine comes on.

NOTE: This action does not cause the propeller to feather unless the high-pressure fuel cock lever has been placed in the FEATHER position. The feather pump button only operates the feather pump and since the governor has control of blade angle, nothing happens. It merely puts another pump on the line. Releasing the button stops the pump and turns off the light.

H. Descent with the Flight Safety Lock switch in EMERGENCY.

(1) Cruise pitch lights cannot come on because the circuit is dead.

(2) 34 1/2° locks have to be removed with the high-pressure fuel cock lever.

(3) Cruise lock out lights come on when high-pressure fuel cock lever is at CRUISE LOCK OUT.

(4) All other lights react normally.

I. Descent with one propeller feathered.

(1) Cruise pitch lights are turned on.

(2) Cruise lock out lights come on in operative engine only. Feathered engine has no oil pressure.

(3) Flight fine pitch lock lights come on when the flight fine pitch lock selector lever is placed in GROUND INTERLOCK position.

(4) Only operative engines below flight fine pitch lock light comes on as it is the only one which can go below 18°.

13. Flat Pitch and Blade Marks

Each propeller blade has a one inch stripe at the cuff end. The adjoining part of the spinner has a similar stripe. These stripes line up when the blades are in flat pitch, 0° This is a visual indication for checking that the respective propeller is in flat pitch before starting engine. If the propeller blade marks do not align, this indicates that the propeller is not in flat pitch. In this case, do not attempt to start the engine until the blades are returned to flat pitch. This can be accomplished quickly by proceeding as follows:

A. Battery switch to NORMAL.

B. High-pressure fuel cock lever to CRUISE LOCK OUT.

C. Press feather pump button for that engine until blade marks align.

CAUTION: DO NOT USE THE FEATHER PUMP INDISCRIMINATELY WHEN THE ENGINE IS NOT RUNNING. USE IT ONLY LONG ENOUGH TO GET FLAT PITCH. THIS IS BECAUSE OIL FROM THE PROPELLER GOES TO THE ENGINE SUMP AND SINCE THE ENGINE IS NOT RUNNING, THE SCAVENGE PUMP CANNOT RETURN THE OIL TO THE TANK. EXCESSIVE USE OF THE FEATHERING PUMP WITH A STATIC ENGINE OVERLOADS THE SUMP.

D. Return high-pressure fuel cock lever to FUEL OFF position.

14. Details

A. Propeller Control — Electrical

All normal controls to the propeller are arranged to be fail-safe and have at least two means of actuation. Continuous indication of propeller behavior and warning of malfunction is provided the crew through a system of indicator lights and by the engine instruments. All propeller lights and switches are located at the center of the eyebrow panel for optimum availability. An electric feathering pump is provided, to assist during feathering or unfeathering. This minimizes time of action and makes available engine oil remaining in the lower regions of the tank. Even after complete drainage of the normal oil system, sufficient fluid is available in this pump to accomplish feathering.

B. Cruise Lock

This lock is controlled electrically by a solenoid in each PCU and mechanically by motion of the high-pressure fuel cock lever to its extreme forward position. Mechanical actuation affects only the engine selected, electric actuation powers the solenoids of both propellers simultaneously. This is classed as a vital circuit and all wiring is run in complete isolation. Under normal conditions, control of this circuit is through the hub switches which close their circuits when their respective blade angle is at 36 1/2° or less. During flight, as the aircraft slows down, the blade angles decrease using the hub switches to close. Since these hub switches are in series, both propellers must be at 36 1/2° or less for the circuit to be complete. Under this condition, the lock solenoids of both propellers are energized, withdrawing the locks and allowing the blades to continue to fine off as conditions warrant.

If some type of malfunction occurs after the locks are removed, the defective propeller might decrease its pitch until stopped by the next lock (FLIGHT FINE PITCH). If such a malfunction does occur, the degree of overspeed which might result is not hazardous.

Indication of blade position and warning of circuit malfunction is provided by the cruise pitch warning lights. There are two of these for reliability, wired in parallel, each individually arranged for press-to. test. The placement of these 12V lights in the circuit, with a separate resistor tapped power source, is designed to give maximum indication reliability and at the same time introduce the minimum hazard into the circuit. Operationally, the airspeeds at which the lock circuit should function is known. At or below one speed, these lights should always be on. Above another, slightly higher speed, they should always be off. Any variation denotes a malfunctioning circuit, except in a case involving a feathered propeller.

WARNING: IT IS UNSAFE TO FLY FASTER THAN 250 KNOTS (TAS) WITH THESE LIGHTS ON.

The flight safety lock switch is provided so the crew may take action if a circuit fault occurs or is suspected. Placing this switch in the EMERGENCY position removes both power and ground from the circuit and shorts both sides of the solenoids together, thus ensuring that the locks remain in place. If such switch operation is necessary, the locks must later be manually withdrawn by means of the high-pressure fuel cock lever, but only after the crew is sure airspeed is low enough that no harm results.

Due to wear and sparking problems on the hub switches, two relays have been added. The hub switches actuate only the relays, X and Z, with the relay contacts taking over the current load of the solenoids. Operation of the circuit as a whole is exactly as before addition of relays.

C. Cruise Lock out

As a second independent method of indicating the cruise lock is about to be withdrawn, a hydraulic pressure switch is fitted to the third oil line leading to the propeller hub. This switch completes a circuit turning on the light for its respective propeller. This light signals that pressure is building up in the line

and that the lock is about to withdraw. If an oil leak occurs during flight, failure of this light to come on warns that the lock is not withdrawing.

D. Flight Fine Pitch Lock Indication

This lock circuit is classed as vital and all wiring is isolated. The lock is in existence because the propeller should be above the lock setting of 20° during all flight conditions. To get maximum braking effect on the ground, it is convenient to drop the blades to zero pitch. It is also mandatory to use this setting during engine starting and during engine ground idle conditions to prevent overheating. Control of this stop is through switches located in the pedestal in the cockpit, actuated by moving flight fine pitch lock lever from FLIGHT to GROUND INTERLOCK position. Both positive and negative sides of the circuit are controlled. Solenoids of both propellers are energized simultaneously. Mechanical interlocks prevent advancing both throttles and the flight fine pitch lock lever with the gust lock positioned to ON. Indication for this lock uses the same type of tapped resistor fed twin lights used for cruise lock indication. The same reasons apply in both cases. The lights are identified FLIGHT FINE PITCH LOCK. Since this lock is used only on the ground, it is unnecessary to provide secondary means of producing actuation.

E. Below Flight Fine Pitch Lock Indication (18° hub switch features)

In order to give the crew positive indication that blade angle has reached a low setting, a hub switch is provided for each propeller which completes a circuit for blade angles of 18° or less. Separate indication for each propeller is provided. To act as an emergency governor, the circuit through this hub switch is connected to the pitch coarsening solenoid by a limit switch in the pedestal operated by the flight fine pitch lock lever. With this lever in FLIGHT position, any blade angle below the setting of this hub switch energizes the solenoid which directs engine oil pressure to move the blades to a coarser pitch (auto coarsening). Indication in either case is given by the BELOW FLIGHT FINE PITCH LOCK warning lights.

F. Feathering-Automatic and Manual

The FEATHER PUMP light, directly above each feather button on the eyebrow panel, indicates to the crew that power is being applied to the pump, whether automatically by auto feathering or manually by depressing the feather button for that engine. These lights merely indicate that the pump is receiving power. Oil pressure from the pump is applied to the system. What this pressure accomplishes depends upon whether the pitch coarsening solenoid is energized or the high-pressure fuel cock lever is positioned to FEATHER. In this case, it diverts the oil to feather. If the governor is released to constant speed operation, the propeller moves as predicated by governor control. When feathering or unfeathering is initiated electrically, the feathering pump assists in supplying oil pressure to the propeller.

Feathering is controlled by the high-pressure fuel cock lever and the pitch coarsening solenoid in the PCU. Manual feathering of the propeller simultaneously utilizes both manual linkage from the high-pressure fuel cock lever to the PCU and electrical circuits from the high-pressure fuel cock switches in the pedestal to the pitch coarsening solenoid in the PCU. Either the manual linkage or the electrical feed to the pitch coarsening solenoid (through the feather button) ports the oil to the increase side of the operating piston, causing the propeller to go toward coarse. Both are utilized in manual feathering as a safety feature. If one fails, the other accomplishes the feathering. Both the mechanical linkage and the pitch coarsening solenoid accomplish the same thing, two different ways, i.e., to restrict the governor to an overspeed mode to port the oil to the increase pitch side of the piston regardless of engine rpm. With the high-pressure fuel cock lever out of FEATHER position, the governor flyweights are released to normal constant speed operation predicated by engine rpm and speeder spring tension. The pitch coarsening solenoid in the PCU places the governor in the overspeed mode to port the oil to the increase side of the piston during auto feathering, since the high-pressure fuel cock lever is out of FEATHER when this takes place. In both manual feathering and auto feathering, the feather pump assists in supplying oil under pressure to feather the propeller. The difference is that in manual feather, the feather button is manually operated by the crew after placing the high-pressure fuel cock lever in the FEATHER position, however, in auto feathering the feather pump energizes automatically through the auto feathering circuit. Automatic feathering is dependent upon five conditions.

(1) The engine in question must have its high-pressure fuel cock lever positioned to ON. (A dead engine cannot auto feather).

(2) The other engine must also have its high-pressure fuel cock lever positioned to ON. (It is undesirable to auto feather with only one engine remaining).

(3) The throttle setting of the engine in question must be more than 11,500 rpm (approximately 30° quadrant travel). (During idle descents, torque pressure surges up and down. It is not proper to have a surge inadvertently cost the use of an engine).

(4) The other engine must not be in an auto feathering cycle (FAA regulation).

(5) With all the above conditions met, a torque sensing switch acts to trigger the sequence. Once begun, the cycle continues until either the torque builds up or until one of the conditions above is changed. The proper action is to move the high-pressure fuel cock lever to FEATHER and proceed with manual feather drill within the three minute time limit. This action replaces the electrical function with a manual function and de-energizes the pump.

G. Unfeathering

Unfeathering as far as the propeller is concerned involves releasing the governor to normal operation by positioning the high-pressure fuel cock lever from FEATHER to CRUISE LOCK OUT and depressing the feather button to give the piston oil pressure to move it toward fine. In flight, once started toward fine, the windmilling action of the airflow across the propeller blades does the rest. Since the propeller is below its minimum governing rpm of 11,000, the governor, when released from feather position of the high-pressure fuel cock, ports the oil from the pump to the decrease pitch side of the operating piston, causing the propeller to unfeather. Airstarts, naturally involve fuel and ignition. (Refer to Gulfstream Flight Manual for Airstart Procedures).

15. Emergency DC Operation — Effect On System

A. The following functions of the Propeller Control Electrical System are not available in emergency.

(1) Auto coarsening

(2) Auto feathering (manual feathering is still available).

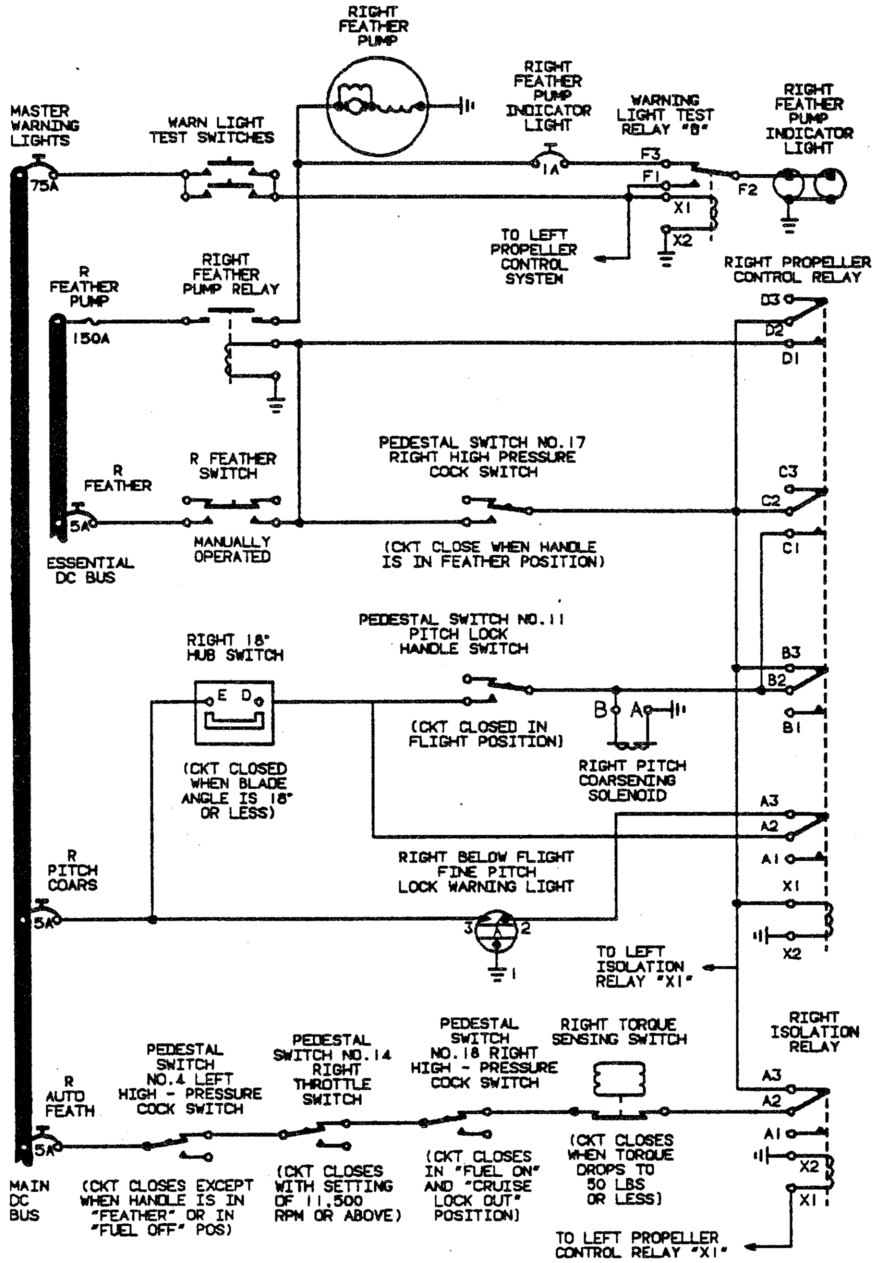
(3) Below flight fine pitch lock warning lights (2).

(4) Cruise lock out warning lights (2).

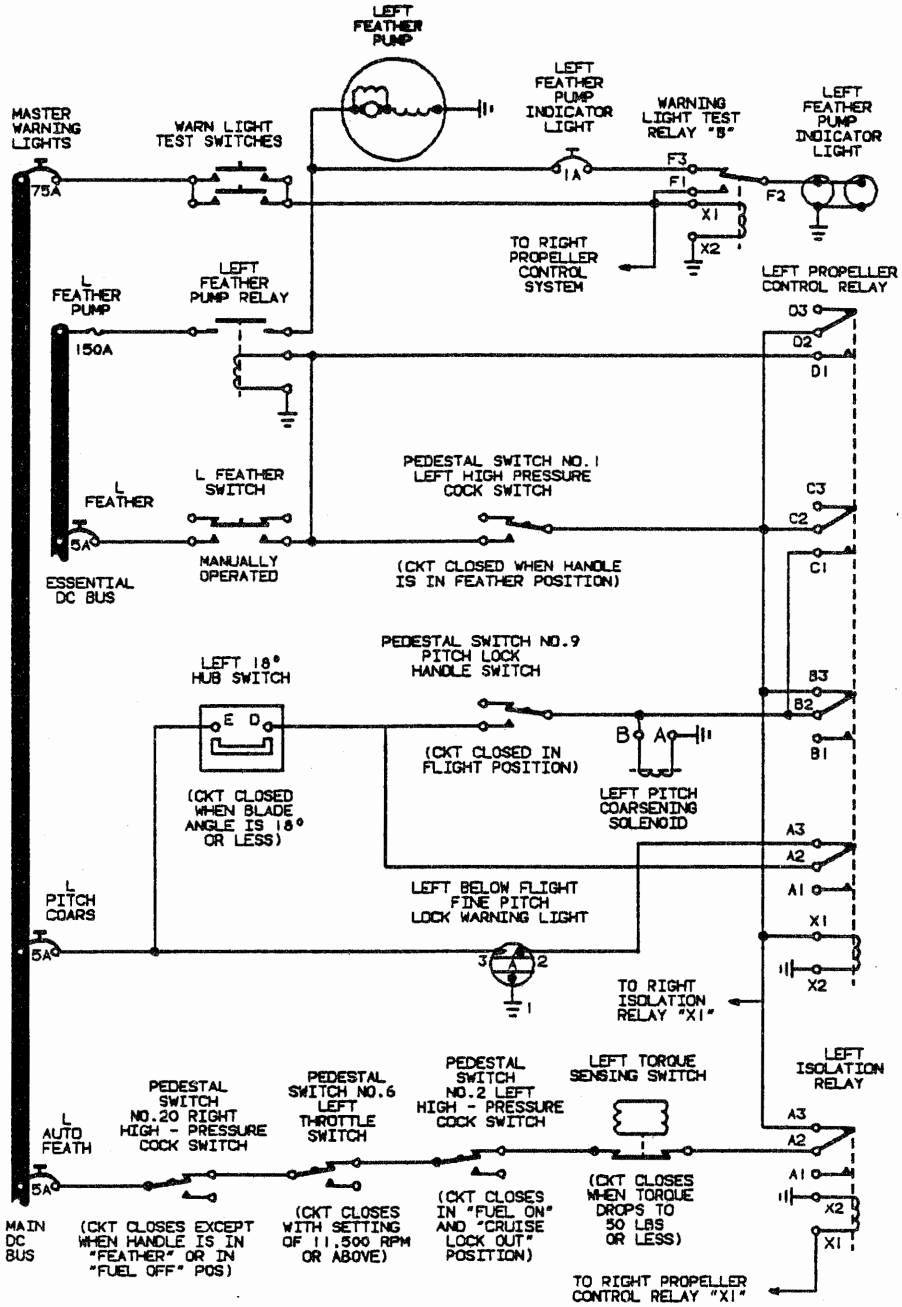
B. All other functions and indications of the propeller system remain operative except those listed above.

16. Pulse Generator Connections

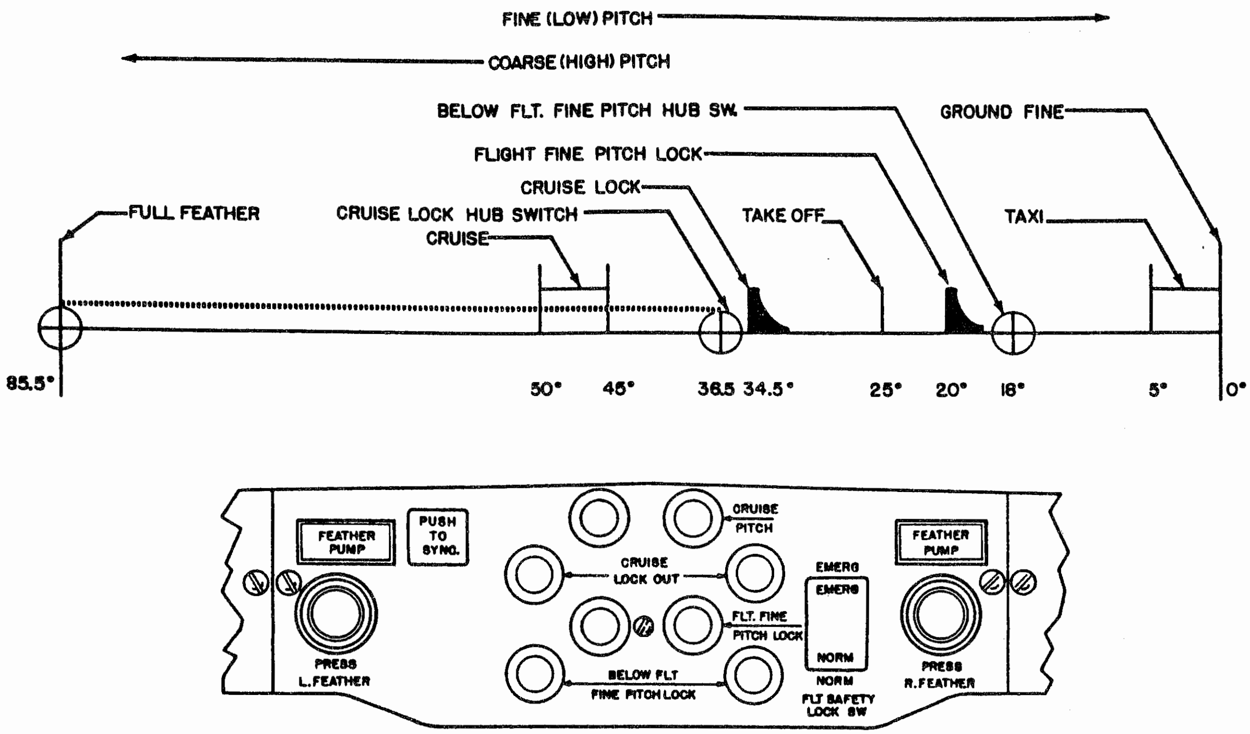
Two special receptacles, which are connected to pulse generators in each propeller, are installed in some aircraft. The receptacles are located above the hydraulic reservoir, in the hydraulic compartment, mounted on the stall warning computer bracket. They are installed in all later model aircraft and some early aircraft and are utilized only by special test equipment involved in dynamically balancing the propellers. They have no other function except for convenience of connecting this equipment.



Right Propeller Control Circuit — Schematic
Figure 1.

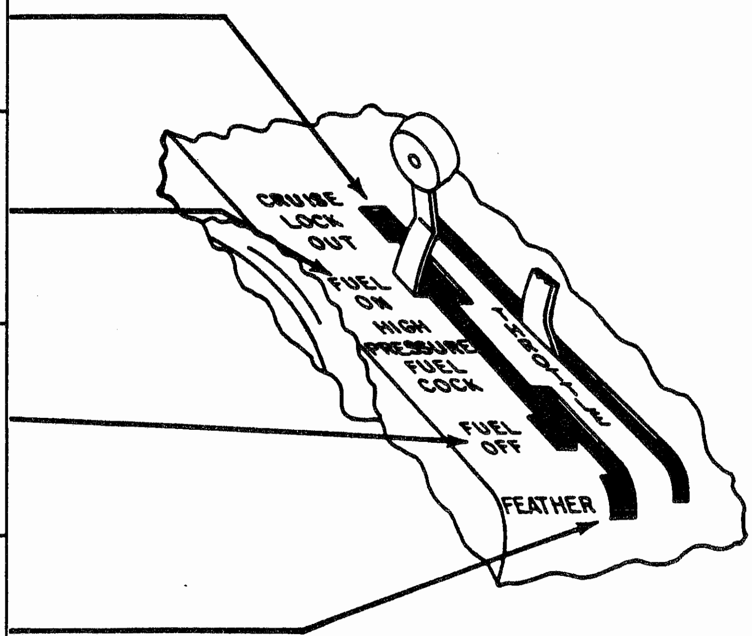


Left Propeller Control Circuit — Schematic
Figure 2.

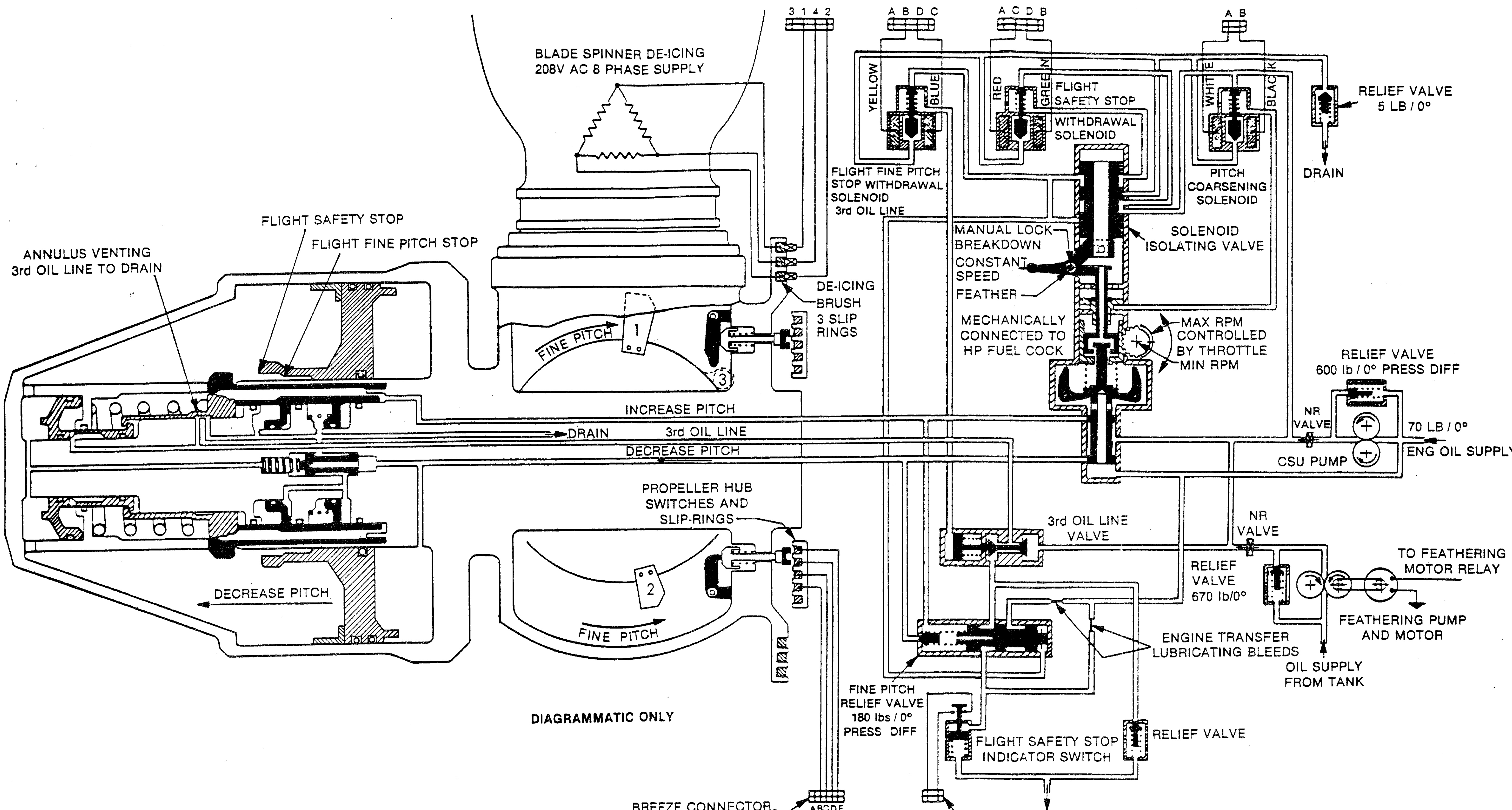


Limit of Propeller Pitch Operation and Indicator Locations
Figure 3.

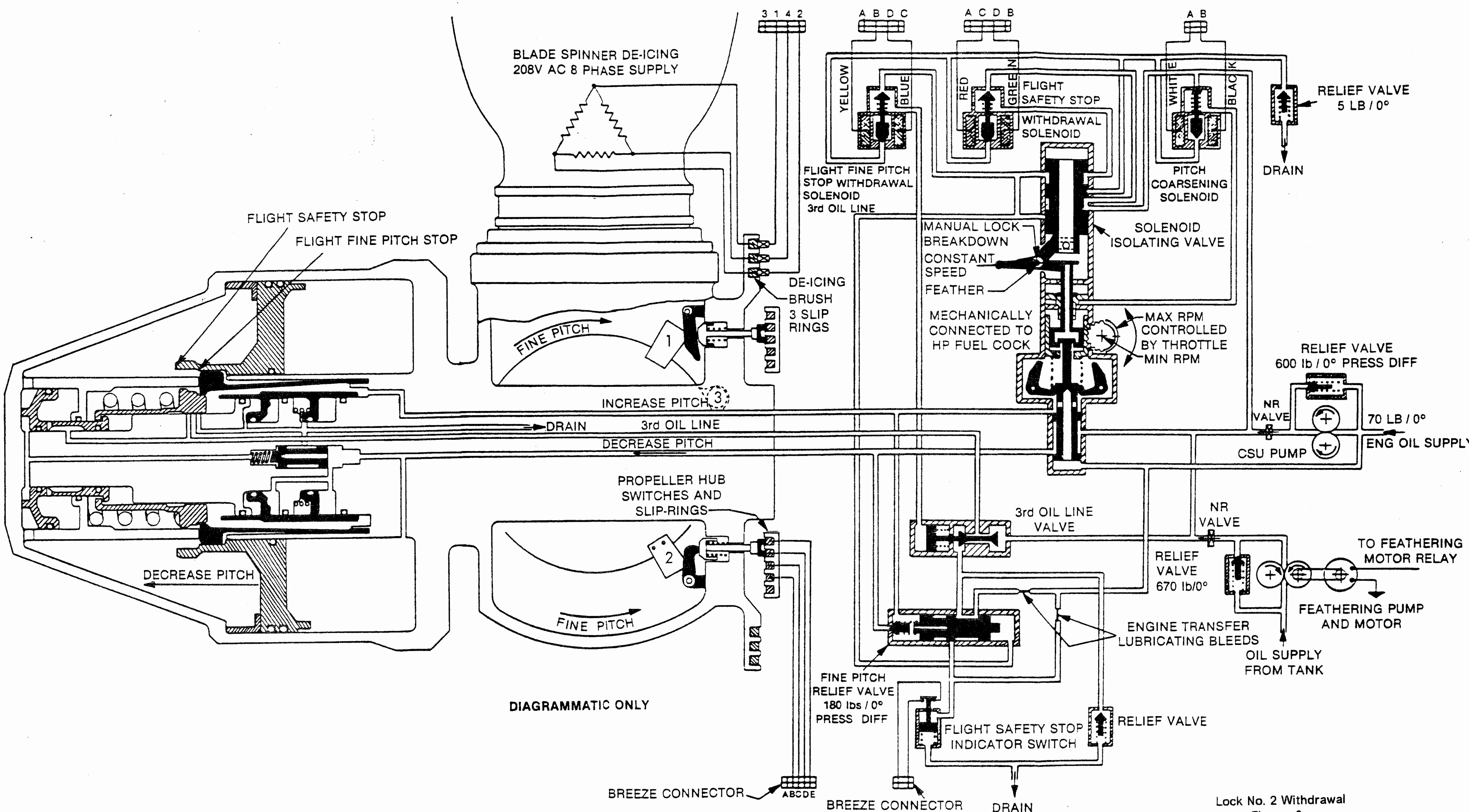
FUEL	PROPELLER
ON	CONSTANT SPEED- 34-1/2° LOCK MANUALLY REMOVED
ON	CONSTANT SPEED
OFF	CONSTANT SPEED
OFF	MANUAL FEATHER



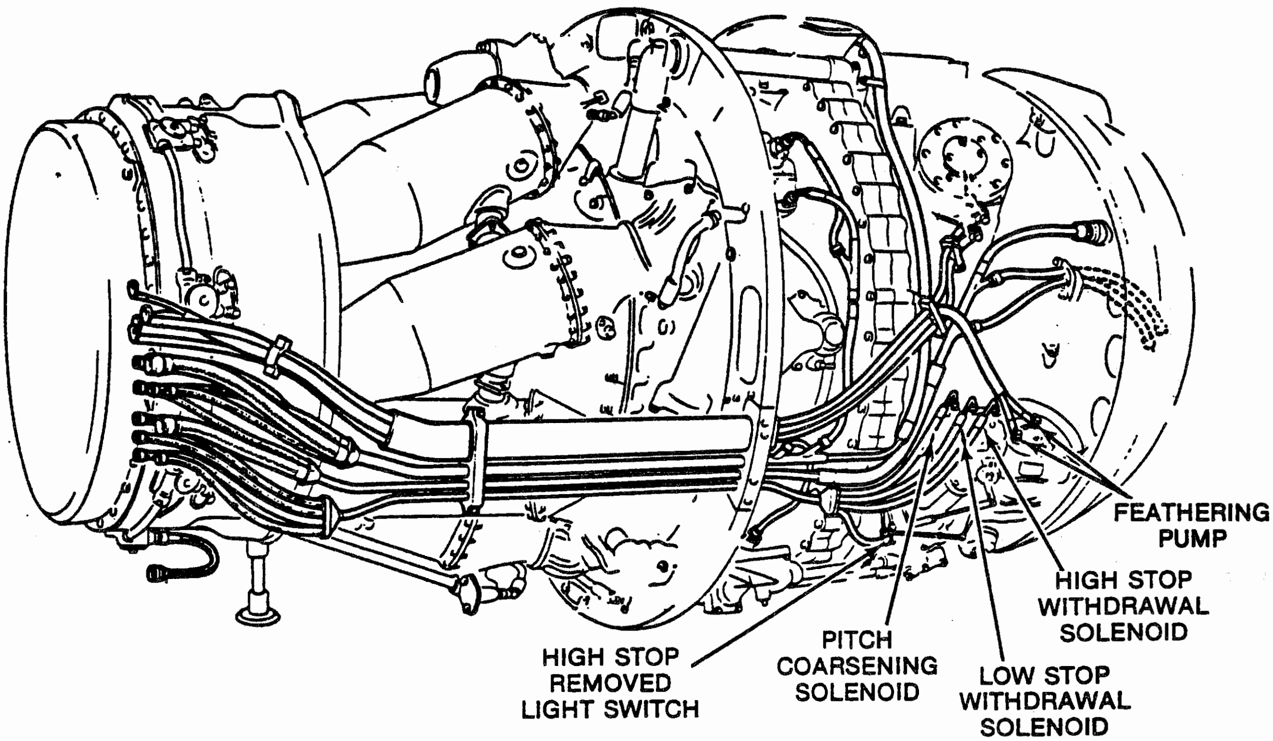
High-Pressure Fuel Cock Lever Positions
Figure 4.



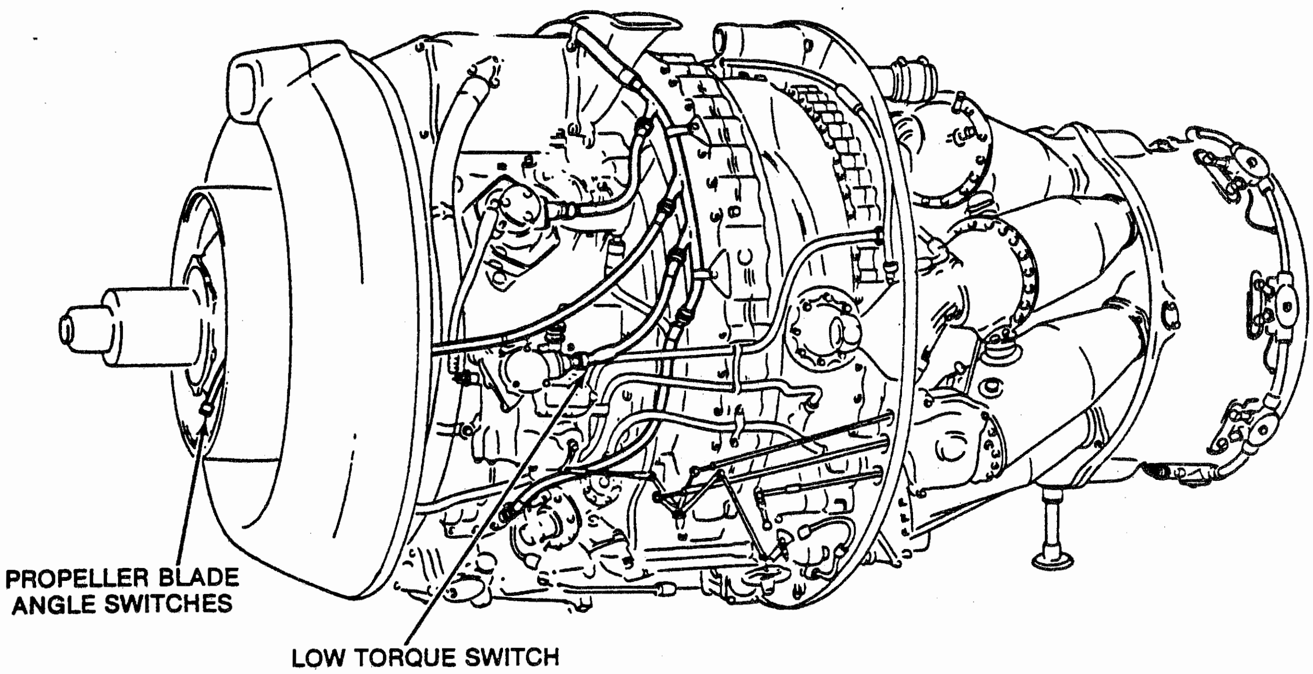
On Speed Condition
Figure 5.



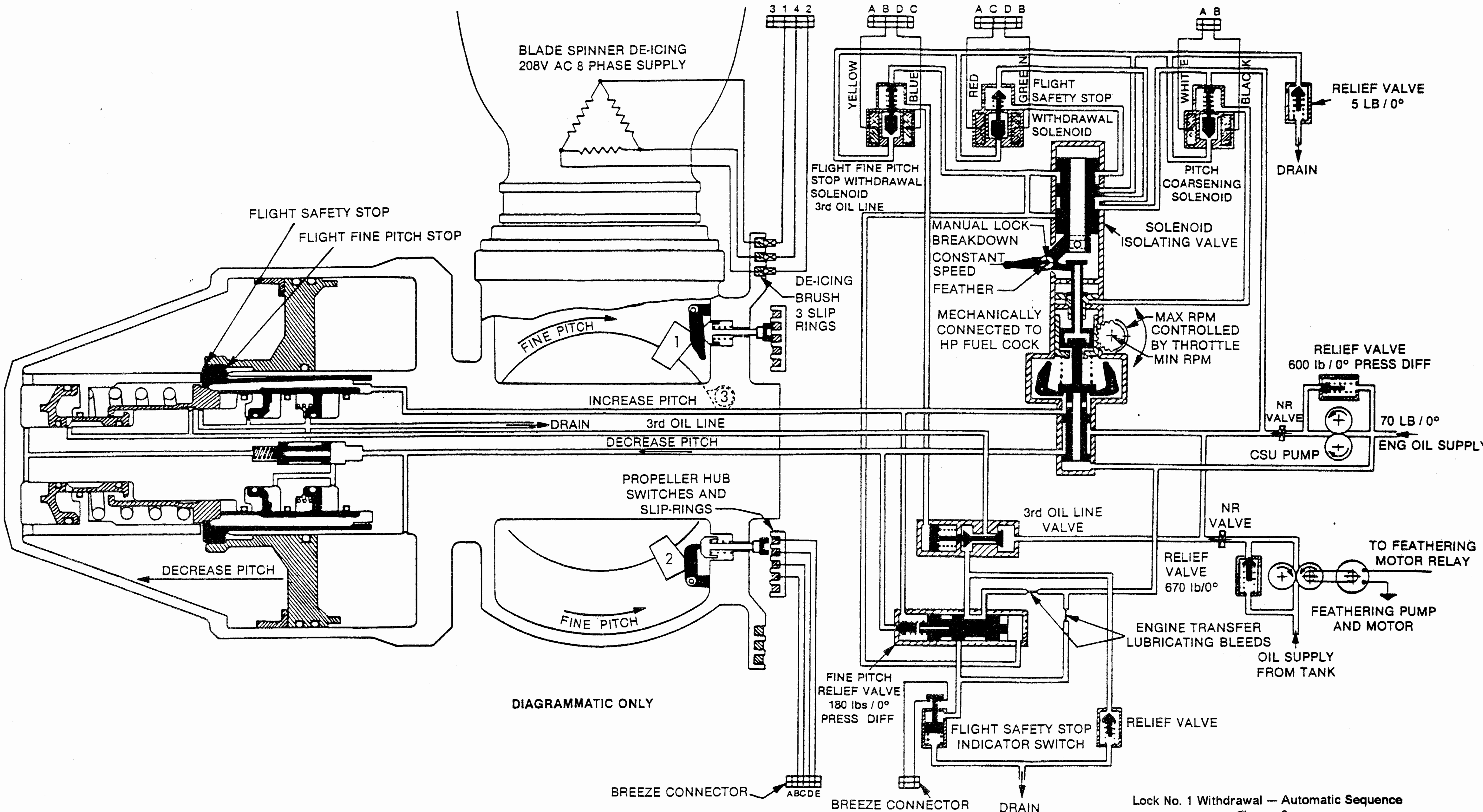
Lock No. 2 Withdrawal
Figure 6.



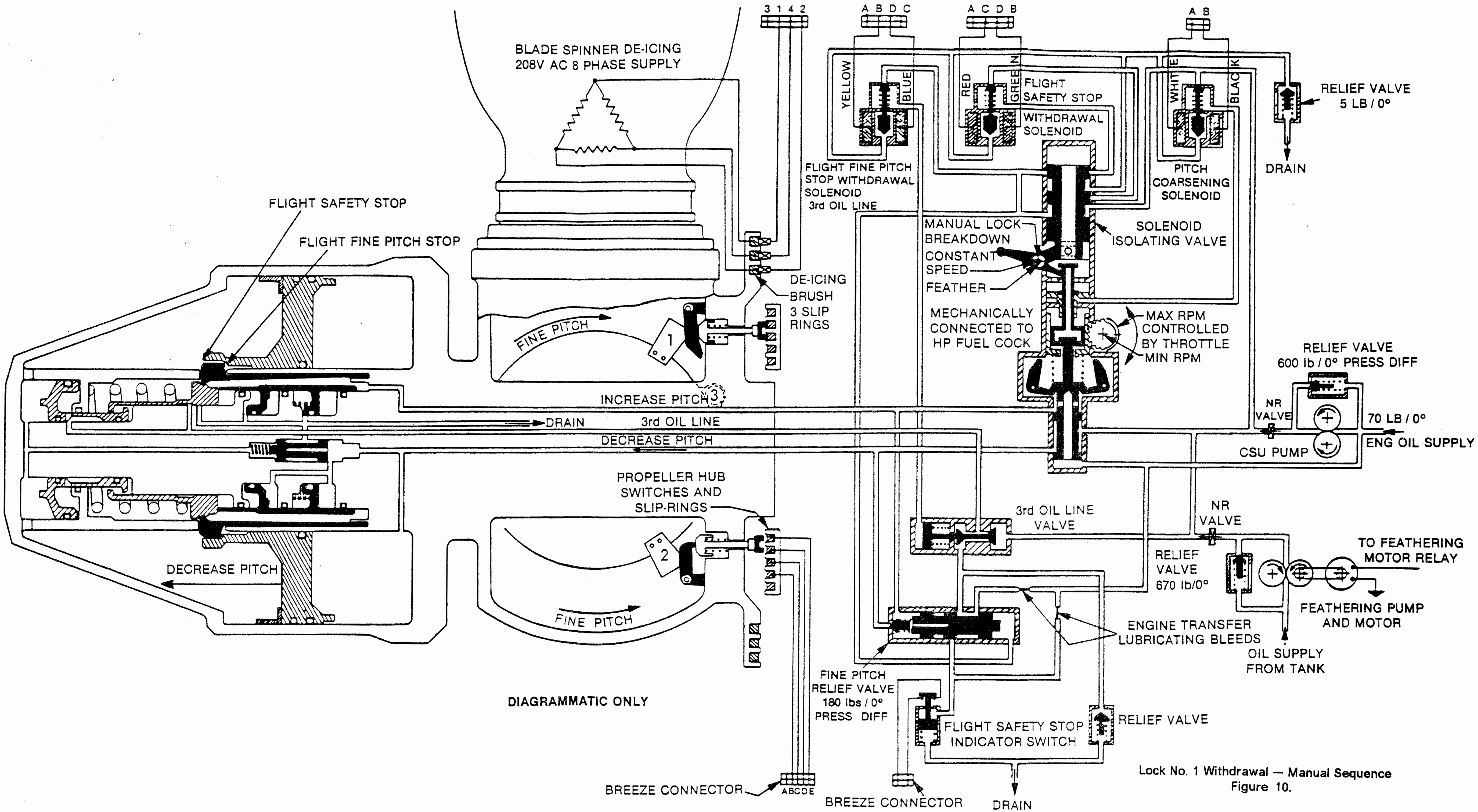
Propeller System Electrical Connections — Right Side Of Engine
Figure 7.



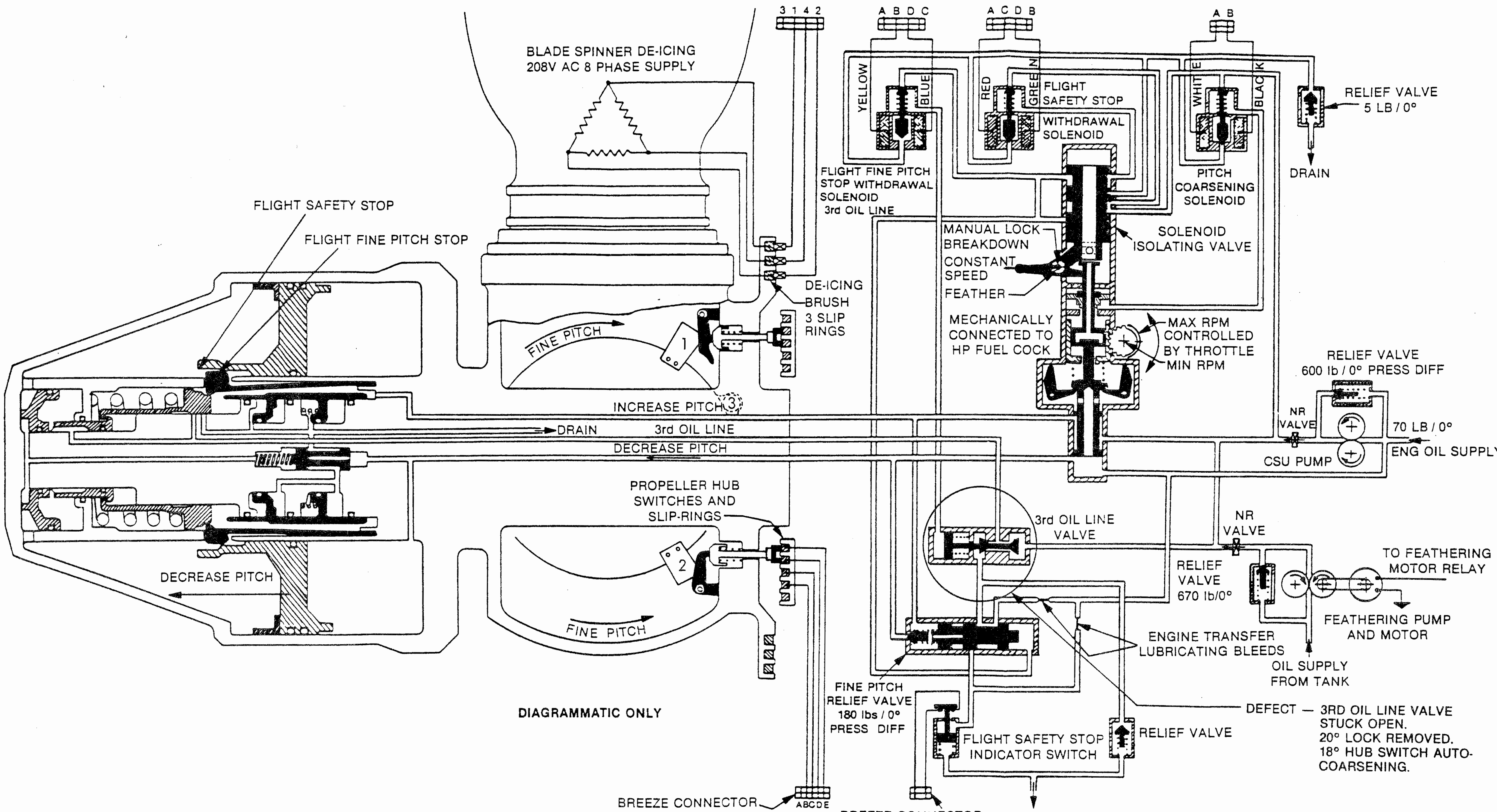
Propeller System Electrical Connections — Left Side Of Engine
Figure 8.



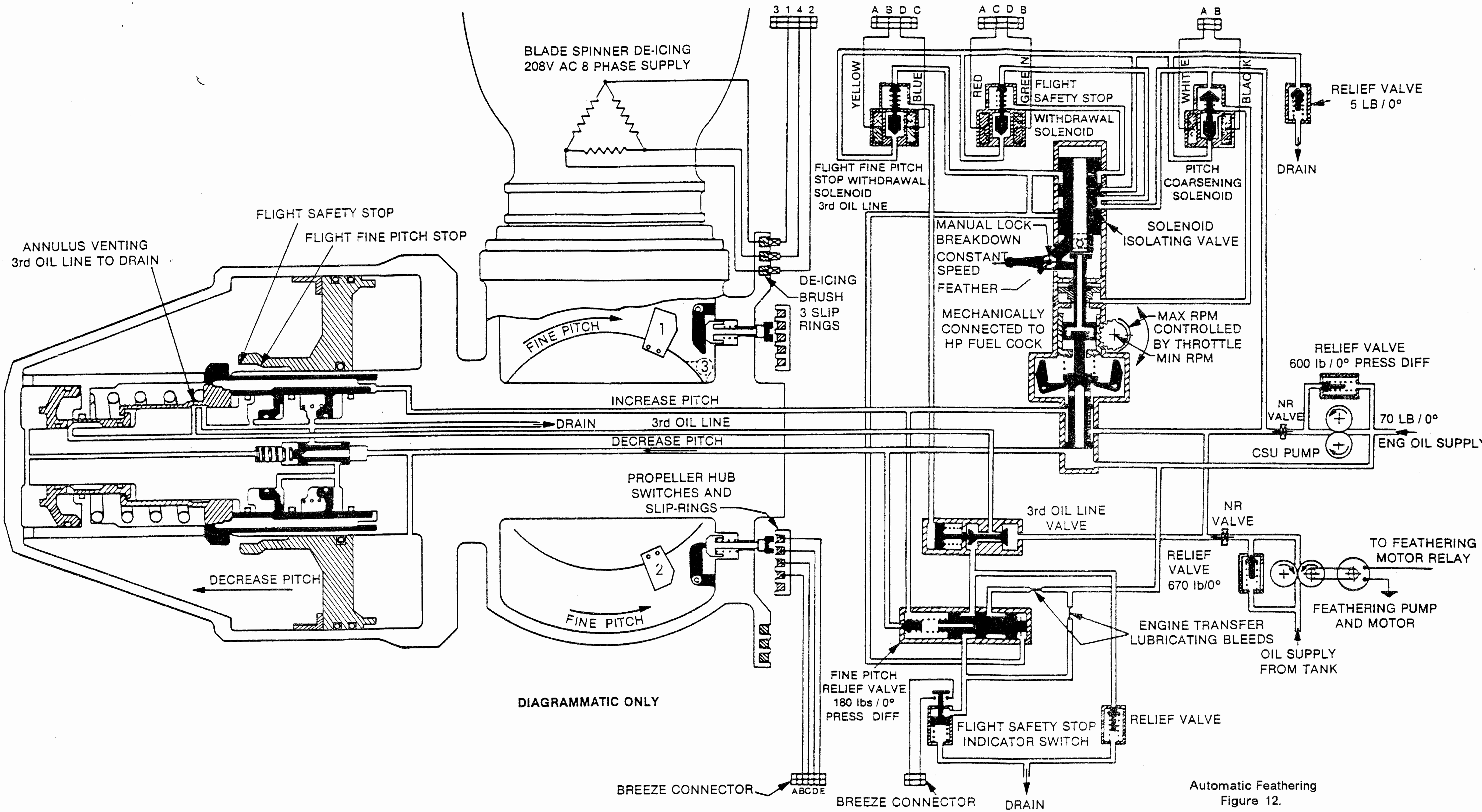
Lock No. 1 Withdrawal — Automatic Sequence
Figure 9.



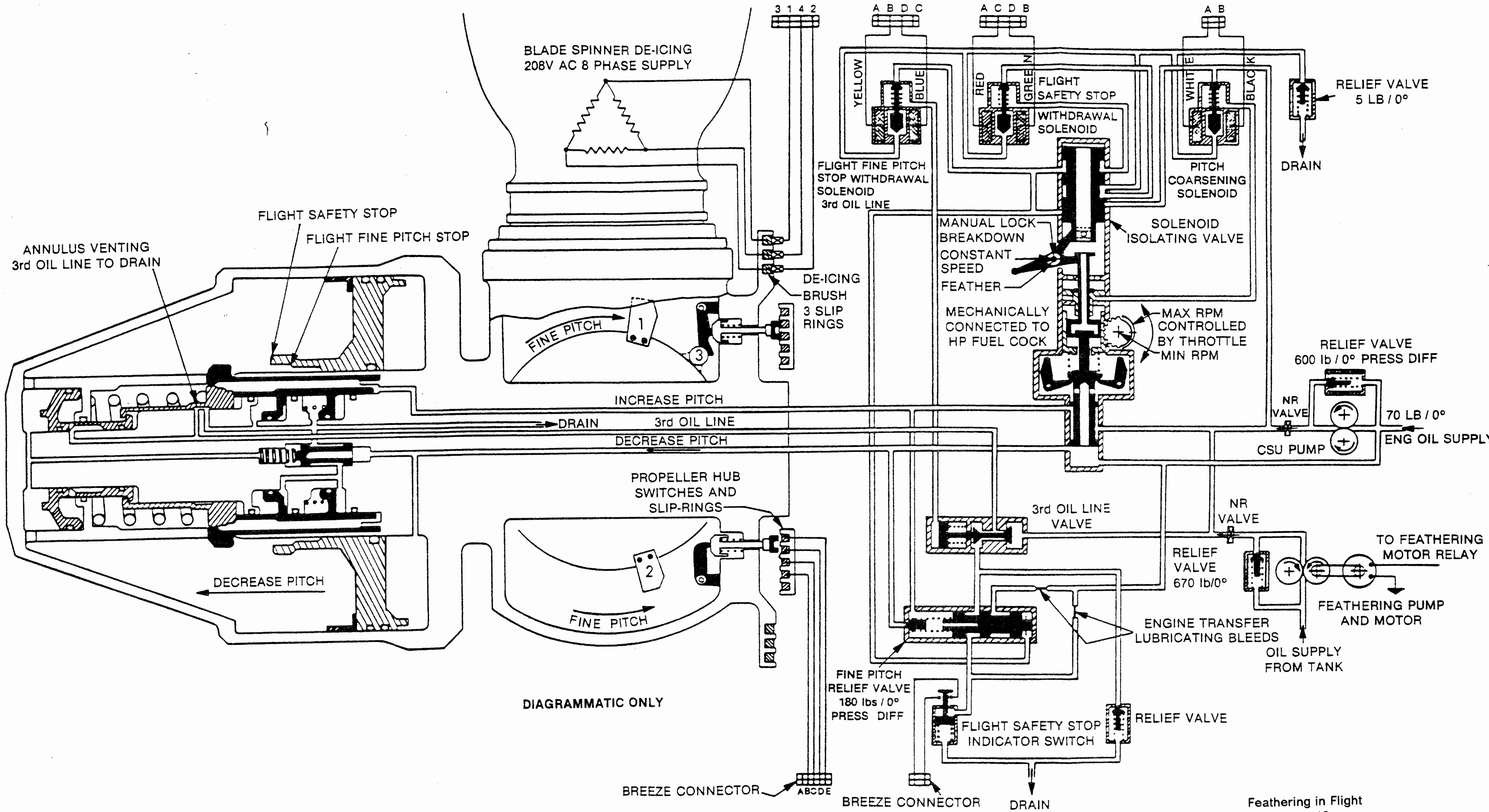
Lock No. 1 Withdrawal — Manual Sequence
Figure 10.



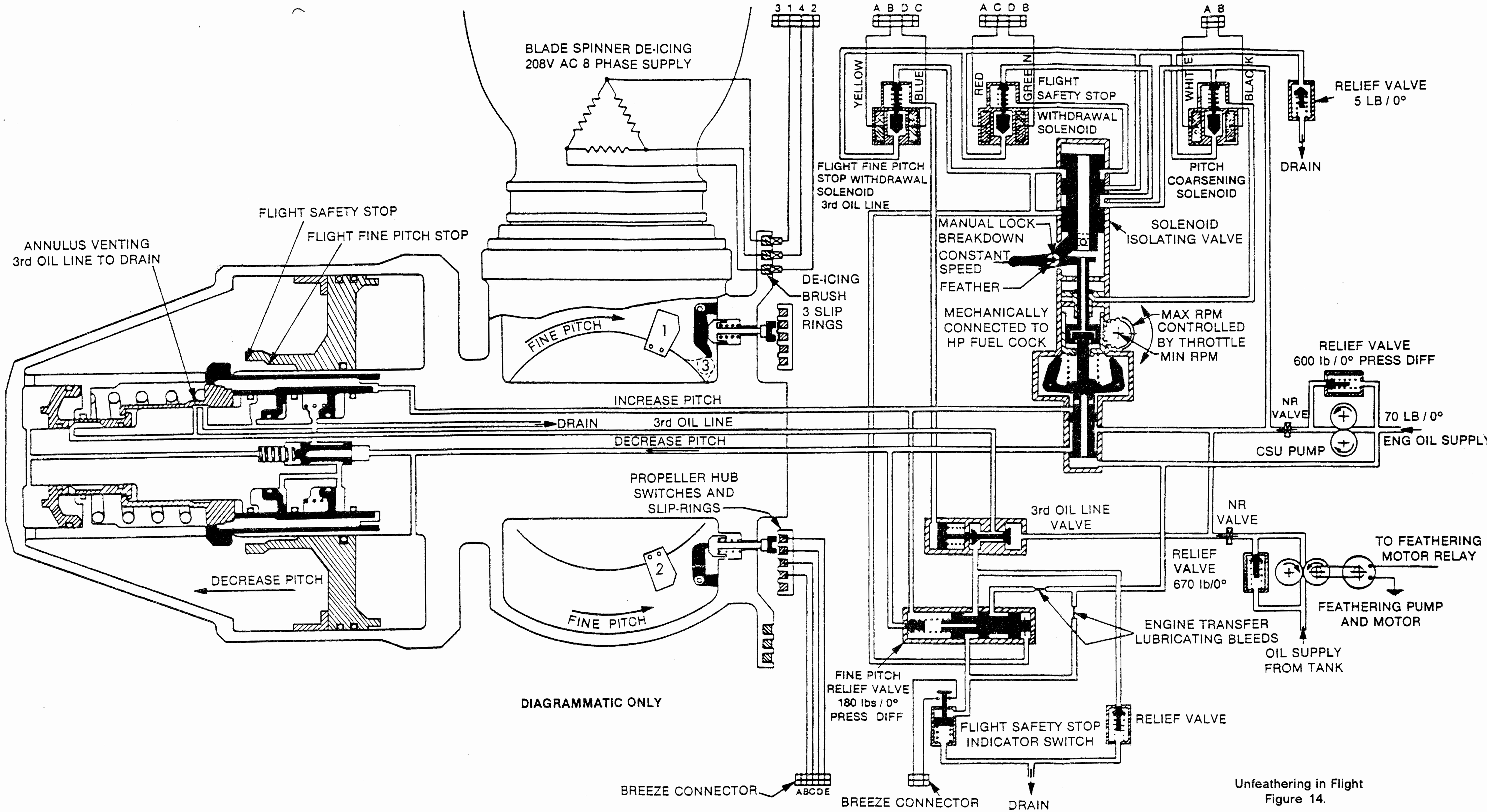
Automatic Coarsening — Hub Switch Operation
(Defect — Third Oil Line Valve Open)
Figure 11.



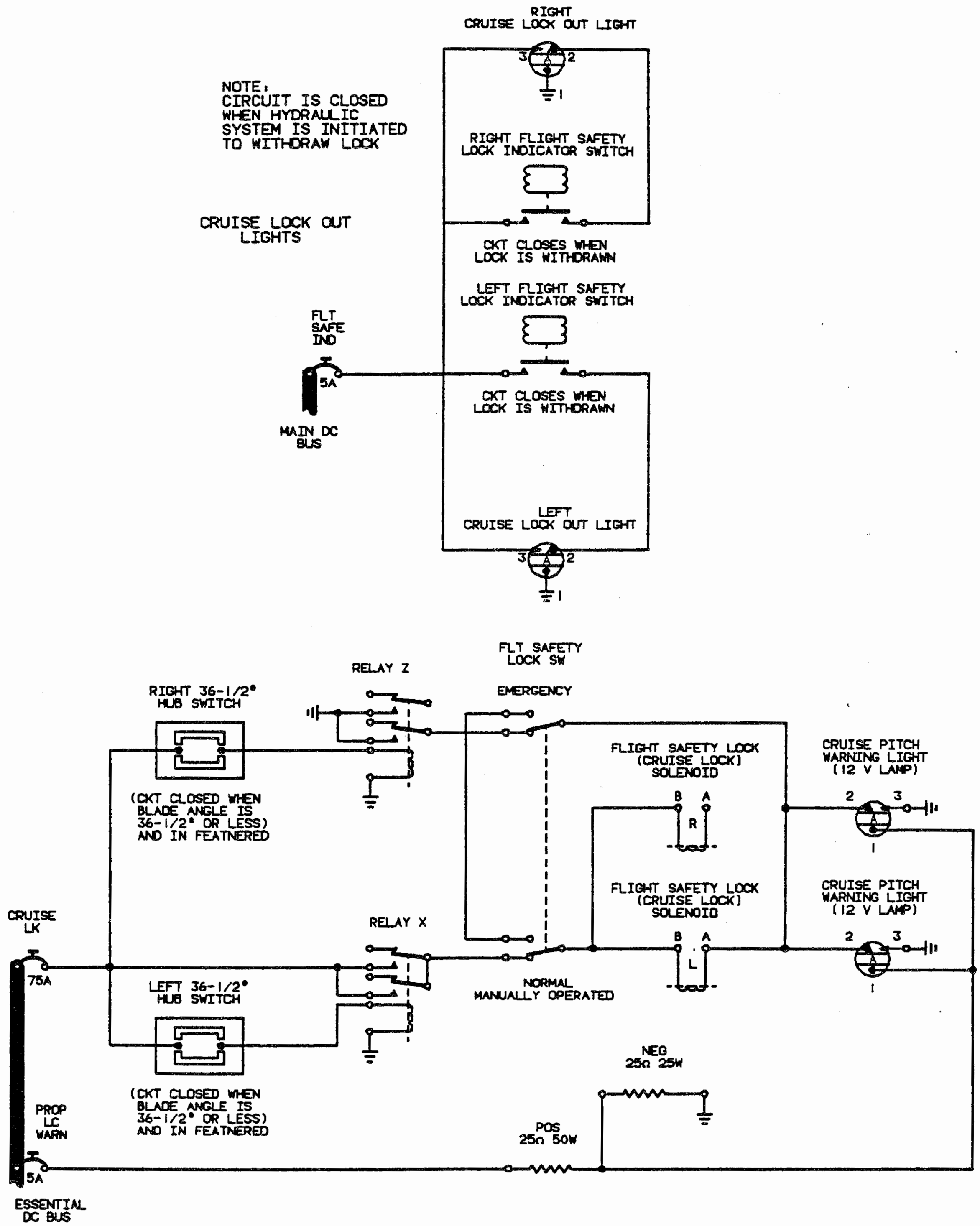
Automatic Feathering
Figure 12.



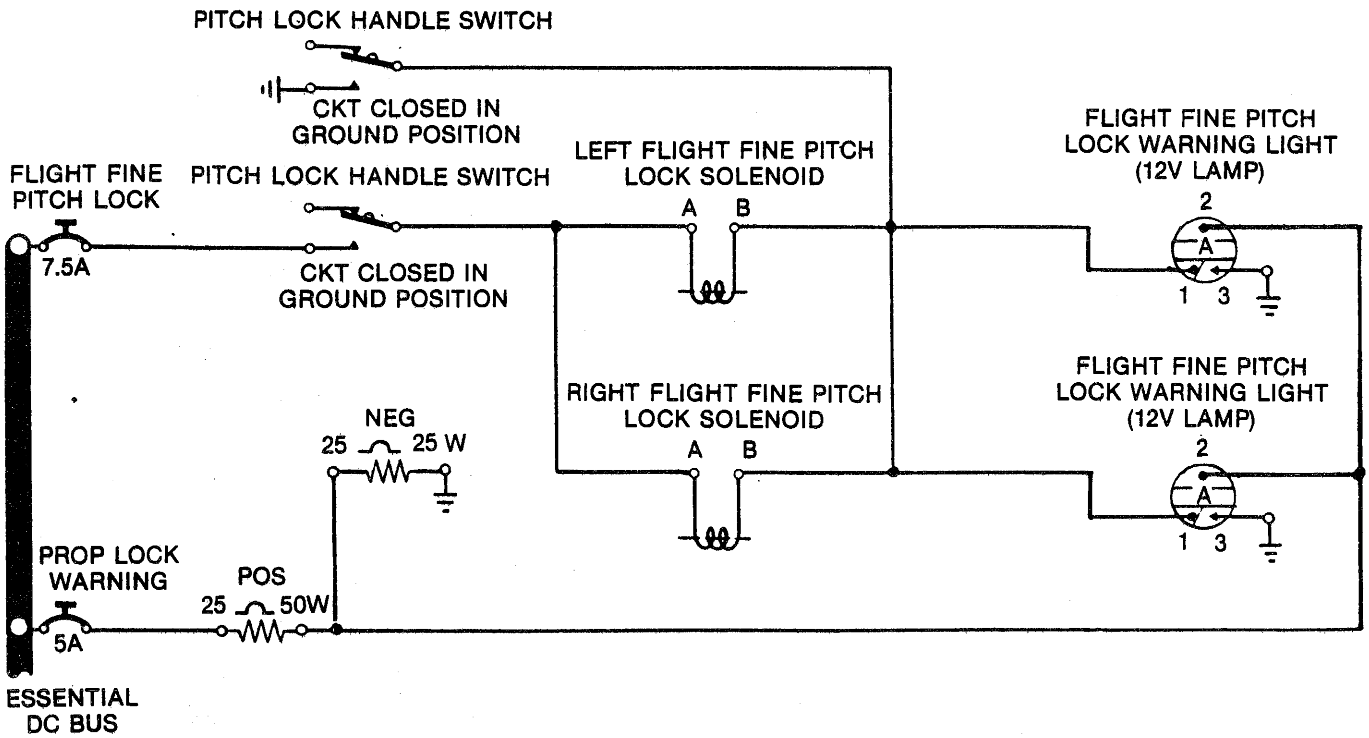
Feathering in Flight
Figure 13.



Unfeathering in Flight
Figure 14.



Cruise Lock and Cruise Lock Warning Lights
Figure 15.



Flight Fine Pitch Lock and Flight Fine Pitch Lock Warning Lights
Figure 16.

PROPELLERS — FAULT ISOLATION

NOTE: Fault isolating of a propeller system in some cases becomes complex, as it involves the engine, propeller and the airframe systems which are all interrelated. In many cases fault isolating can be accomplished with the aid of information contained in the Dowty Rotol Propeller Manual and the Rolls-Royce Dart Maintenance Manual. In other cases it requires information from both of the above sources, plus a detailed knowledge of the aircraft systems which are involved. As a possible aid in fault isolating two specific conditions involving the cruise lock automatic removal system, the following procedures are suggested to illustrate a simplified method of coping with them. As in any fault isolating, the suggested method cannot be construed as the only possible course of action, nor can the possible cause and rectification be considered a 100% cure-all for all cruise lock problems. The data presented herein, along with the fault isolating charts are not to be construed as an indication of a continual problem, but should be considered as a composite of actual and theoretical information which may aid the operator in analyzing the situation should a problem of this sort arise.

1. **Single Cruise Lock Withdrawal Failure**

   A. Problem

   One propeller goes through its cruise lock, opposite propeller does not. This problem is based on the probability that the defective propeller will go through its cruise lock in manual cruise lock out.

   B. Probable Causes

   (1) Open coil in cruise lock solenoid.

   (2) No power to cruise lock solenoid.

   (3) No ground for cruise lock solenoid.

   (4) Internal malfunction in propeller control unit.

   C. Preliminary Fault Isolating Procedures

   (1) Both propellers at 0° blade angle.

   (2) External power on aircraft.

   (3) Battery switch in normal.

   (4) Flight safety lock switch in normal.

   (5) High-pressure cocks in fwd OFF position.

   (6) Pull PROP LK WARN circuit breaker (on copilots circuit breaker panel).

   (7) Ensure CRUISE LK circuit breaker is in (on copilots circuit breaker panel).

   (8) Open cowling on engine with malfunctioning propeller.

   D. Perform Audible Check

   (1) Have helper in cockpit alternately move flight safety lock switch from NORMAL to EMERGENCY several times.

   (2) While Step (1) above is being performed, hold hand on cruise lock solenoid and listen for clicks at most forward electrical connection on the propeller control unit. See Figure 101 for Detailed Fault Isolation.

2. **Dual Cruise Lock Withdrawal Failure**

   A. Problem

   Neither propeller will fine below the 34.5° locks in automatic, but will go through in manual cruise lock out.

   B. Probable Causes

   (1) Hub Switches not making or set too low.

(2) Relay X or Z inoperative. With this in mind for the purposes of fault isolating this problem, we will discount wiring problems.

C. Preliminary Fault Isolating Procedures

(1) Install oil bypass equipment on both engines.

(2) External power on aircraft.

(3) Battery switch in normal.

(4) Flight safety lock switch in EMERGENCY.

(5) High-pressure fuel Cocks in FEATHER.

(6) Both throttle levers retarded.

(7) Depress both feather pump buttons until both propellers are above the cruise locks and release buttons.

(8) Both high-pressure fuel cocks to forward ON.

(9) Depress both feather pump buttons until both propellers hang on cruise locks and release buttons.

(10) Pull PROP LK WARN circuit breaker (on copilots circuit breaker panel).

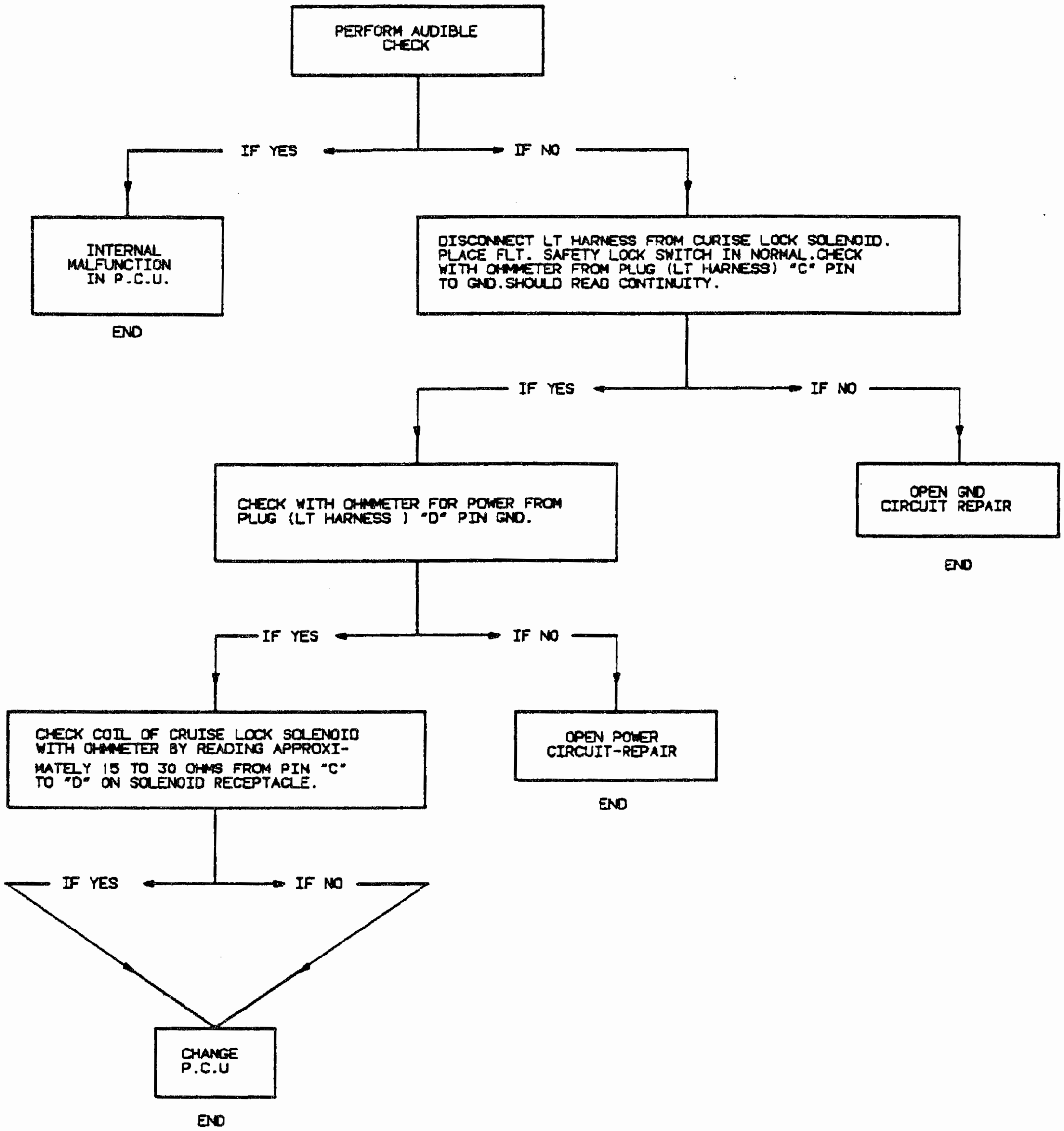
(11) Ensure CRUISE LK circuit breaker is in (on copilots circuit breaker panel).

(12) Place flight safety lock switch in NORMAL.

(13) Open either cowling and disconnect left harness from cruise lock solenoid at the propeller control unit. (Most forward electrical connector on the propeller control unit).

(14) With ohmmeter check continuity from pin C to ground of left harness plug.

D. Detailed Fault Isolation (See Figure 102)



Single Cruise Lock Withdrawal Failure — Fault Isolation Chart
Figure 101.

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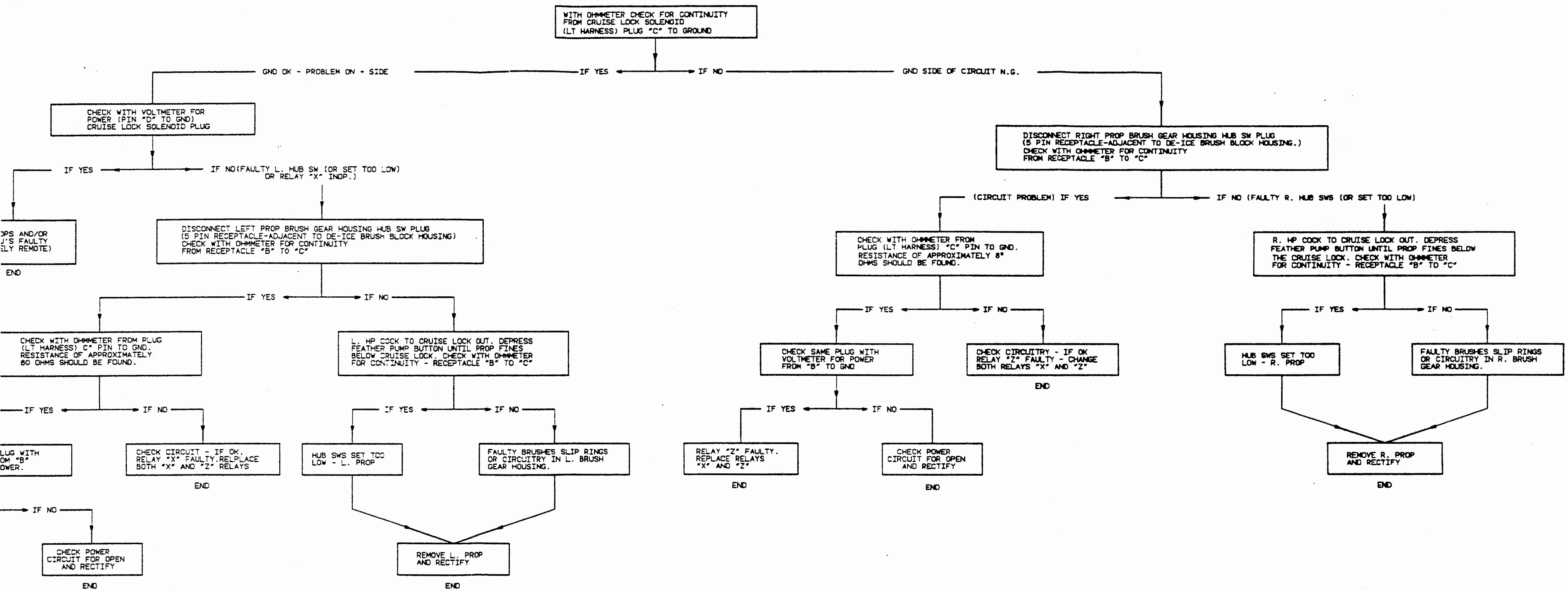


Dual Cruise Lock Withdrawal Failure — Fault Isolation Chart
Figure 102.

PROPELLERS — MAINTENANCE PRACTICES

1. **General**

NOTE: For applicable Maintenance Practices not covered in this chapter, refer to the following manuals:

Dowty Rotol Ltd. Maintenance Manual No. 865/1 - Propeller Equipment, Gulfstream/Dart 529.

Dowty Rotol Ltd. Inspection Schedule 35/58 - Dowty Rotol Propellers (as revised).

2. **Propeller — Inspection**

A. Clean propeller blades and inspect for cracks, dents, burns and corrosion.

B. Inspect deicing boots and root band for lifting, cuts and blistering.

C. Inspect screening plates for security, locking and signs of damage.

D. Inspect external rubber components and visual cable from blades to slip ring for serviceability.

E. Ensure security of cable connections at slip ring housing terminals.

F. Inspect propeller for leakage.

3. **Propeller — Removal / Installation**

A. Removal

(1) Ensure electrical power is OFF.

(2) Remove propeller spinner.

(a) Ensure propeller blades are at an angle approximately midway between flight fine pitch and feather to enable spinner to be removed without fouling.

CAUTION: DO NOT ATTEMPT TO TURN ANY SINGLE LOCKPIN TO FULLY UNLOCKED POSITION IN ONE MOVEMENT AS DAMAGE MAY RESULT.

(b) Supporting spinner and using screwdriver in slots of four lockpins, turn each lockpin towards unlocked position as far as it will turn without undue force being used, work progressively around spinner from lockpin to lockpin, turning each a little at a time until all four are at unlocked position.

(c) Support and withdraw spinner forward until clear of blade and hub.

NOTE: To prevent propeller rotation while releasing hub retaining nut, a blade must be supported at 0.7 radius station, preferably with blade at feathered angle. At any other angle however, blade stress will still be acceptable.

(3) Remove cylinder cover and pitch lock:

(a) Ensure propeller blades are feathered.

(b) Remove retaining ring from groove in serrations of cylinder cover nut and remove locking segment.

(c) Remove retaining ring from groove in serration of cylinder cover.

(d) Using spanner TL3745, unscrew and remove cylinder cover nut, LH thread. Tap spanner with a rawhide mallet to release initial tightness.

NOTE: Place a suitable container beneath propeller to catch oil spillage when propeller is being removed.

It is important that rubbing ring is removed independently to avoid any possibility of damage to rubbing ring cogs and cylinder threads.

(e) Withdraw rubbing ring. Two pieces of wire inserted under rubbing ring will assist removal or two small magnets can be used.

(f) Screw cylinder nut into cylinder a sufficient distance to allow retaining ring to be refitted to cylinder cover.

(g) Fit retaining ring in groove in serrations of cylinder cover.

(h) Using spanner TL3745, unscrew cylinder nut. As nut is unscrewed, cylinder cover and pitch lock assembly will be partially withdrawn from cylinder.

(i) On completion of disengagement of nut with cylinder thread, remove spanner. Fit tool TL3913 to nose of cylinder cover and while supporting cylinder cover, ease out cylinder cover and pitch lock assembly.

(4) Remove hub retaining nut and front cone:

(a) Fit propeller lifting tackle TL3916 and take up weight of propeller.

(b) Screw support TL3744 into mouth of cylinder (LH thread) and tighten with tommy bar.

(c) Insert spanner TL4458 through support and engage it with hub retaining nut serration.

(d) Fit 1 1/4 inch Whitworth adapter to hexagon end of spanner and using Acratork spanner B 7/1 unscrew hub retaining nut (RH thread).

(e) Unscrew and remove support, complete with spanner and hub retaining nut and front cone assembly.

(5) Remove propeller from engine shaft:

(a) Ensure weight of propeller is taken on lifting tackle.

(b) Screw support TL3744 into mouth of cylinder (LH thread) and tighten with tommy bar.

(c) Insert extractor TL3615 through bore of support, engaging thread of extractor with threaded bore of support (LH thread).

(d) Screw extractor by hand until end face abuts end of engine shaft.

(e) Using spanner TL3094 on hexagonal end of extractor continue to turn until propeller is free of shaft splines.

CAUTION: CARE MUST BE TAKEN TO AVOID DAMAGE TO ENGINE SHAFT THREADS AND PROPELLER OIL TUBES.

(f) Remove tools and withdraw propeller clear of oil tubes.

(g) Lower propeller onto a suitable stand and remove lifting tackle.

(h) If hub rear oil seal has remained on engine shaft, remove seal and seal rings and attach these to propeller.

NOTE: If propeller is being removed for overhaul or replacement, oil tubes should be removed from engine shaft and treated as part of propeller.

(6) If required, remove oil tubes from engine shaft as follows:

(a) Remove circlip retaining lockpiece in outer oil tube. Remove lockpiece.

(b) Fit tool TL3925 into outer oil tube serrations to provide a hexagon for spanning and unscrew and remove outer oil tube (RH thread).

(c) Fit lockpiece into inner oil tube serrations to provide a hexagon for spanning and unscrew and remove inner oil tube (LH thread).

(d) Remove engine shaft holding tool.

B. Installation

NOTE: When mounting propeller, check that when it is rotated, keepers on slip ring housing will not foul pulse generator pole pieces. If necessary, temporarily adjust pulse generator to a rearward position.

(1) Install propeller for check of cones surface contact and hub switch tracking check.

(a) Make four marks with chalk, equally spaced, across all slip rings of brush gear housing.

(b) Lightly coat front and rear cone seatings of hub driving center with marking glue.

(c) Lightly coat engine shaft, at rear cone mounting position with engine oil.

NOTE: Ensure cone surface is clean and free from oil and grease.

(d) Fit rear cone in position on engine shaft.

(e) Remove rear sandwich seal from hub driving center to avoid possibility of damage to seal.

(f) Fit lifting tackle TL3916 to propeller and mount propeller on engine shaft. Ensure that blades are in feathered position.

NOTE: Ensure cone surface is clean and free from oil and grease.

(g) Fit hub retaining nut and front cone assembly using spanner TL4458. Tighten by hand only.

(h) Thread and tighten support TL3744 into mouth of cylinder to guide and support spanner. Tighten hub retaining nut using spanner TL3094 on hexagon end of TL4458.

(i) Remove lifting tackle.

CAUTION: WHEN TURNING BLADES BY HAND, ALL FOUR BLADES MUST BE HELD WHERE BLADES ARE OF SUBSTANTIAL THICKNESS AND MUST BE TURNED TOGETHER. CARE MUST BE TAKEN TO AVOID DAMAGE TO DE-ICING EQUIPMENT.

(j) Turn propeller blades to approximately ground fine pitch position.

(k) Rotate propeller at least one complete revolution.

(l) Return blades to feathered position.

(m) Fit lifting tackle and take weight off propeller.

(n) Remove hub retaining nut and front cone assembly using support TL3744 and spanner TL4458 and TL3094.

(o) Withdraw propeller from engine shaft.

(p) Remove rear cone from engine shaft.

(q) Check blueing on cone surface. Contact area must not be less than 80% of whole surface and must be uniformly distributed. High spots on cone seating should be removed by light stoning.

(r) Remove marking blue from cones and seatings.

(s) Check brush gear housing slip rings for correct tracking of hub contact switch.

(t) Remove all traces of chalk from brush gear housing.

(2) If removed, fit oil tubes in engine shaft.

(a) Fit engine shaft holding tube TL3759. This is engaged with engine shaft splines and held by hand when tightening oil tubes.

(b) Lubricate threads and oil seals of oil tubes with clean engine oil.

(c) Screw inner oil tube by hand (LH thread) into oil transfer plug in engine shaft bore.

(d) Engage serrations of oil tube lockpiece into oil tube serrations to provide a hexagon for spanning.

NOTE: Before tightening, check that spanner is set to give correct torque

(e) Using Acratork spanner T14036 and socket T14013, torque oil tube to 20 foot-pounds, withdraw lockpiece.

(f) Pass outer oil tube over inner oil tube and thread it by hand (RH thread) into oil transfer plug. Take care to avoid damage to oil seals.

(g) Engage serrations of adapter TL3925 into outer oil tube serrations. Using Acratork spanner and socket, torque outer oil tube to 20 foot-pounds. Remove adapter.

CAUTION: DO NOT UNSCREW OIL TUBES TO AFFECT A LOCKING POSITION.

NOTE: Careful sighting of serrations will almost always ensure that vernier arrangement of serrations will enable locking to be carried out. Should serrations not engage, very slightly tighten outer oil tube by approximately one-half a serrations spacing.

(h) Lock oil tubes by engaging serrations of lockpiece into both oil tubes.

(i) Fit circlip into groove in outer oil tube serrations to retain lockpiece. If difficulty is experienced in fitting circlip, check that lockpiece is fully home.

(j) Remove engine shaft holding tool.

CAUTION: ENSURE HUB SWITCHES ARE INSTALLED PROPERLY.

(3) Mount propeller on engine shaft.

(a) Lightly coat engine shaft at rear cone mounting position with engine oil.

(b) Lightly coat seating surface of rear cone with grease and position cone on engine shaft.

(c) Position hub rear oil seal, sandwiched between two seal rings, on engine shaft in front of, and abutting rear cone.

(d) Protect engine shaft thread and ends of oil tubes with suitable protectors.

(e) Fit lifting tackle of TL3916 and mount propeller on engine shaft first checking that split distance sleeve is fitted in hub rear bore, bevelled end to front.

(f) Remove protectors from engine shaft thread and oil tubes.

(4) Tighten propeller

CAUTION: GREAT CARE MUST BE TAKEN TO AVOID DAMAGE TO CYLINDER AND COLLET SLEEVE BORES. DAMAGE TO BORES, ESPECIALLY TO LEADS FOR OIL SEALS CAN RESULT IN DAMAGE TO PITCH LOCK OIL SEALS ON ASSEMBLY TO PROPELLER.

(a) Ensure blades are in feathered position.

(b) Lightly smear front cone seating surface with grease and place hub retaining nut and front cone assembly on serrated end of spanner TL4458.

(c) Using spanner as an extension piece, pass assembly through center of cylinder and collet sleeve and engage nut thread with shaft (RH thread).

(d) Tighten nut as far as possible by hand.

(e) Thread and tighten support TL3744 into mouth of cylinder to act as a guide and support for spanner.

(f) Using spanner TL3094 on hexagon end of TL4458, tighten hub retaining nut as far as possible by hand, then loosen off half a turn and tighten with four sharp blows using a 4 pound rawhide mallet. Remove spanner TL3094.

NOTE: Before tightening, check that Acratork spanner is set to give correct torque. Whenever a hub retaining nut has been overtightened, a dimensional check of cone must be carried out with particular reference to bottom face of counter bore. If check is satisfactory cone may be returned to service.

(g) Using Acratork spanner No. B 7/1 together with its 1 1/4 inch Whitworth adapter on end of spanner TL4458, torque hub retaining nut to 1200 foot-pounds.

(h) Remove tools and lifting tackle.

(5) Fit cylinder cover and pitch lock.

(a) Ensure blades are in feathered position.

(b) Remove retaining ring and cylinder nut from cylinder cover and pitch lock assembly.

(c) Lubricate pitch lock assembly, cylinder cover oil seal and cylinder thread with engine oil.

(d) Offer up assembly to cylinder inserting small diameter into collet sleeve. Carefully push it in as far as possible and at same time, slightly rotate pitch lock until it is felt that rear serrations have engaged with hub retaining nut serration. Push assembly completely home with cylinder cover flange abutting cylinder shoulder.

(e) Lubricate rubbing ring and insert in cylinder. Ensure it is positioned correctly, with one tab facing forward and three tabs facing aft (MOD (C) VP1677), or with all four tabs facing aft (PRE-MOD (C) VP1677). All tabs must be correctly located in cylinder slots. Push rubbing ring into cylinder until it butts cylinder cover flange.

(f) Using spanner TL3745, thread cylinder nut (LH thread) hand tight.

(g) Obtain correct torque of cylinder nut by ensuring that cylinder nut is hand tight, then slacken off and tighten again hand tight. Mark a line on face of cylinder nut and continue line on to end face of cylinder. From line on cylinder face measure a distance of 0.500 inch in direction of tightening. Mark a line on cylinder face at this position. Using a rawhide mallet, tap spanner to tighten cylinder nut until line marked on cylinder nut coincides with second line marked on cylinder face. Erase marks.

(h) Fit retaining ring in groove in serrations of cylinder cover.

(i) Fit locking segment to engage in serrations of cylinder nut and cylinder cover without forcing. Move locking segment round cylinder nut until a suitable position is found.

(j) Fit retaining ring groove in cylinder nut serrations.

(k) Check that cylinder bleed setscrews are tight and tabbed up.

(l) Adjust pulse generator.

(6) Install propeller spinner as follows.

NOTE: The spinner de-icer leads can be damaged if not properly fitted between cable clip and grommet where lead passes through the back plate. When fitting the spinner to the propeller ensure that any excess lead between the clip and grommet is pulled through to the front of the back plate and suitably routed to the terminal block.

(a) Ensure propeller blades are at an angle approximately midway between flight fine pitch and feather to enable spinner to be installed without fouling.

(b) Rub spinner flex ring with French chalk and coat lockpegs and guide pins of backplate with grease. Ensure all lockpins on spinner are in unlocked position.

(c) Position spinner on propeller hub with offset lockpin housing which is marked with red paint, coinciding with offset lockpeg on backplate.

CAUTION: DO NOT ATTEMPT TO TURN ANY SINGLE LOCKPIN TO FULLY LOCKED POSITION IN ONE MOVEMENT AS DAMAGE MAY RESULT.

(d) While supporting and easing spinner into position, use a screwdriver in slots of four lockpins and turn each toward locked position as far as it will turn without undue force being used. Work progressively around spinner from lockpin to lockpin, turning each a little at a time until spinner is fully home and each lockpin is in locked position.

(e) Turn propeller blades through their complete pitch range and ensure there is no fouling between fixed and moving parts.

(7) Replace engine oil lost on pitch lock removal.

(8) Perform Propeller — Functional Check.

CAUTION: THE HUB RETAINING NUT MUST NOT BE TIGHTENED WHILE ENGINE SHAFT IS HOT, AS SHRINKAGE ON COOLING MAY RENDER SUBSEQUENT REMOVAL DIFFICULT.

DO NOT SLACKEN OFF RETAINING NUT DURING RETORQUING.

NOTE: After initial ground run check for signs of grease leakage from the blade bearings and oil leakage from cylinder cover or propeller shaft.

If there is any evidence of vibration on the initial ground run the hub retaining nut must be checked for tightness.

It is recommended that the propeller be feathered (manually or autofeathered) on the initial flight.

It is recommended that the propeller be retorqued after first flight. However, retorque may be delayed up to 24 flight hours with the following restrictions applied:

(a) No evidence of propeller vibration, oil leakage or grease leakage during initial ground run or the 24 hours of operation.

(b) If propeller vibration, oil leakage or grease leakage is experienced any time during the 24 hours of operation, the propeller must be retorqued prior to further flight.

(9) Retorque hub retaining nut after first flight to 1200 foot-pounds.

4. Propeller System — Functional Checks

A. PROPELLER CONTROL STATIC CHECK:(Perform left propeller system check first, then repeat for right propeller system.

NOTE: Propeller checks are to be performed after an engine, propeller or PCU change, after any disturbance to propeller control system and at inspection periods noted in the Dowty-Rotol Maintenance Manual 865/1.

If during the performance of this procedure, any erroneous readings or indications are observed, repeat procedure from the beginning.

ACTION	CHECK	RECTIFICATION
Move HP cock lever to cruise lock out.	HP cock pickup lever contacts engine control box LO stop with cockpit lever still clear of its pedestal stop.	Adjust aircraft controls to ensure pickup lever contacts stops. Ensure clearance at cruise lock out end of cockpit lever travel.
Move HP cock lever to feather.	HP cock pickup lever contacts engine control box F stop.	
Select HP cock fuel off.	HP cock pickup lever is opposite S mark on engine control box.	Eliminate backlash in aircraft linkage by moving cockpit lever towards feather. Adjust quadrant shut stop plate to contact cockpit lever.
Select HP cock fuel on.	HP cock pickup lever is opposite zero mark on engine control box.	Eliminate backlash in aircraft linkage by moving cockpit lever towards cruise lock out. Adjust quadrant open stop plate to contact cockpit lever.
Disconnect control at aircraft bulkhead. Select HP cock feather (pickup lever contacting F stop on engine control box).	1. Cross shaft end of HP cock control rod is clear of compressor setting block by 0.075 in. to 0.125 in.	1. Adjust control rod between cross shaft and HP cock pickup lever on engine control box.
	2. HP cock lever on FCU is on short (cold setting) line of feather markings or is within 0.010 in. of it toward shut marking.	2. Adjust control rod between cross shaft and unit with unit backlash taken up by rotating FCU lever towards

ACTION	CHECK	RECTIFICATION
	NOTE:Disregard set-hot instructions on FCU quadrant plate.	
	3. Feathering selector lever on PCU is on F band line nearer to UF, or is within 0.015 in. of it towards UF marking.	3. Adjust control rod between cross shaft and unit with unit backlash taken up by rotating PCU lever towards LO (lock out).
Select HP cock fuel off (pickup lever opposite S mark).	1. HP cock lever is within shut marking on FCU.	
	2. Feathering selector lever is within UF markings on PCU.	
Select HP cock fuel on (pickup lever opposite O mark).	1. HP cock lever is within open markings on FCU.	
	2. Feathering selector lever is within run automarkings on PCU.	Suspect distorted control levers and/or linkage.
Select HP cock cruise lock out (pickup lever contacting LO stop on engine control box. Hook up control at aircraft bulkhead.	1. HP cock lever is within open markings on FCU.	
	2. Feathering selector lever is within LO markings on PCU.	

B. Propeller Functional Check

(1) Ensure water/methanol switch is off during all functional checks of propeller system. Failure to do so can result in inadvertent injection of water/methanol into engine when power lever is advanced while oil returned under pressure from action of propeller is reaching water methanol unit.

(2) Switch fuel boost pumps to off when effecting propeller static checks to minimize fuel leakage when HP fuel cock is at fuel on and lock out position.

(3) Install oil bypass equipment as follows:

CAUTION: OIL SPECIFIED FOR USE IN OIL SYSTEM IS A SYNTHETIC PRODUCT AND MUST NOT BE MIXED WITH ANY OTHER OIL, IT IS INJURIOUS TO PAINT WORK AND CERTAIN TYPES OF RUBBER AND MUST NOT BE ALLOWED TO COME INTO CONTACT WITH ANY PART OF ENGINE OTHER THAN OIL SYSTEM. ANY SPILLED DURING SERVICING MUST BE WIPED UP IMMEDIATELY.

COMPLETE BREAKDOWN OF ENGINE SYNTHETIC OIL CAN BE CAUSED THROUGH CONTAMINATION BY CERTAIN ALKALINE CLEANING FLUIDS HAVING THE SAME APPEARANCE AS OIL. ENSURE CONTAINERS AND EQUIPMENT ARE CLEAN BEFORE USING.

(a) Remove pressure oil filter and bypass valve assembly.

(b) Fit oil bypass equipment adapter block. Ensure that sealing ring is located in circumferential recess in adapter block. Fit adapter block into pressure filter housing without dislodging sealing ring and with dowel pin aligned with its locating hole in casing. Finger tighten captive retaining bolt.

(c) Remove engine oil tank dipstick.

(d) Fit bypass equipment oil return pipe into dipstick housing and secure in position with dipstick retaining spring which locates on plate provided.

(4) Effect propeller checks in manner and sequence detailed in TABLE 1 of propeller system functional checks as follows:

CAUTION: SELECT CRUISE LOCK OUT ON HP FUEL COCK LEVER BEFORE ANY UNFEATHERING OTHER THAN DETAILED IN TABLE OF PROPELLER SYSTEM FUNCTIONAL CHECK. THIS WILL ENSURE THAT PROPELLER IS NOT HELD AT CRUISE LOCK, WITH CONSEQUENT FLOODING OF ENGINE OIL SYSTEM. DO NOT OPERATE FEATHERING PUMP MOTOR FOR LONGER THAN IS ESSENTIAL TO CONFIRM A CHECK AND IN ANY CASE DO NOT OPERATE IT FOR MORE THAN 3 MINUTES IN ANY 1 HOUR.

TABLE 1
Propeller System Functional Checks — Engine Stationary

ACTION	CHECK	RECTIFICATION
1. Preliminary (If both propeller systems are to be checked, Steps 1B thru 3 perform left propeller system check first, then repeat for right propeller system).	Propeller on ground fine pitch stop stop (marks on spinner and blades aligned)	Move HIGH-PRESSURE FUEL COCK to CRUISE LOCK OUT and press FEATHER button, release when propeller reaches ground fine pitch stop and return HIGH-PRESSURE FUEL COCK to FUEL OFF
A. Ensure:	BELOW FLT. FINE PITCH LOCK light is ON	Rotate propeller, if defect persists, rectify flight. fine pitch lock hub switch circuit, including appropriate contacts in auto-coarsen relay (assuming lamp serviceable)
(1) Adequate oil supply available. (2) Oil bypass equipment is fitted. (3) Ground electrical supply connected.	Both flight. FINE PITCH LOCK lights are ON	Rectify flight fine pitch lock withdrawal circuit (assuming lamps serviceable), including fine pitch lock selector lever switches and switch linkage and warning light circuit
(4) All circuit breakers IN. (5) Both THROTTLE LEVER closed to IDLE. (6) Both HIGH PRESSURE FUEL COCKS at FUEL OFF.		
(7) FLT. SAFETY LOCK SWITCH at NORM.	CRUISE LOCK OUT light is OUT (oil pressure required to operate)	Change oil pressure operated switch on PCU
(8) FINE PITCH LOCK SELECTOR lever at GROUND INTERLOCK (rearward).	Both CRUISE PITCH lights are on.	Rotate propellers, if defect persists, rectify cruise pitch lock withdrawal circuit, including warning light circuit (assuming lamps serviceable)
(9) GUST LOCK lever at OFF.		

ACTION	CHECK	RECTIFICATION
B.Press and hold PRESS FEATHER button:	FEATHER PUMP warning light is ON	Rectify manual feathering circuit (assuming lamp serviceable)
	Feather pump starts	If current is reaching feather motor terminals, change feather pump unit
	Propeller remains at ground fine pitch stop	If propeller coarsening ceases at flight fine pitch lock angle, rectify the fine pitch lock selector lever switch and switch linkage isolating the electro-hydraulic stop circuit. If propeller feathers fully, rectify control static settings.
	CRUISE LOCK OUT light may be ON (operated by feather pump oil pressure)	Check later with engine running, if still no light, change PCU switch (assuming lamp serviceable)
C. Release PRESS FEATHER button	FEATHER PUMP warning light is OUT and feather pump stops (CRUISE LOCK OUT light is OUT)	Trip manual feather circuit breaker and rectify feather button or feather pump relay
2. ELECTRO-HYDRAULIC STOP (PITCH COARSENING) CHECK		
A. Move FINE PITCH LOCK SELECTOR lever to FLIGHT (forward)	Both FLT. FINE PITCH LOCK lights are OUT	Rectify flight fine pitch lock withdrawal circuit or appropriate fine pitch lock selector lever switches or switch linkage.
B. Press FEATHER button, hold until pitch coarsening confirmed, then release	Propeller coarsens (FEATHER PUMP warning light is ON) until flight fine pitch lock angle (approx.) is reached and remains at this angle (BELOW FLT. FINE PITCH	Rectify electro-hydraulic stop circuit, including appropriate fine pitch lock selector lever switch or switch linkage, or change PCU
C. Visually check that blade angle is hunting at approximately 18°.	LOCK light is OUT) or hunts around it (intermittent flashing of BELOW FLT. FINE PITCH LOCK light)	
NOTE: One or both BELOW FLIGHT FINE PITCH LOCK lights may be on at this time.	**NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump is running	
3. MANUAL FEATHERING CHECKS:		

ACTION	CHECK	RECTIFICATION
A. Functional check		
(1) Move HIGH-PRESSURE FUEL COCK to FEATHER.		
(2) Press FEATHER button until propeller coarsening confirmed, then release immediately.	Propeller starts to coarsen (FEATHER PUMP warning light is ON	Carry out full installation manual feathering check
(3) Return HIGH-PRESSURE FUEL COCK to FUEL OFF.	**NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump is running.	
B. Alternative check on installation or when required.		
(1) Disconnect HIGH-PRESSURE FUEL COCK control at aircraft bulkhead.		
(2) Position PCU feathering selector lever within UF markings.		
(3) Move pilots HIGH-PRESSURE FUEL COCK to FEATHER.		
(4) Press FEATHER button until propeller coarsening confirmed, then release immediately.	Propeller starts to coarsen (FEATHER PUMP warning light is ON) under electrohydraulic FEATHER selection only.	Rectify HIGH-PRESSURE FUEL COCK microswitch or circuit through or circuit through auto coarsening relay to pitch coarsening solenoid.
(5) Return pilots HIGH-PRESSURE FUEL COCK		
(6) Position PCU feathering selector lever at its F (FEATHER) position by moving HIGH-PRESSURE FUEL COCK pickup lever on engine control box to its F limit stop.	**NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump is running.	
(7) Press FEATHER button until propeller coarsening confirmed, then release immediately.	Propeller restarts coarsening (FEATHER PUMP warning light is ON) under mechanical FEATHER selection only.	Rectify control static settings or change PCU.
(8) Connect HIGH-PRESSURE FUEL COCK control at aircraft bulkhead.	**NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump is running.	
(9) Verify appropriate control static checks and ensure security.		

ACTION		CHECK	RECTIFICATION
4. Auto coarsening (auto feathering) checks (covering both LH and RH propeller systems)			
A. Ensure both propellers at flight fine pitch lock angle or above (BELOW FLT FINE PITCH LOCK Lights are OUT) but below fully feathered			If BELOW FLT FINE PITCH LOCK lights are ON while propeller is above 20° lock, check propeller hub switches for proper installation.
B. Ensure FINE PITCH LOCK SELECTOR lever at FLIGHT (forward)			
C. Move the controls in the sequence numbered below:			
LH PROPELLER SYSTEM	**RH PROPELLER SYSTEM**		
(1) HIGH-PRESSURE FUEL COCK to FEATHER			
	(2) HIGH-PRESSURE FUEL COCK to FUEL ON		
(3) POWER LEVER to MAXIMUM POWER	(3) POWER LEVER to MAXIMUM POWER	RH propeller does not auto coarsen (FEATHER PUMP warning light remains OUT) while LH propeller SYSTEM is at FEATHER, although the LH propeller blades may not be at full FEATHER.	Rectify appropriate microswitch and circuit on LH HIGH-PRESSURE FUEL COCK
(4) HIGH-PRESSURE FUEL COCK to FUEL OFF		Neither propeller auto coarsens (FEATHER PUMP warning lights remain OUT)	Rectify appropriate microswitch on LH HIGH-PRESSURE FUEL COCK
	(5) HIGH-PRESSURE FUEL COCK to FEATHER		
(6) HIGH-PRESSURE FUEL COCK to FUEL ON		LH propeller does not auto coarsen (FEATHER PUMP warning light remains OUT) while RH propeller SYSTEM is at FEATHER, although the RH propeller blades may not be at full FEATHER.	Rectify appropriate microswitch and circuit on RH HIGH-PRESSURE FUEL COCK
	(7) HIGH-PRESSURE FUEL COCK to FUEL OFF PUMP	Neither propeller auto coarsens (FEATHER PUMP warning lights remain OUT)	Rectify appropriate microswitch on RH HIGH-PRESSURE FUEL COCK

ACTION		CHECK	RECTIFICATION
(8) POWER LEVER to IDLE	(8) POWER LEVER to IDLE		
	(9) HIGH-PRESSURE FUEL COCK to FUEL ON	Neither propeller auto coarsens (FEATHER PUMP warning lights remain OUT)	Rectify POWER LEVER switch(s) in auto coarsening circuit(s)
(10) POWER LEVER to MAXIMUM POWER		LH propeller starts to auto coarsen (FEATHER PUMP warning light is ON)	Rectify LH propellers auto coarsening circuit, including its isolating relay and appropriate microswitch on RH HIGH-PRESSURE FUEL COCK, when rectified, repeat auto coarsening checks from commencement.
		LH BELOW FLT. FINE PITCH LOCK light is OUT	Rectify LH below flight fine pitch lock light circuit, including appropriate contacts in LH auto coarsen relay.
	(11) POWER LEVER to MAXIMUM POWER as soon as LH propeller auto coarsening confirmed	LH propeller continues to auto coarsen (FEATHER PUMP warning light remains ON)	
		RH propeller does not auto coarsen (FEATHER PUMP warning light remains OUT) while LH propeller auto coarsening is energized	Rectify RH propellers isolating relay and circuit
(12) POWER LEVER to IDLE as soon as nonauto coarsening of RH propeller confirmed		LH propeller stops auto coarsening FEATHER PUMP warning light is ON) FEATHER PUMP warning light is OUT) and	Rectify RH propellers auto coarsening circuit, including its isolating relay and appropriate microswitch on LH HIGH-PRESSURE FUEL COCK when rectified, repeat auto coarsening checks from commencement.
		RH propeller now starts to auto coarsen FEATHER PUMP warning light is ON)	
		RH BELOW FLT. FINE PITCH LOCK light is OUT	Rectify RH below flight fine pitch lock light circuit, including appropriate contacts in RH auto coarsen relay

ACTION		CHECK	RECTIFICATION
(13) SYSTEM POWER LEVER to MAXIMUM POWER as soon as RH propeller auto coarsening confirmed		RH propeller continues to auto coarsen (FEATHER PUMP warning light remains ON) and	
		LH propeller does not restart auto coarsening (FEATHER PUMP warning light remains OUT) while RH propeller auto coarsening is energized.	Rectify LH propellers isolating relay and circuit.
	(14) POWER LEVER to IDLE as soon as nonauto coarsening of LH propeller confirmed	RH propeller stops auto coarsening (FEATHER PUMP warning light is OUT) and	
		LH propeller restarts auto coarsening (FEATHER PUMP warning light is ON)	Rectify LH propellers isolating relay
(15) POWER LEVER to IDLE to stop LH propeller auto coarsening as soon as autocoarsening confirmed			
(16) HIGH-PRESSURE FUEL COCK to FUEL OFF	(16) HIGH-PRESSURE FUEL COCK to FUEL OFF		
5. UNFEATHERING AND CRUISE LOCK CHECKS			
NOTE: These checks written for checking LH propeller system, to check RH propeller system read RH for LH and vice-versa.			
A. Ensure FINE PITCH LOCK SELECTOR lever at FLIGHT			
B. Ensure both POWER LEVERS closed			
LH PROPELLER SYSTEM	**RH PROPELLER SYSTEM**		
C. CHECK HUB/SWITCH AT FEATHER			

ACTION		CHECK	RECTIFICATION
(1) Ensure propeller fully feathered	(1) Ensure RH propeller above cruise lock (not fully feathered		If BELOW FLT. FINE PITCH LOCK lights are ON while propeller is above 20° lock, check propeller hub switches for proper Installation.
	(2) Move HIGH-PRESSURE FUEL COCK to FUEL ON		
	(3) Press FEATHER button until propeller breaks through cruise lock	RH propeller breaks through cruise lock (audible) **NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump is running and both CRUISE PITCH lights will be ON when either propeller is below the cruise lock or is fully feathered.	Suspect LH propeller hub switch is not made when propeller is fully feathered, rotate propeller before investigating further
	(4) Move HP COCK to FEATHER		
	(5) Return propeller to above cruise lock (e.g., about 60°)	CRUISE LOCK OUT light is OUT	
D. PROVE UNFEATHERING AND LOCK EFFECTIVE			
(1) Move HIGH-PRESSURE FUEL COCK to FUEL ON			
(2) Press FEATHER button until propeller is halted, then release immediately		Both CRUISE PITCH lights are OUT as LH propeller moves away from feathered position (FEATHER PUMP warning light is ON) and both are ON Just before propeller is halted	Rotate LH propeller before rectifying cruise lock withdrawal circuit or LH propeller hub switch
		LH propeller is held at cruise lock	If LH propeller passes cruise lock, ensure withdrawal solenoids not energized, if not energized, check control static settings. If control static settings satisfactory, change propeller lock unit or PCU.

ACTION		CHECK	RECTIFICATION
		LH CRUISE LOCK OUT light is OUT	Change PCU switch or suspect propeller third oil line drain blocked
E. PROVE NORMAL ELECTRICAL WITH-DRAWAL OF LOCK			
	(1) Move HIGH-PRESSURE FUEL COCK to FUEL ON		
	(2) Press FEATHER button until propeller breaks through cruise lock	RH propeller breaks through cruise lock (audible) **NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump Is running and both CRUISE PITCH lights will be ON when either propeller is below the cruise lock or is fully feathered.	Suspect LH propeller hub switch Is not made when it is held at cruise lock
(3) Ensure HIGH-PRESSURE FUEL COCK still at FUEL ON			
(4) Press FEATHER button until propeller breaks through cruise lock		LH propeller breaks through cruise lock (audible) (FEATHER PUMP warning light is ON). **NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump is running.	Rectify cruise lock withdrawal circuit, or change propeller lock unit or PCU.
(5) Return propeller to above cruise lock (place HP COCK to FEATHER and press FEATHER PUMP button)			
F. PROVE MANUAL WITHDRAWAL OF LOCK			
(1) Select EMERGENCY on FLT. SAFETY LOCK switch		Both CRUISE PITCH lights are OUT as cruise lock withdrawal solenoids are disarmed	Rectify FLT. SAFETY LOCK switch

ACTION	CHECK	RECTIFICATION
(2) Move HIGH-PRESSURE FUEL COCK to CRUISE LOCK OUT		
(3) Press FEATHER button until propeller breaks through cruise lock	LH propeller breaks through cruise lock (audible), (FEATHER PUMP warning light is ON) **NOTE:** The CRUISE LOCK OUT light may be ON while the leather pump is running.	Check control static settings, if these are correct, change PCU.
(4) Select NORMAL on FLT. SAFETY LOCK switch **NOTE:** Comply with note listed after Step 5 of UNFEATHERING and CRUISE LOCK CHECKS.	Both CRUISE PITCH lights are ON as cruise lock withdrawal solenoids are rearmed	
6. FLIGHT FINE PITCH LOCK CHECK A. Ensure: (1) FINE PITCH LOCK SELECTOR lever at FLIGHT. (2) Both POWER LEVERS at IDLE. (3) Propeller Is above flight line pitch lock but below cruise lock. (4) HIGH-PRESSURE FUEL COCK at CRUISE LOCK OUT		
B. Press FEATHER button until propeller is halted	Propeller fines off (FEATHER PUMP warning light is ON) until it is held at flight pitch lock. **NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump is running.	If propeller passes flight fine pitch lock, ensure withdrawal solenoids not energized, if not energized, change propeller lock unit or PCU.
C. Move FINE PITCH LOCK SELECTOR to GROUND INTERLOCK (rearward)	Both FLT. FINE PITCH LOCK lights are ON	
D. Move HIGH-PRESSURE FUEL COCK to FUEL ON		

ACTION	CHECK	RECTIFICATION
E. Press FEATHER button until propeller is halted	Propeller breaks through flight fine pitch lock (audible), (FEATHER PUMP warning light is ON)	Move HIGH-PRESSURE FUEL COCK to CRUISE LOCK OUT and press FEATHER FUEL ON button again. If propeller now breaks through flight fine pitch lock, rectify cruise lock withdrawal circuit (If not already checked, as above). If propeller is still held at flight fine pitch lock, rectify flight fine pitch lock withdrawal circuit or change propeller lock unit or PCU.
	BELOW FLT. FINE PITCH LOCK light is ON **NOTE:** The CRUISE LOCK OUT light may be ON while the feather pump is running.	Rotate propeller before investigating further
	Propeller fines off to ground fine pitch stop (blade marks aligned)	
F. With its HIGH-PRESSURE FUEL COCK at FUEL ON fine off other propeller to its ground fine on pitch stop		
G. With both HIGH-PRESSURE FUEL COCK to FUEL OFF		
H. Remove oil bypass equipment		
I. Replace pressure oil filters		
J. Move GUST LOCK lever to ON		
K. Before motoring over or ground running engine observe usual oil level precautions		

5. **Propeller — Retorque**

CAUTION: THE HUB RETAINING NUT MUST NOT BE TIGHTENED WHILE ENGINE SHAFT IS HOT, AS SHRINKAGE ON COOLING MAY RENDER SUBSEQUENT REMOVAL DIFFICULT.

DO NOT SLACKEN OFF RETAINING NUT DURING RETORQUING.

NOTE: After initial ground run check for signs of grease leakage from the blade bearings and oil leakage from cylinder cover or propeller shaft.

If there is any evidence of vibration on the initial ground run the hub retaining nut must be checked for tightness.

A. Ensure electrical power is off.

B. Remove propeller spinner as follows:

(1) Ensure propeller blades are at an angle approximately midway between flight fine pitch and feather to enable spinner to be removed without fouling.

CAUTION: DO NOT ATTEMPT TO TURN ANY SINGLE LOCKPIN TO FULLY UNLOCKED POSITION IN ONE MOVEMENT AS DAMAGE MAY RESULT.

(2) Supporting spinner and using screwdriver in slots of four lockpins, turn each lockpin towards unlocked position as far as it will turn without undue force being used, work progressively around spinner from lockpin to lockpin, turning each a little at a time until all four are at unlocked position.

NOTE: To prevent propeller rotation while releasing hub retaining nut, a blade must be supported at 0.7 radius station, preferably with blade at feathered angle. At any other angle however, blade stress will still be acceptable.

(3) Support and withdraw spinner forward until clear of blade and hub.

C. Remove cylinder cover and pitch lock as follows

CAUTION: GREAT CARE MUST BE TAKEN TO AVOID DAMAGE TO CYLINDER AND COLLET SLEEVE BORES. DAMAGE TO BORES, ESPECIALLY TO LEADS FOR OIL SEALS CAN RESULT IN DAMAGE TO PITCH LOCK OIL SEALS ON ASSEMBLY TO PROPELLER.

(1) Ensure propeller blades are feathered.

(2) Remove retaining ring from groove in serrations of cylinder cover nut and remove locking segment.

(3) Remove retaining ring from groove in serration of cylinder cover.

(4) Using spanner TL3745, unscrew and remove cylinder cover nut (LH thread).

NOTE: Place a suitable container beneath propeller to catch oil spillage when cover and pitch lock are being removed.

It is important that rubbing ring is removed independently to avoid any possibility of damage to rubbing ring cogs and cylinder threads.

(5) Withdraw rubbing ring. Two pieces of wire inserted under ring will assist in removal, or two small magnets can be used.

(6) Screw cylinder nut into cylinder a sufficient distance to allow retaining ring to be refitted to cylinder cover.

(7) Fit retaining ring in groove in serrations of cylinder cover.

(8) Using spanner TL3745, unscrew cylinder nut. As nut is unscrewed, cylinder cover and pitch lock assembly will be partially withdrawn from cylinder.

(9) On completion of disengagement of nut with cylinder thread, remove spanner. Fit tool TL3913 to nose of cylinder cover and while supporting cylinder cover, ease out cylinder cover and pitch lock assembly.

(10) Screw support TL3744 into mouth of cylinder (LH thread) and tighten with tommy bar.

(11) Insert spanner TL4458 through support and engage it with hub retaining nut serration.

NOTE: Before tightening, check that Acratork spanner is set to give correct torque. Whenever a hub retaining nut has been overtorqued, a dimensional check of cone must be carried out with particular reference to bottom face of counter bore. If check is satisfactory, cone may be returned to service.

(12) Using Acratork spanner No. B 7/1 together with its 1 1/4 inch Whitworth adapter on end of spanner TL4458, torque hub retaining nut to 1200 foot-pounds (RH thread).

(13) Remove all tools.

D. Install pitch lock and cylinder cover as follows:

(1) Remove retaining ring and cylinder nut from cylinder cover and pitch lock assembly.

(2) Lubricate pitch lock assembly, cylinder cover oil seal and cylinder thread with engine oil.

(3) Install pitch lock assembly into cylinder inserting small diameter into collet sleeve. Carefully push it in as far as possible and at same time, slightly rotate pitch lock until it is felt that rear serrations have engaged with hub retaining nut serration. Push assembly completely home with cylinder cover flange abutting cylinder shoulder.

(4) Lubricate rubbing ring and insert in cylinder. Ensure it is positioned correctly, with one tab facing forward and three tabs facing aft (MOD (C) VP1677), or with all four tabs facing aft (PRE-MOD (C) VP1677). All tabs must be correctly located in cylinder slots. Push rubbing ring into cylinder until it butts cylinder cover flange.

(5) Using spanner TL3745, thread cylinder nut (LH thread) hand tight.

(6) Obtain correct torque of cylinder nut as follows:

(a) Ensure cylinder nut is hand tight then slacken off and again hand tighten.

(b) Mark a line on face of cylinder nut and continue line on to end face of cylinder.

(c) From line on cylinder face measure a distance of 0.500 inch in direction of tightening. Mark a line on cylinder face at this position.

(d) Using a rawhide mallet, tap spanner to tighten cylinder nut until line marked on cylinder nut coincides with second line marked on cylinder face.

(e) Erase all marks.

(7) Fit retaining ring in groove in serrations of cylinder cover.

(8) Fit locking segment to engage in serrations of cylinder nut and cylinder cover without forcing.

(9) Fit retaining ring groove in cylinder nut serrations.

(10) Ensure cylinder bleed setscrews are tight and tabbed up.

(11) Ensure propeller blades are at an angle approximately midway between flight fine pitch and feather to enable spinner to be fitted without fouling.

(12) Rub spanner flex ring with French chalk and coat lockpegs and guide pins of backplate with grease. Ensure all lockpins on spinner are in unlocked position.

(13) Position spinner on propeller hub with offset lockpin housing which is marked with red paint, coinciding with offset lockpeg on backplate.

CAUTION: DO NOT ATTEMPT TO TURN ANY SINGLE LOCKPIN TO FULLY LOCKED POSITION IN ONE MOVEMENT AS DAMAGE MAY RESULT.

(14) While supporting and easing spinner into position, use screwdriver in slots of the four lockpins and turn each toward locked position as far as it will turn without undue force being used. Work progressively around spinner from lockpin to lockpin, turning each a little at a time until spinner is fully home and each lockpin is in locked position.

E. Replace engine oil lost on pitch lock removal.

F. Perform a functional check as follows:

ACTION	CHECK
PRELIMINARY Fit oil bypass equipment	Propeller on ground fine pitch stop. (Marks on spinners and blades aligned) BELOW FLT FINE PITCH lock light ON. Both FLT FINE PITCH LOCK lights ON. CRUISE LOCK OUT light OUT (oil pressure required to operate). Both CRUISE PITCH lights ON.
Ensure: Adequate oil supply available. Ground electrical supply connected. Both throttles closed to IDLE. Both HP cocks at FUEL OFF. Flight safety lock switch at NORMAL. fine pitch select lever at GROUND. INTERLOCK (rearward) gust lock lever OFF.	
Press and hold feathering button.	Feathering warning light comes ON. Feathering pump starts. Propeller remains at ground fine pitch stop. CRUISE LOCK OUT light may come ON (operated by feathering pump oil pressure).
Release feathering button.	Feathering warning light goes OUT and feathering pump stops. (CRUISE LOCK OUT light OUT).
ELECTRO-HYDRAULIC STOP (PITCH COARSENING) CHECK	
Move fine pitch lock select lever to FLIGHT (forward)	Both FLT FINE PITCH LOCK lights go OUT.
HP Cock to FUEL ON.	
Press feathering button, hold until pitch coarsening confirmed, the release.	Propeller coarsens (feathering warning light ON) until flight fine pitch lock angle (approximately) is reached and remains at this angle (BELOW FLT FINE PITCH LOCK Light goes OUT) or hunts around it (intermittent ON/OUT of BELOW FLT FINE PITCH LOCK light).
MANUAL FEATHERING CHECK	
Move HP Cock to FEATHER and press feathering button until prop is feathered.	Prop coarsens (feathering warning light ON). Cruise Pitch lights - ON Cruise Lock Out lights - OUT Flight Fine Pitch Lock Lights - OUT Below Flight Fine Pitch Lock Lights: OUT on test prop. ON on other prop.
UNFEATHERING AND PITCH LOCK CHECK	

ACTION	CHECK
With test prop at feather, other prop at ground fine, throttle at IDLE, HP Cock at FUEL ON,	Prop held at cruise lock due to disarming of cruise lock withdrawal circuit. fine pitch select lever at FLT (full forward) and flight safety lock switch at EMERGENCY, press feathering button until prop reaches cruise lock.
Return flight safety lock switch to NORMAL and press feathering button until prop reaches flight fine pitch lock.	Prop passes through cruise lock. Cruise Pitch lights - ON Cruise Lock Out light may come ON (operated by feathering pump oil pressure).
Move HP Cock to LOCK OUT, fine pitch lock select lever to GRD INTERLOCK (full aft) and press feathering button until prop reaches ground fine stop.	Prop passes through flt fine pitch lock. Cruise Pitch lights - ON. Cruise Lock Out lights - may come ON. Flight Fine Pitch Lock lights - ON. Below Flight Fine Pitch Lock lights - ON.
Remove oil bypass equipment. Replace pressure oil filter. Move gust lock lever to ON. Before motoring over or ground running engine, observe usual oil level precautions.	

6. **X and Z Relay — Removal / Installation**

NOTE: If either X or Z relay is to be replaced, it is recommended that BOTH relays be changed. It is further recommended that X and Z relay assembly P/N 1595B10996-1 be used as a replacement.

By using relay assembly it offers easy removal/installation.

A. Removal

(1) Remove electrical power from aircraft.

(2) Remove FLR No. 11, Fuselage Station 232 to 288.

(3) Remove propeller junction box cover.

(4) Disconnect relay assembly attaching plate to junction box.

(5) Disconnect wires from terminal board to relay assembly.

(6) Remove relay assembly.

B. Installation

(1) Install electrical wires H66L18N, K21G16, K65L16, K61L16, K66M18N, K20G16 and K60L16 to junction box terminal board (see Figure 201

(2) Install relay assembly.

(3) Install propeller junction box cover.

(4) Perform appropriate Propeller System — Functional Checks.

7. Auto Feathering Torque Pressure Switch — Removal / Installation

Refer to Rolls-Royce Dart Maintenance Manual M-Da7-G.

8. Propeller Feathering Pump Relay — Removal / Installation

A. Removal

(1) Ensure electrical power is off.

(2) Gain access to relay located in inverter box, radio rack area, entrance compartment.

(3) Disconnect, identify and insulate electrical connections.

(4) Remove relay.

B. Installation

(1) Install relay.

(2) Connect electrical connections as previously identified. Ensure good electrical connection.

(3) Perform Propeller Feathering Pump — Operational Test, see this Section.

(4) Inspect area for presents of foreign objects, security of all attachments.

(5) Close access panels.

9. Propeller Feathering Pump — Operational Test.

A. Apply electrical power to aircraft.

CAUTION: LIMIT USE OF PUMP TO INDICATE MOMENTARY PROPELLER MOVEMENT ONLY.

B. Place appropriate HP cock to FEATHER.

C. Press applicable feather button until propeller moves toward feather.

D. Place HP cock to cruise lockout.

E. Press feather button until propeller moves to flat pitch.

F. Place HP cock lever to FUEL OFF.

G. Remove electrical power from aircraft.

10. Propeller and Associated Components — Maintenance Practices

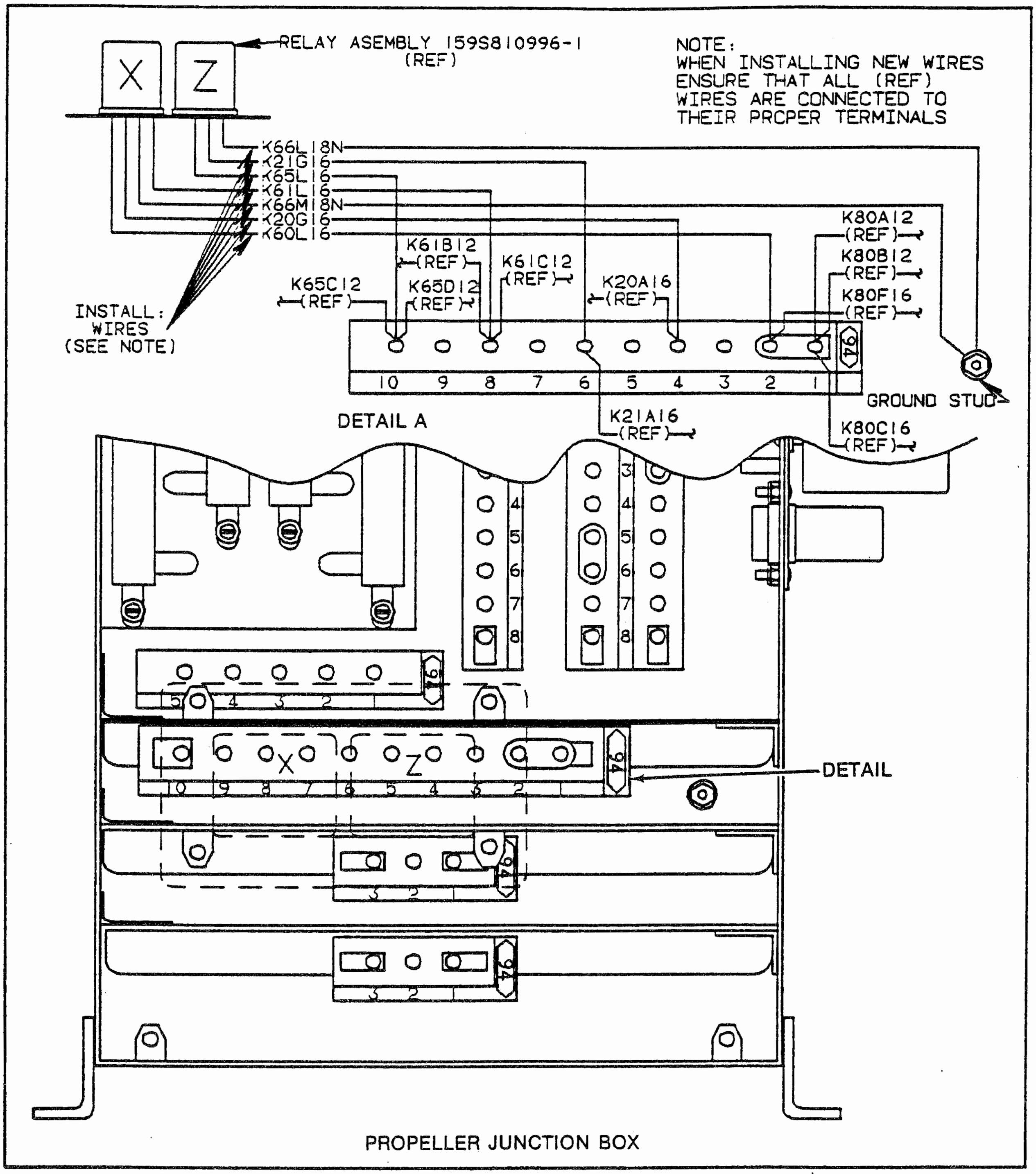
Refer to Dowty Rotol Propeller Maintenance Manual 865/1 and applicable overhaul manual.

11. Propeller Control Unit — Maintenance Practices

Refer to Rolls-Royce Dart Maintenance Manual M-Da7-G and Dowty Rotol Maintenance Manual 865/1.

12. Propeller Feathering Pump — Maintenance Practices

Refer to Rolls-Royce Dart Maintenance Manual M-Da7-G and Dowty Rotol Maintenance Manual 865/1.



X and Z Relay — Schematic
Figure 201.

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PROPELLER SYNCHRONIZING SYSTEM — DESCRIPTION / OPERATION

1. Description

The synchronizing system incorporated in this aircraft consists of two accessory gearbox mounted alternators, one corrector motor mounted in the right nacelle and coupled to throttle lever linkage and a sync button, located on the center of the eyebrow panel just to the left of the cruise pitch lights. When selected by crew, this equipment functions to bring the engine speeds into synchronization and maintain them in that condition. The left engine acts as a master engine, the right is the slave. Each synchronizer alternator armature rotates in a direct relationship with its particular engine rpm. The sync button, when engaged, completes the circuit to the single synchronizer corrector motor installed on the right engine (slave) throttle linkage only. (See Figure 1) The corrector motor compares the output of each synchronizer alternator and adjusts right engine throttle linkage to compensate for differences between the two engine rpms. A friction device, which is a part of the throttle linkages ensures that the motion of the corrector motor is transmitted forward to the engine and not back to the pedestal.

As the engine speeds equalize, the corrector motor stops. Because of the vernier nature of this system and depending upon system friction, matching of engine speeds to within one third of an rpm is possible. In case of failure of the left engine, the right engine is saved down a maximum of 400 rpm from datum. It cannot be decreased lower since this is the limit of travel of the corrector motor ($\pm$ 400 rpm from datum). When the sync button is pulled to deactivate the system, the corrector motor automatically returns to its datum (neutral) position, so that when the system is reengaged full correction in both directions is available. The sync button is the only magnetically held button supplied with the basic airframe. To engage the system the button is pushed and if activated holds in. To release the button, it is pulled out.

A control circuit which consists of switches and relays is incorporated to give the aircraft crew on-off control of the equipment. To prevent the synchronizing system from being operative at times when it impairs aircraft safety (such as during takeoff and landing), it is impossible to engage or operate the synchronizing system when the landing gear handle is in the DOWN position. This is accomplished by a disabling switch, attached to the landing gear handle linkage, which breaks the circuit to the solenoid holding the button. If an attempt is made to land with the synchronizing system engaged, the button is automatically disengaged by placing the landing gear handle down and the SYNC system is deactivated.

NOTE: The automatic disengagement does not depend upon landing gear position, only on landing gear handle position.

It is recommended, for optimum performance of this system, that no power lever changes be made in flight with the synchronizing system engaged. Unsynchronize by pulling the button, synchronize the engines as closely as possible by utilizing the power levers and tachometers, then push the sync button in to engage the system. In this way the synchronizing corrector motor has full travel in both directions for correction purposes.

NOTE: It is not possible to check the propeller synchronizer system on the ground with the landing gear handle in the down position without bypassing the disabling switch. (See Propeller Synchronizer System — Check).

2. Operation

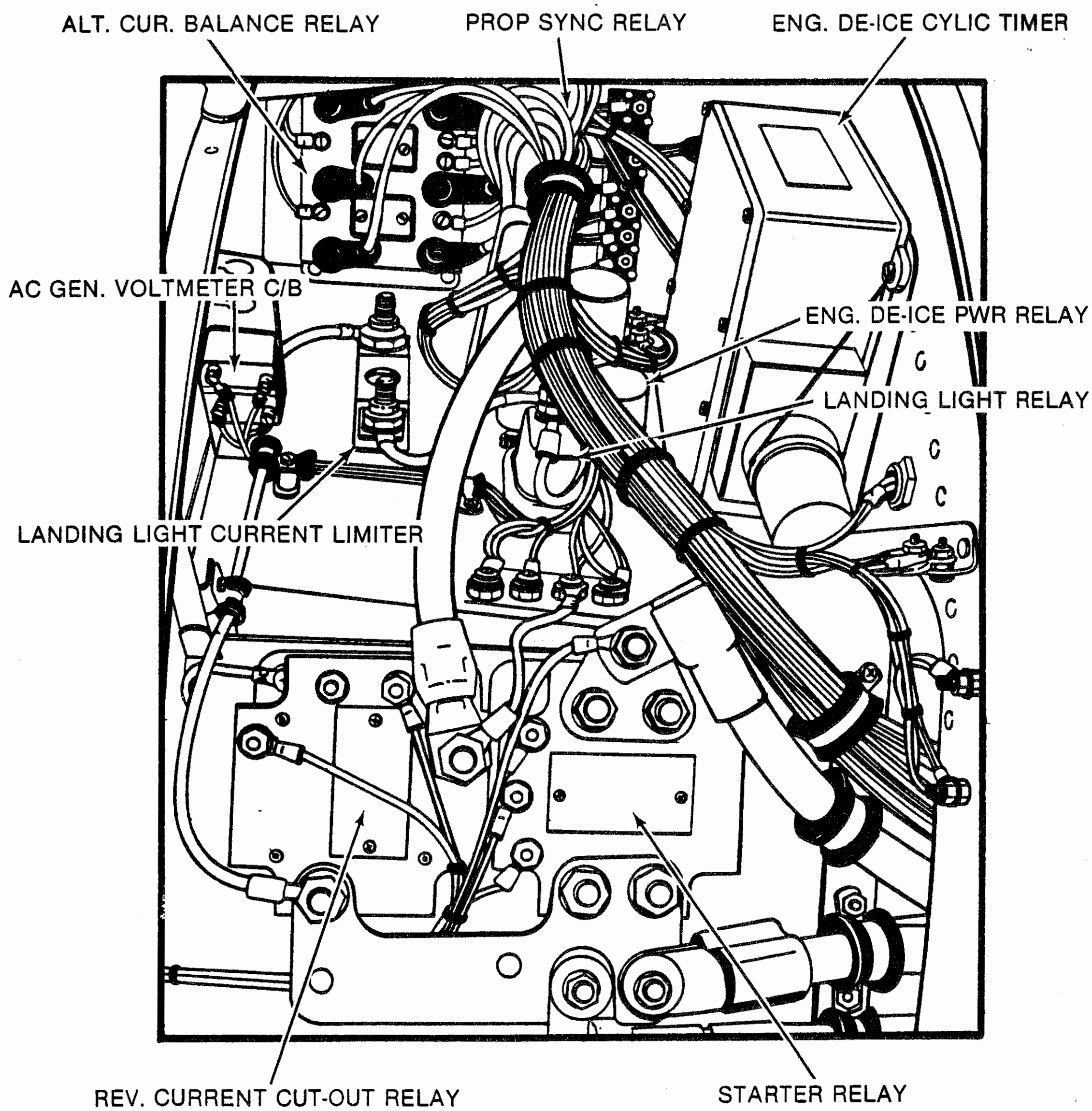
Synchronization is achieved by automatic movement of the slave engine controls until its speed equals that of the master engine. Movement of the slave engine controls is automatically accomplished by the corrector motor (See Figure 2) This is accomplished by first closely synchronizing the engines manually before switching the system on. The automatic synchronization will only take place if the difference in engine rpm is not greater than $\pm$ 400 rpm. The eccentric shaft on the end of the corrector motor moves the power lever bellcrank which is part of the right engine control linkage. To obtain this automatic control linkage movement, the two stators of the corrector motor are fed, in phase opposition, one by the master engine alternator and the other by slave engine alternator. One stator is fed in a clockwise phase direction by the master alternator while the other stator is fed in a counterclockwise direction by the slave alternator. Both stators react on a common rotor. Because the speed of the alternator is relative to that of the engines, the frequency of the current they generate is also proportionate to the engine speed. Therefore, any difference between the master and slave engine speeds results in one stator receiving a higher input frequency than the other. The corrector motor then

revolves in the direction of the stator receiving the higher frequency and moves a throttle lever bellcrank. The direction of rotation of the rotor will always be that required to drive the bellcrank in the proper direction to apply the necessary correction to the speed of the slave engine. As speed of the slave engine approaches that of the master engine, the input frequency of the stator fed by the slave alternator approaches that of the stator fed by the master alternator. When the engine speeds are synchronized, the input frequencies to both stators will be equal and the rotor will cease to revolve. When the control circuit is switched to the off position, a circuit is completed by-way of the relaxed contacts of the synchronizing relay between the corrector motor limit switches and the slave alternator. These switches having been activated by the rotor in an off-datum position complete circuits with the necessary polarity to cause the corrector motor to run toward neutral, until the datum cams open the switches, therefore, the system automatically resets itself to the mid-position (datum) when not in use.

NOTE: With the engine below 12,000 rpm, the corrector motor may not have enough power to return the system to datum, therefore, unsynchronize at 12,000 rpm or above.

The control circuit for the synchronizing system receives its power from the monitor dc bus through the five ampere propeller SYN-N circuit breaker. The landing gear handle (disabling) switch, together with the synchronizing switch, controls power to the synchronizing relay. (See Figure 3) The synchronizing switch is a solenoid held push button type and is opened electrically any time it is pulled out manually or whenever the landing gear handle is placed in the DOWN position through the landing gear handle switch. Depressing the synchronizing switch when the landing gear is in the UP position, energizes the synchronizing relay. The synchronizing relay, when energized, connects the two synchronizing alternator outputs to the corrector motor stator windings.

Propeller synchronizing is not operative in emergency dc operation.



Propeller synchronizing System Equipment in Right Nacelle Relay Box
Figure 3.

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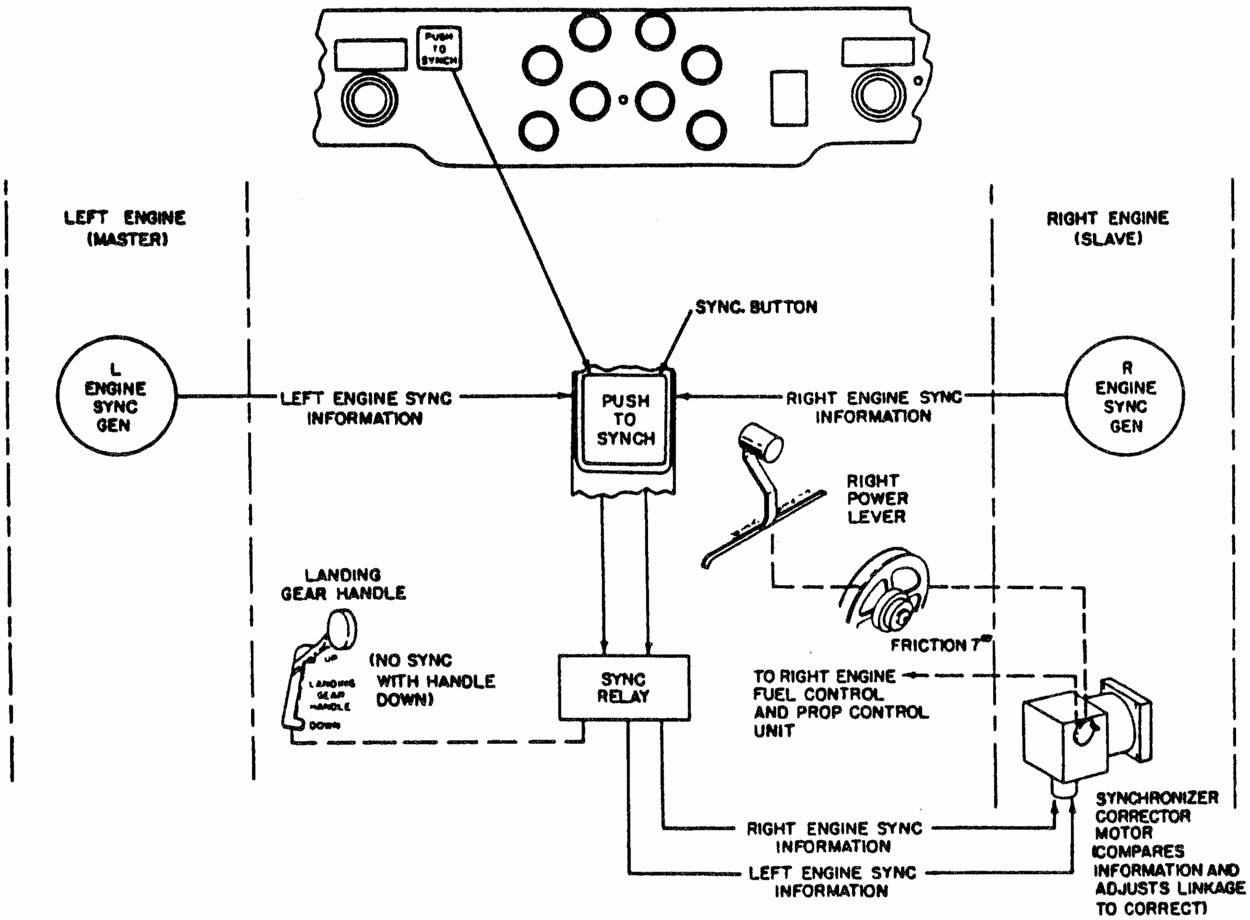
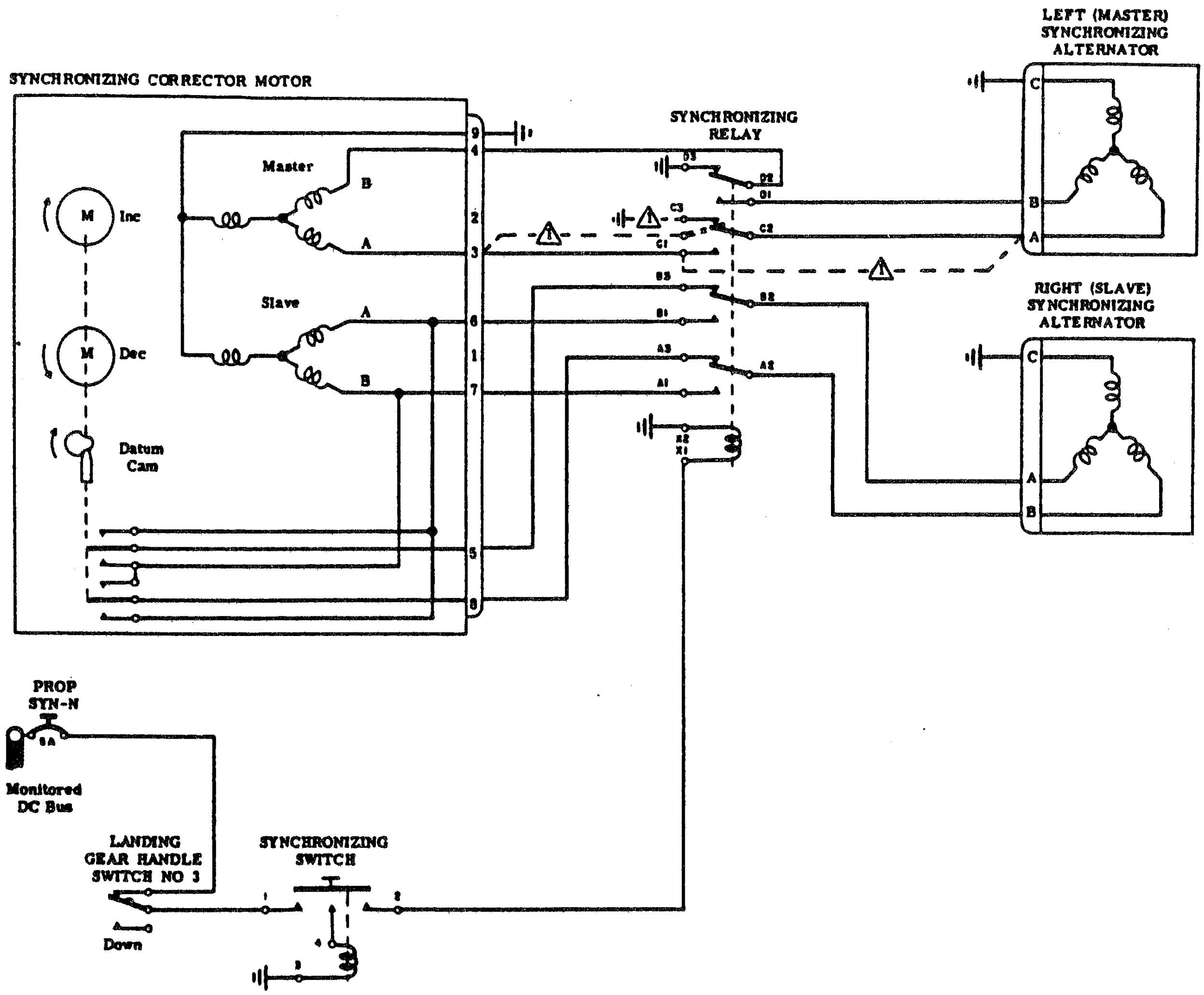


Propeller Synchronizing System — Functional Block Diagram
Figure 1.



NOTES:

⚠1 USED ON SERIAL NO. 52 AND SUBSEQUENT ONLY

Propeller Synchronizing System — Schematic
Figure 2.

PROPELLER SYNCHRONIZING SYSTEM — MAINTENANCE PRACTICES

1. **Propeller Synchronizer System — Check**

NOTE: To perform a ground check of the propeller synchronizing system both engines must be running and a temporary jumper wire must be installed across terminal points 106-1 and 106-2 of the pedestal terminal panel. The jumper wire bypasses the landing gear handle switch which would normally deactivate the propeller synchronizer system while in the down position.

The procedure must be performed with both engines running and synchronizer system off. (monitor bus must be energized)

A. Set both engines to 13,500 rpm

B. Push sync button. RPM should remain at 13,500.

C. Set left engine to 14,000 rpm. Right engine should increase 400 rpm, to 13,900 rpm.

D. Set left engine to 13,000 rpm. Right engine should decrease 400 rpm to 13,100 rpm.

E. Pull sync button. Left engine should remain at 13,000 rpm. Right engine, due to corrector motor return to neutral, will increase to 13,500 rpm.

F. Set left engine to 12,000 rpm.

G. Set right engine to 15,000 rpm.

H. Push sync button. Right engine should decrease 400 rpm to 14,600 rpm.

I. Apply full throttle to right engine and note rpm. It should be 15,000.

J. Pull sync button. The rpm of both engines should remain at their last setting (left, 12,000 rpm, right, 15,000 rpm).

K. Set both engines to 12,000 rpm.

L. Push sync button. The rpm should remain at 12,000 rpm.

M. Pull sync button.

N. Shut down engines.

O. Remove jumper wire from pedestal terminal panel.

2. **Propeller Synchronizing System — Resistance Check**

NOTE: See Figure 202 for connector locations.

A. Remove right nacelle panel to gain access to corrector motor.

B. Remove Plessey (breeze) from corrector motor.

C. Using a digital volt-ohmmeter, perform electrical checks 1, 2, 3, and 4 on Figure 201.

NOTE: Resistance values provided are overhaul limits for returning a unit back to service. Values are to be used for comparison purposes and further fault isolating.

D. Install jumper between terminal point 106-1 and 106-2. Terminal board is located behind the forward right panel of the center console.

E. Energize main and monitored dc buses.

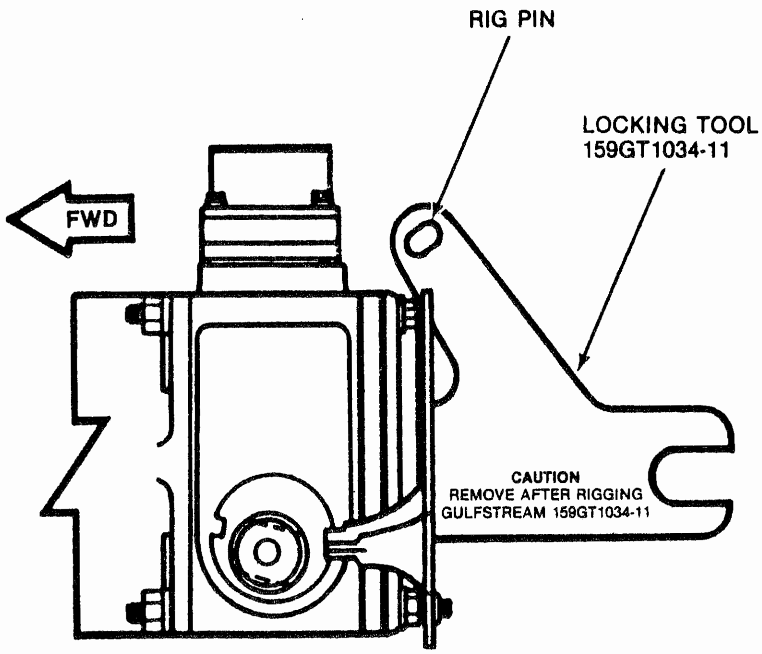
F. Push prop sync button in to energize synchronizing relay.

G. Perform electrical check 5.

NOTE: Resistance values provided are overhaul limits for returning a unit back to service. Values are to be used for comparison purposes and further fault isolating. When checking line to line voltages, compare voltage readings and engine RPM.

CHECK	PINS	READING
1. Corrector Motor Centered	3 to 9 4 to 9 6 to 9 7 to 9 5 to 9 8 to 9	6.16 to 7.18 OHMS 6.16 to 7.18 OHMS 6.16 to 7.18 OHMS 6.16 to 7.18 OHMS Open Open
2. Motor Rotated Fully Clockwise	5 to 6 5 to 7 6 to 8 7 to 8	Continuity 6.16 to 7.18 OHMS 6.16 to 7.18 OHMS Continuity
3. Motor Rotated Fully Counter-clockwise	5 to 6 5 to 7 6 to 8 7 to 8	6.16 to 7.18 OHMS Continuity Continuity 6.16 to 7.18 OHMS
4. Sync Alternators Relay De-energized	3 to 9 4 to 9 6 to 9 7 to 9 5 to 9 8 to 9	See Note Below Continuity Open Open 1.71 to 1.99 OHMS 1.71 to 1.99 OHMS
5. Sync Alternators Relay Energized	3 to 9 4 to 9 6 to 9 7 to 9 5 to 9 8 to 9	1.71 to 1.99 OHMS 1.71 to 1.99 OHMS 1.71 to 1.99 OHMS 1.71 to 1.99 OHMS Open Open
NOTE: A/C 1 - 51 and 114 will read open A/C 52 - 200, 322 and 323 will read continuity		

Corrector Motor/Sync Alternator Circuits Electrical Checks
Figure 201.



Locking Tool — Installation
Figure 202.

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PROPELLER SYNCHRONIZER ALTERNATOR — DESCRIPTION / OPERATION

1. Description

A propeller synchronizing alternator is mounted on and driven by engine-driven gearboxes. The alternator consists of a shaft mounted, permanent magnet type rotor rotating within a three phase, coil wound stator. Current is induced in the windings of the stator. Alternator will generate three phase alternating current at a frequency proportional to engine speed. Driven end of the rotor shaft, which runs in a single row ball bearing, is externally splined to mate with driving member of gearbox. Stator and rotor are housed in a cylindrical case having one end closed to form a mounting base for the alternator. Open end of the case receives a flange mounted end frame which incorporates a single row ball bearing for rotor and also electrical connector for leads.

NOTE: Refer to Dowty Rotor Manual for Maintenance Practices not covered in this section.

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PROPELLER SYNCHRONIZER ALTERNATOR — MAINTENANCE PRACTICES

1. **Propeller Synchronizer Alternator — Removal / Installation**

   A. Removal

      (1) Remove accessory gearbox forward access cover.

      (2) Remove electrical power from aircraft.

      (3) Remove rubber boot and safety-wire from electrical connector, disconnect connector, cap and stow connector.

      (4) Remove alternator.

      NOTE: Driving quill shaft remains with gearbox and oil seal remains with alternator.

   B. Installation

      (1) Remove blanking devices from replacement alternator.

      (2) Check gearbox stud and connector threads.

      (3) Ensure connector pins and serrations on quill shaft are not damaged.

      (4) Ensure that alternator and gearbox mounting faces are clean, dry and undamaged.

      (5) Coat driving quill shaft with clean engine oil and position in gearbox.

      (6) Ensure that a serviceable gasket is fitted to gearbox mounting face.

      (7) Ensure oil seal fitted to rotor shaft driving end is undamaged and correctly fitted.

      (8) Check rotor shaft serrations for a good sliding fit in driving quill.

      (9) Position alternator on gearbox mounting face and ensure rotor shaft serrations engage freely with those in quill shaft.

      (10) Inspect alternator electrical connections and wiring for condition and security.

      (11) Secure alternator, tighten nuts evenly.

      (12) Connect and safety-wire electrical connector, install rubber boot over connector.

      (13) Install access cover.

      (14) Perform Propeller Synchronizer System — Check.

2. **Propeller Synchronizer Alternator - Inspection**

   A. Remove electrical power from aircraft.

   B. Remove gearbox access panel.

   C. Inspect alternator nuts for security.

   D. Inspect Breeze plug and electrical connections for security.

   E. Inspect for oil leaks at mounting face, replace gasket if leaking.

   F. Remove access cover and check condition of electrical leads, replace gasket and install access cover. Apply a light coat of Endolac varnish to stud threads.

   G. Inspect area for presents of foreign objects, security of all attachments.

   H. Install access panel.

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PROPELLER SYNCHRONIZING CORRECTOR MOTOR – DESCRIPTION / OPERATION

1. **Description**

The corrector motor consists of a squirrel cage type rotor rotating within two three phase coil wound stators, a shaft mounted cam, worm gear wheel and clutch assembly, a limit switch and the electrical connector. The body case is closed at one end to form the mounting base for corrector motor and houses rotor and stators. Open end of case receives the flange mounted gearbox containing the worm gear wheel output shaft assembly, limit switch and electrical connector.

2. **Operation**

The rotor shaft at gearbox end terminates in a worm gear which drives worm wheel gear in the gearbox. The worm wheel gear drives its output shaft through a spring-loaded clutch. One end of output shaft terminates in an eccentric pin on which aft throttle lever bellcrank is mounted. The cam mounted on worm wheel gear shaft assembly has integral stop faces which contact a stop peg in the gearbox and limit travel of the output shaft in either direction. Profile of this cam operates the limit switch assembly which is incorporated in this unit to return output shaft to its midposition when propeller synchronizing system is switched off.

NOTE: Refer to Dowty-Rotol Manual for Maintenance Practices not covered in this section.

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PROPELLER SYNCHRONIZING CORRECTOR MOTOR — MAINTENANCE PRACTICES

1. **Propeller Synchronizer Corrector Motor — Removal / Installation**

   A. Removal

      (1) Remove inboard right nacelle access cover.

      (2) Disconnect engine controls from lever assembly on corrector motor.

      (3) Remove safety-wire and rubber boot, remove electrical plug.

      (4) Remove motor.

      (5) Remove operating lever by removing split-pinned nut and washer securing it to spigot.

   B. Installation

      (1) Position unit on its mounting so output shaft eccentric spigot lines up with aircraft control linkage.

      (2) Secure unit on its mounting.

      (3) Connect electrical lead socket to unit plug. Tighten and safety-wire.

      CAUTION: THE V-NOTCH IN THE SHAFT FLANGE MARKED B MUST BE ALIGNED WITH THE DATUM LINE ON THE GEARBOX ON THIS UNIT TYPE SNC4/B WHEN ENGAGING THE TOOL IN THE VERTICAL SLOT.

      (4) Using spannering end of special locking plate 159GT1034-11 as necessary, set output shaft to its datum position i.e., turn shaft to line up V-notch marked B in shaft flange with datum line marked on gearbox case.

      (5) Bolt locking plate on long studs of corrector unit with its tongue engaged in vertical slot of output shaft flange and temporarily secure with slave 4BA. plain nuts. (See Figure 201)

      (6) Inspect area for foreign objects.

      (7) Install rubber boot.

      (8) Install lever assembly to corrector motor using rig pin to align lever assembly with special locking plate, secure lever assembly to corrector motor with hardware and safety with cotter pin.

      (9) Install engine controls to lever assemble.

      (10) Remove locking plate and rig pin.

      (11) Inspect area for presents of foreign objects, security of all attachments.

      (12) Install access cover.

      (13) Perform Propeller Synchronizer System — Check.

2. **Propeller Corrector Motor - Inspection**

   A. Remove inboard right nacelle access cover.

   B. Inspect for following:

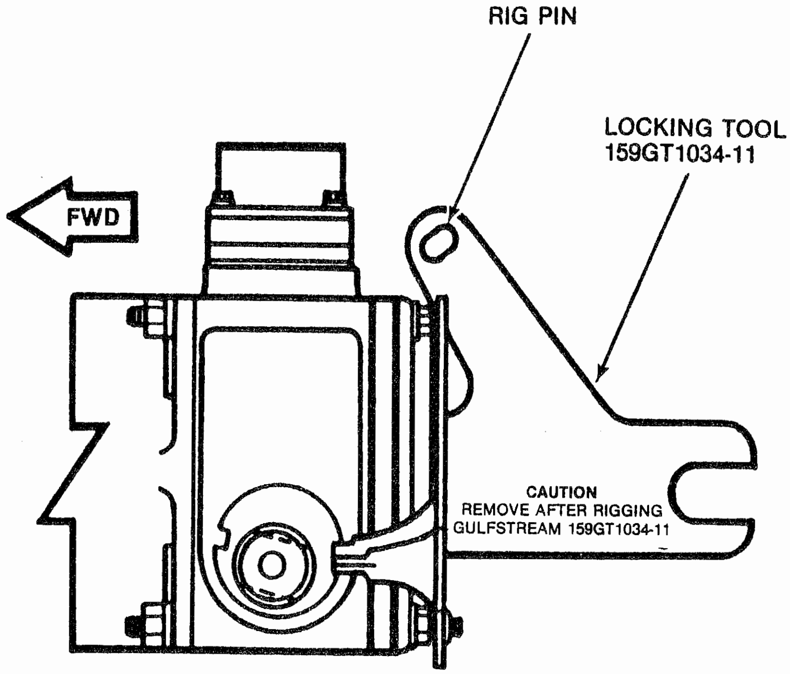
      (1) Corrector motor for security, ensure engine control lever is secure on output shaft.

      (2) Electrical connector and wiring for security.

      (3) Remove terminal lid and inspect electrical connectors and contacts.

      (4) Install terminal lid, apply varnish to stud threads.

   C. Inspect for presence of foreign objects then install access cover.



Locking Tool — Installation
Figure 201.

LIST OF EFFECTIVE PAGES

CHAPTER 71

POWER PLANT

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*71	ii	July 10/92
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*71-0	2	July 10/92
*71-0	3	July 10/92
*71-0	4	July 10/92
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** Pages added by Revision 47
*** Pages deleted by Revision 47

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CHAPTER 71

POWER PLANT

POWER PLANT — DESCRIPTION / OPERATION

1. General

(See Figure 1)

This aircraft is powered by two turboprop engines. These engines are mounted in the nacelles slightly above the wings. Each is fitted with a four-blade, 11 1/2 foot diameter propeller. The complete power plant forward of the firewall includes the engine, engine mount, propeller, engine and propeller controls, cowling and firewall. The engine mount is a tubular steel truss which attaches to the engine at three points, top, both sides and to the firewall at four points. The oil system is completely self-contained within the engine, and provides lubrication for the main shaft bearings, the reduction gearing, and the various power takeoff gearing and support bearings. Oil for normal operation and for feathering of the propeller is also supplied from the same system. The oil tank, integral with the engine, is located around the rear of the air intake casing. Its capacity is approximately 30 US pints.

Air is admitted to the engine through an annular duct formed by the air intake casing and reduction gear housing. The air intake casing, one of the main structural components of the engine, provides a smooth entry for the air and forms the nose ring of the engine cowling. An airscoop, incorporated at the top of the air intake casing, leads to the integral engine-mounted oil cooler. This scoop and the main annular air intake are electrically protected against icing.

The accessory gearboxes located in each nacelle aft of the firewalls, are mounted to the airframe. They are connected to the engine's power takeoff pads by splined and jointed universal drive shafts. Each gearbox is fitted with a hydraulic pump, a dc generator, an alternator, a synchronizer generator and a tachometer generator. The left gearbox incorporates a brake device for stopping propeller rotation after engine shutdown (Aircraft 1 - 148, 322 and 323). The right gearbox is fitted with a supercharger for pressurization and air conditioning. The engines are so located that the exhaust is ducted aft over the top of the wing for maximum thrust utilization. The tailpipe is basically circular in cross section, with a funnel shaped forward sections which forms an ejector with the engine exhaust pipe to induce a cooling flow of air through zone 2 (the engine hot zone). The forward section has a flange which is secured to the firewall. The tailpipe is supported at four points (two on each side) by brackets and arms from the airframe structure. Protection against heat and fire is provided by a blanket around the tailpipe and a radiation shield between the tailpipe and upper surface of the wing.

An engine fireshield is an integral part of the engine and is secured to the aft side of the second stage compressor casing. The annular fireshield separates the engine cold section and hot section (fire zone No. 1 from fire zone No. 2). The annular shield has a U shaped seal channel around its periphery, providing a seat for the fire-resistant rubber seal gasket. The gasket is joined at the top of the channel and an asbestos sleeve is fitted over the joint.

The engine cowling is of the clamshell type with the top panel and left side panel forming one section, and the bottom panel and the right side panel forming the other section of the shell. The panels are interchangeable between the nacelles. The engine cowl panels incorporate Vee shaped fireseal channels which mate with and compress the fire-resistant rubber seal gasket when the cowl panels are closed and secured. The arrangement of the annular shield, the fire-resistant rubber seal gasket and the fireseal channels of the cowl panels form a fire and liquid tight seal.

NOTE: The Dart 529-8E is installed on Aircraft 1 - 146. The DART 529-8H is a modified 529-8E engine that is installed on Aircraft 147 - 164, 322 and 323. The 529-8X (RR MOD 1365) engine is installed on Aircraft 165 - 200 excluding 322 and 323, and Aircraft 1 - 164, 322 and 323 having ASC 176.

The Dart 529-8X (RR MOD No. 1814), 529-8Y and 529-8Z engines provide a higher maximum continuous power and can be used only on Aircraft having ASC 246.

CAUTION: IT IS ACCEPTABLE TO INSTALL DART MK529 ENGINES OF DIFFERENT POWER RATINGS IN THE AIRCRAFT, HOWEVER, BOTH ARE TO BE OPERATED AT THE STANDARD OF LOWEST RATED ENGINE. ASC STATUS MUST BE CONSIDERED, I.E. A/C HAVING ASC 176, 176A AND/OR 246, PRIOR TO INSTALLATION OF A DIFFERENT STANDARD ENGINE MODIFICATION MAY BE REQUIRED TO BRING AIRFRAME AND HIGHER RATED ENGINE TO REQUIREMENTS OF LOWER RATED ENGINE. IT IS RECOMMENDED THAT ENGINE OPERATING LIMITATIONS CARD AND BOTH TGT INDICATORS BE REPLACED WITH CARD AND INDICATORS REQUIRED FOR LOWEST RATED ENGINE. OPERATIONAL RESTRICTIONS AND LIMITATIONS TO BE OBSERVED WILL BE AS PUBLISHED IN GI AFM OR APPROPRIATE GI ASC 246 SUPPLEMENT FOR LOWEST RATED ENGINE INSTALLED.

2. **Power Plant Data**

A. Manufacture .... Rolls-Royce Limited
B. Model .... Dart Mark 529-8E, 8H, 8X, 8Y or 8Z
C. Propeller reduction gear ratios 0.093:1
D. Engine operating limitations:

RPM AND TGT LIMITATIONS				
CONDITIONS	**RPM**		**MAX. TGT °C**	**TIME LIMIT**
During Starts			930	Momentary
Static Ground	6500-7500		550	Unrestricted
Takeoff				
With W/M Without W/M	15,000 15,000		860 825	5 minutes 5 minutes
Max Continuous MK529 8E, 8H and 8X	15,000		850	Unrestricted
MK529-8X (RR Mod 1814) (ASC 246)	15,000		See Note Below 870	Unrestricted
MK529-8Y (ASC 246)	15,000		See Note Below 910	Unrestricted
MK529-8Z (ASC 246)	15,000		See Note Below 920	Unrestricted
Other Conditions	Below 15000 14000 13000 10400	Not Below 14000 13000 10400	785 760 730 550	Unrestricted Unrestricted Unrestricted Unrestricted
Overspeed	17,000 Maximum			20 Seconds

NOTE: Avoid all continuous operation below 7000 rpm.

ASC 246 must be installed in order to utilize higher maximum continuous power of the Dart 529-8X (RR MOD 1814), Dart 529-8Y or Dart 529-8Z.

E. Minimum guaranteed powers:

(1) Takeoff power (dry) and maximum continuous power.

- Sea level static SHP 1910
- Turbine gas temperature .... See Engine Calibration Card in cockpit
- RPM 15,000

(2) Takeoff power (wet)

CAUTION: AT LOW ALTITUDE AND TEMPERATURES BELOW ISA. THIS ENGINE MAY PRODUCE MORE POWER FOR TAKEOFF THAN THE AIRCRAFT HAS BEEN CERTIFIED FOR. UNDER THESE CONDITIONS, DOWNTRIMMING THE ENGINE IS REQUIRED TO ENSURE THAT THE MAXIMUM POWER STATIC TORQUE LIMITATIONS SHOWN ON THE ENGINE CALIBRATION CARD (POSTED IN THE COCKPIT) ARE NOT EXCEEDED.

- Sea level static SHP .... 1990(529-8E or 8H) 1990(529-8X, 8Y or 8Z)
- Water/methanol power check pressure .... See Engine Calibration Card in cockpit
- RPM 15,000

3. **Propeller Data**

- Manufacturer .... Dowty-Rotol, Ltd.
- Model .... (c) R 184 / 4-30-4 / 50
- Tip to tip diameter .... 138 inches
- Pitch range .... 0° to 85° at 0.7 radius station
- Pitch locks:
  Flight fine .... 20° at 0.7 radius station
  Cruise .... 34.5° at 0.7 radius station
- Maximum governing RPM 15,000
- Minimum governing RPM 11,000
- RPM limits .... 17,000 (Max 20 Seconds)

4. **Power Plant Instrument Markings**

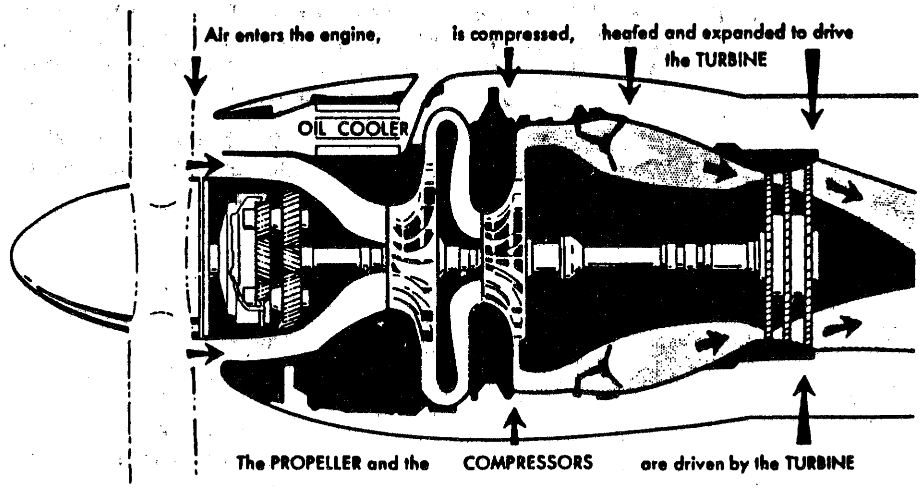
- Maximum and minimum limitations — red radial line
- Takeoff and caution range — yellow arc
- Normal operating ranges — green arc

NOTE: Calibration cards are changed or corrected whenever engine calibration data alters or an engine is changed. The procedure for filling out this card can be found under Engine — Removal / Installation this Section.

5. **Foreign Object Damage Prevention**

Entry of moisture or foreign objects into the engine ducts can cause serious damage to power plant, which, could result in engine failure. Intake and exhaust ducts should be inspected for the presence of loose objects before running the engines.

When the aircraft is not flying or the engine is not operating, the engine inlet duct cover (159GT1026), engine tailpipe cover (159GT1025) and oil cooler intake cover (59GT1013) must be installed.



Dart Propeller Turbine Engine
Figure 1.

POWER PLANT — MAINTENANCE PRACTICES

1. **Power Plant Exterior — Inspection**

   A. Engine cowling for condition, security, legibility of markings and evidence of leakage.

   B. Accessory gearbox/nacelle access panels for condition and security.

   C. Air intake for condition, security, evidence of overheating and freedom from obstructions.

   D. Compressor first stage rotating guide vanes for condition and evidence of failure.

   E. Rotate propeller and check for internal engine noise.

   F. Visually inspect propeller for condition and evidence of leakage.

2. **Power Plant Interior — Inspection**

   A. Open power plant cowling.

   B. Power plant cowling for condition, security and legibility of markings.

   C. Air intake cowling and scoops for condition, security, evidence of overheating and freedom of obstructions.

   D. Fuel, oil and water/methanol installations including components, lines, hoses, fittings, connectors, control rods and linkages for condition, security and evidence of leakage.

   E. Fuel heater for condition, security and evidence of leakage.

   F. Firewall for condition, security and evidence of failure. Fuel, water/methanol, fire protection and engine breather installations attached to firewall for condition and security.

   G. Inspect engine control rods, linkages and ball sockets, (while moving controls through their full travel range) for condition of tightness, proper connections and loose rivets. Replace any worn ball sockets and any rods with working rivets.

   H. Engine mount (Vickers) and mount bolts for condition, security and evidence of failure. Ensure that ID Band installation has proper part number (81037-29) and due date.

   I. Electrical connections and plugs at firewall for security, safeties, condition and evidence of overheating.

   J. Fire detector wire for evidence of sharp bends and condition of rubber inserts.

   K. Igniter cables for condition, security of clamps and connectors and evidence of corrosion.

   L. Power plant harness for condition, signs of chafing, security of plugs, safeties and signs of corrosion.

   M. All wires for condition, clamp security, adequate lacing and evidence of chafing.

   N. Fire extinguisher pipe for chafing and security of clips and safety-wire.

   O. Discharge holes for obstruction; apply 50 - 90 psi air pressure to discharge tube in main gear wheel well and check that all holes vent freely. (Clean blocked holes with No. 60 drill or straight 20 S.W.G. steel wire.

   P. Clamp insulators for condition in fire detector system.

   Q. Annular gap between exhaust unit and tailpipe; clearance should be 1/4 inch minimum.

   CAUTION: WHEN DRAINING A SAMPLE FROM FUEL FILTER, ENSURE THAT AN APPROPRIATE FUEL BOOST PUMP IS OPERATING.

   R. Drain sample from fuel filter.

   S. Inspect for security of all attachments.

   T. Inspect area for presence of foreign objects then close cowlings.

3. **Engine — Operational Test**

CAUTION: GROUND RUNNING SHOULD BE KEPT TO A MINIMUM. AVOID UNNECESSARY STARTS, RAPID CHANGES IN ENGINE CONDITIONS AND PROLONGED RUNNING AT HIGH TURBINE GAS TEMPERATURE.

DO NOT MAKE CONTROL INTERCONNECTION GROUND RUNS IN FOG OR RAIN OR STRONG CROSS WINDS AS CORRECTION CHARTS FOR THESE CONDITIONS ARE NOT PRACTICABLE AND THE RESULTS, THEREFORE, WILL BE INCONSISTENT.

NOTE: There are engines in service which have mintorque declared but which do not have **Torque Pressure Upper Limit Line** values required for application of new in service torque pressure upper limits.

A. If an engine has mintorque values declared but no **Torque Pressure Upper Limit Line** values proceed as follows:

(1) From engine data plate, subtract minimum turbine gas temperature value from maximum turbine gas temperature value.

(2) To value obtained in Step (1) add 20, then add the sum obtained to mintorque psi values at 70° and 100°C oil inlet temperature.

(3) These two new psi values are referred to as the **Torque Pressure Upper Limit Line** values and are to be used in this test and recorded in appropriate aircraft documents.

(4) Following serviceability check limits and corrections pertain to in service engines only. For new engine installations, pre-removal and reinstalled engine ground run limits and corrections, (see Rolls-Royce Manual M-Da529-G, Chapter 71).

B. Preparation

Perform the following actions as required.

(1) Perform Engine Control — Static Check before carrying out an engine serviceability ground run, see Section 76-0.

(2) After propeller, engine or PCU change or after any disturbance to propeller control system, perform Propeller System - Functional Check before ground run, see Section 61-0.

(3) Check fuel pump engine overspeed governor setting on first ground run following installation of a replacement engine or fuel pump, see Rolls-Royce Manual M-Da529-G, Chapter 73).

(4) Engine power checks should not be carried out in ambient conditions in excess of ISA + 35°C (see Rolls-Royce Manual, Chapter 71).

(5) Before operating engine, perform Engine TGT Indicator (EMF CK) - Functional Test and TGT Harness and Thermocouple - Check, see Section 77-5-0.

(6) Validate RPM indicators during operational test, perform Engine Tachometer Indicator - Validation, see Section 77-1-0.

(7) Install master torque gauge.

(8) Suspend a mercury thermometer from a safe position, adjacent to and in shade of the aircraft. Record ambient air temperature.

(9) Obtain prevailing pressure altitude from aircraft altimeter by setting it to 29.92 in.HG. (1013 millibars) and reading off the pressure altitude in feet, above or below sea level.

(10) Record following for future reference:

(a)	Engine serial number	Left ______________		Right ______________
(b)	Outside air temperature	Left ______________°C		Right ______________°C
(c)	Pressure altitude	Left ______________Feet		Right ______________Feet
(d)	Fuel datum position	Left ______________%		Right ______________%
(e)	Dew point	Left ______________°C		Right ______________°C

NOTE: If fuel datum is 100%, use TGT corrections, see Figure 201. If fuel datum is less than 100%, record fuel datum position in Step (d) above and Step K.(1).

(11) TGT corrections - See Figure 201.

NOTE: Subtract correction value (see Figure 201) from appropriate data plate limitation to obtain corrected value of the following items:

• Data plate maximum takeoff TGT	Left	_____________°C	Right	_____________°C
• Correction figure	Left	_____________°C	Right	_____________°C
• Corrected TGT	Left	_____________°C	Right	_____________°C

NOTE: Record Corrected TGT data above in Step K.(5) below.

(12) Torque pressure (Refer to Engine Logbook) and record the following readings:

• OIT	Left	_____________°C	Right	_____________°C
• Mintorque at 70°	Left	_____________psi	Right	_____________psi
• TPUL at 70°	Left	_____________psi	Right	_____________psi
• Mintorque at 100°	Left	_____________psi	Right	_____________psi
• TPUL at 100°	Left	_____________psi	Right	_____________psi

NOTE: Record figures obtained above for Step K.(6) and enter on a graph.

C. Exterior Inspection and Ground Run Preliminaries

(1) Check all systems complete, warning placards, rig pins and warning streamers are removed.

(2) Head aircraft into wind.

(3) Ensure torque link steering disconnect is attached to steering cylinder, and pip-pin is inserted.

(4) Check brake accumulator for condition (nose wheel well left hand side).

(5) Ensure all static air openings, air intake, exhaust openings and louvers are free from obstructions and all covers are removed.

(6) Check engine oil level. (Do not operate engines with oil level less than 4 pints from full.)

(7) Check accessory gearbox oil level; ensure accessory gearbox dipstick is secure.

(8) Open aft fire extinguisher door (spring loaded door) and check engine control rods for security and firewall disconnects.

(9) Check that all access doors and caps are secured.

(10) Ensure that landing gear ground locks are on and wheels are properly chocked.

(11) Ensure both batteries are connected and battery compartment doors are closed.

(12) Observe local prestart orders and fire precautions.

(13) Rotate each propeller in direction of rotation; if unusual noise or resistance to rotation is encountered, cause must be investigated and remedial action taken before starting engine.

(14) Energize external power supply, if required.

NOTE: If prop sync system check is required, install a temporary jumper wire across 106-1 and 106-2 of pedestal terminal panel.

D. Interior checks

(1) Check brake accumulator preload (800 $\pm$ 25 PSI) ....CHECKED

(2) Parking brake ....ON

(3) Landing Gear Selector ....DOWN

(4) Auxiliary hydraulic pump switch ....OFF

(5) Check all circuit breakers ....As required.

(6) Monitor bus test switch ....OFF

(7) Battery switch ....NORMAL

(8) E-Inverter..ON

(9) Check battery voltage is above 22V (E-inverter and 1 boost pump on) pull L NORM BATT circuit breaker. Read minimum of 22V on DC bus voltmeter (right battery voltage). Reset L NORM BATT circuit breaker and pull R NORM BATT circuit breaker. Read left battery voltage. Reset R NORM BATT circuit breaker..CHECKED

(10) External power switch..ON (MIN 28V)

(11) Generator switches..OFF

(12) Alternator switches..OFF

(13) Emergency flap handle..NEUTRAL

(14) Blower switch..OFF

(15) Air conditioning vent switch..AIR COND

(16) Fuel crossfeed switch..CLOSED

(17) Wing and tail de-ice switch..OFF

(18) Engine and prop de-ice switches..OFF

(19) Pitot de-icing switch..OFF

(20) Windshield de-ice defog switch..OFF

(21) Wing flap handle..UP

(22) Water/methanol quantity gages..CHECKED

(23) Water/methanol arming switch..OFF

(24) Check warning lights with test button..CHECKED

(25) Press fire warning light test, lights on (E-inverter on)..CHECKED

(26) High pressure fuel cock..FUEL OFF

(27) Gust lock..ON

(28) Flight safety lock switch..NORMAL

(29) Propeller (check marks on spinner and blades)..FLAT PITCH (Ground Fine)

(30) Below flight fine pitch lock light..ON

(31) Flight fine pitch lock lights..ON

(32) Cruise lockout lights (press to test)..OUT

(33) Cruise pitch lights..ON

(34) Propeller brake (if installed)..OFF

(35) Propeller synchronizing system..OFF

(36) Fire pull T-handles..FULL FORWARD

(37) Fuel filter heat switches..AUTO

(38) Fuel filter heat lights..OUT

(39) Air start switches..OFF

(40) Throttle levers..IDLE

(41) Fuel trim..SET/CHECK

NOTE: Set fuel trim at 50% for OATs of +15°C or above or at 100% for OATs below +15°C.

(42) Fuel boost pump switches..ON

(43) Fuel pressure lights..OUT

(44) Oil pressure lights..ON

(45) Oil temperature indicators .... Above −30°C

(46) Beacon light .... ON

NOTE: Minimum oil temperature for starting is -30°C. Heating is required if below limit.

E. Prestart Checks

(1) Gearbox oil pressure warning check (when applicable): Advance throttle lever until warning light comes on; retard throttle lever, light should go out. (Aircraft having ASC 68B)

F. Auto feather check.

(1) Battery disconnect switches - indicator lamps out.

NOTE: For aircraft having ASC 192, batteries may be individually disconnected from dc system by the L & R BATT DISC switches. When the indicator lamp is on, applicable battery is disconnected.

(2) Battery - NORMAL

(3) Both HP Cocks - CRUISE LOCKOUT.

(4) Ensure gust lock engaged.

(5) Advance both throttle levers with left leading.

(6) Advance right throttle lever as soon as left propeller auto feathering confirmed.

(7) Retard left throttle lever as soon as non auto feathering of right propeller is confirmed.

(8) Advance left throttle lever as soon as right propeller auto feathering is confirmed.

(9) Retard right throttle lever to Idle as soon as non auto feathering of left propeller confirmed.

(10) Retard left throttle lever to idle left hand propeller auto feathering as soon as auto feathering confirmed.

(11) Press left and right feathering pump button until ground fine pitch alignment marks are realigned.

(12) HP Cocks - FUEL OFF.

G. Engine Start

(1) Throttle levers - IDLE

(2) Fuel datum (trim) - Set fuel trim at 50% for OATs of +15°C or above or at 100% for OATs below +15°C.

(3) Three generator switches - OFF

(4) Engine selector switch - LEFT / RIGHT

(5) Start selector switch - START

(6) Momentarily press engine START button.

(7) Engine RPM 1200 to 1500; HP cock - FUEL ON.

CAUTION: MONITOR TGT FOR MAX 930°C MOMENTARILY: IF TGT RAPIDLY APPROACHES 930°C, SHUT DOWN ENGINE. (DURING A NORMAL START, THE ENGINE WILL ACCELERATE SMOOTHLY, THE TGT RISING RAPIDLY AT FIRST, THEN MORE SLOWLY AS 700° TO 800°C ARE APPROACHED, FINALLY FALLING WITHIN IDLING LIMITATIONS.

FOR A SATISFACTORY START, AT LEAST 3500 RPM WITHOUT EXCESSIVE RISE IN TGT.

IGNITION AND STARTER DROPOUT LIGHTS - OUT

IF ENGINE RPM FAILS TO RISE ABOVE 2400 RPM IN 30 SECONDS SHUT DOWN ENGINE.

(8) At 3000 RPM, check that oil pressure indicator needle is off stop.

CAUTION: IMMEDIATELY SHUT DOWN ENGINE IF NO OIL PRESSURE AT 3000 RPM.

(9) Check oil temperature - maximum 1200C.

H. Engine Start Emergency Abort

(1) At 0 to 1200 RPM neutralize either start or selector switch.

(2) At 1200 to 3500 RPM HP Cock to FUEL OFF; neutralize either start or selector switch.

(3) At 3500 RPM and above HP Cock to FUEL OFF (it is not necessary to neutralize any switches at this point as all electrical and ignition is de-energized)

I. After Start Check

(1) External power switch (if used) .... OFF

(2) Oil pressure warning lights .... Out (above 5 to 6 psi)

(3) TGT .... 5500C Maximum

(4) Start selector switch .... OFF

(5) Engine selector switch .... OFF

(6) Fine pitch lock selector lever .... Check ground interlock

(7) Gust lock lever .... Checked ON

(8) HP fuel cock .... Move to CRUISE LOCKOUT

(9) Cruise lockout light .... ON

(10) Monitor Bus .... Energized

(11) Generator switch above 8000 RPM .... ON

(12) Alternator switch above 10,000 RPM .... ON

(13) Hydraulic pressure .... 1400-1500 psi

J. Leak Check

(1) Run engine for a minimum of 2 minutes at 8000 rpm.

(2) MAX TGT — 550°C idle speed.

(3) Oil temperature — 15 to 120°C above 11,000 rpm.

(4) Shut down engine.

(5) Check for leaks.

(6) Restart engine.

K. Full Throttle Maximum Power Check

CAUTION: RAPID ACCELERATION FROM MINIMUM CONSTANT SPEED RPM TO TAKEOFF RPM OR RAPID DECELERATION FROM TAKEOFF RPM TO MINIMUM CONSTANT SPEED RPM CAN PRODUCE HIGH PROPELLER BLADE STRESSES WHEN THE AIRCRAFT IS STATIONARY. OPEN OR CLOSE THE THROTTLE LEVERS SMOOTHLY AND PROGRESSIVELY THROUGH THESE RANGES SO THAT THE RATE OF MOVEMENT DOES NOT TAKE LESS THAN 5 SECONDS TO ACCOMPLISH; EXERCISE PARTICULAR CARE BETWEEN 14,000 AND 15,000 RPM.

(1) Set fuel datum position % (Step B. (10) above), for ambient ground running conditions __________%.

(2) Open throttle full and run engine for 4 minutes minimum but do not exceed 5 minutes to permit engine to stabilize.

(3) Check for 15,000 +50, – 0 RPM and record readings.

Left __________HZ Left __________RPM
Right __________HZ Right __________RPM

NOTE: See Engine Tachometer Indicator - Validation, Section 77-1-0. for frequency to RPM Conversion Chart to determine if tachometer indicator is within tolerance.

(4) Oil temperature should fall between 85°C to 100°C, record readings.

Left __________°C Right ___________°C

NOTE: For maximum data plate takeoff turbine gas temperature corrected for ambient conditions see See Figure 201.

For operating limitations takeoff turbine gas temperature corrected for ambient conditions See Figure 201.

(5) TGT shall not exceed operating limits for takeoff (dry) of 825°C and corrected data plate maximum take off TGT minus 5°C; record corrected TGT (Step B. (11) above) and observed readings for comparison.

(6) Run engine at an oil inlet temperature as near as possible to what mintorque was declared: record observed torque pressure on master gage and cross check with cockpit gage. Then correct to ISA-SL conditions using charts (refer to Rolls-Royce Dart Maintenance Manual, M-DA-7, Chapter 71)

- Observed Oil Temp Left _____________°C Right _____________°C
- Observed Torque Pressure Left _____________psi Right _____________psi

L. Set throttle for 12000 RPM and allow engine to stabilize and check for:

- Oil temperature between 55°C and 120°C.

  Observed Left __________°C Right _____________°C

- Oil pressure between 13 1/2 psi at 55°C and prorated down to 12 psi at 120°C and a maximum of 35 psi at any temperature.

  Observed Left _____________psi Right _____________psi

M. Ground idling RPM check

(1) Close throttle, set FUEL DATUM to FULL INCREASE (100%), RPM should be between 6500 and 7500.

Observed Left _____________RPM Right _____________RPM

N. Min Governor RPM Check.

(1) Set FUEL DATUM to 50% and operate throttle to obtain 50 psi torque pressure; vary trim between 25% and 75%; RPM should remain constant in the range between 10,850 and 11,150 RPM.

O. Fuel Heater Check

(1) Set throttle to obtain approximately 10,000 RPM and allow engine to stabilize; perform Step (a) or (b) as applicable:

(a) Mod 1397 (3 position valve) select half heat, a small reduction in RPM should be noted; select switch to OFF and RPM returns to original.

(b) Mod 1434 (2 position valve or 3 position valve with half heat unused) select full heat, a small reduction in RPM should be noted; select switch to OFF and RPM returns to original.

Observed Left _____________RPM Right _____________RPM

P. Synchronization Check (as required):

CAUTION: IF SYNCHRONIZATION CHECK IS REQUIRED, A TEMPORARY JUMPER WIRE MUST BE INSTALLED ACROSS 106-1 AND 106-2 OF PEDESTAL TERMINAL PANEL.

(1) Trim to ambient −30%.

(2) Run both engines.

(3) Synchronizer OFF.

(4) Gust Lock OFF.

(5) Monitor Bus Energized.

(6) Set engine RPM, L and R engines at 13,500 RPM.

CAUTION: GROUND INTERLOCK WILL BE IN FORWARD POSITION (FLIGHT POSITION) IN CASE OF EMERGENCY ENSURE THAT GROUND INTERLOCK IS MOVED TO GROUND POSITION WHEN THROTTLES ARE RETARDED.

(7) Turn synchronizer ON.

(8) Set left engine to 14,000 RPM; R ENG RPM should be approximately 13,900 RPM).

(9) Set left engine to 13,000 RPM; R ENG RPM should be approximately 13,100 RPM).

(10) Turn synchronizer OFF; note L ENG is at 13,000 RPM and R ENG engine is at 13,500 RPM.

R ENG Reading __________RPM

(11) Set L ENG to 12,000 RPM and R ENG to 15,000 RPM.

(12) Turn synchronizer ON; L ENG should be 12,000 RPM and R ENG engine 14,600 RPM.

(13) Apply full throttle to right engine and note RPM (should be approximately 15,000 RPM).

R ENG Reading __________RPM

(14) Turn synchronizer OFF; L ENG is 12,000 RPM and R ENG is 15,000 RPM.

(15) Set L ENG to 12,000 RPM; set R ENG to 12,000 RPM.

(16) Turn synchronizer ON; L ENG should be at 12,000 RPM and R ENG at 12,000 RPM.

(17) Turn synchronizer OFF.

(18) Move Flight Fine Pitch Lock Selector AFT.

CAUTION: CHECK FOR CORRECT PROPELLER INDICATION LIGHTS, TGT IN SAFE RANGE AND CORRECT RPM.

(19) Set Gust Lock ON.

Q. Water / Methanol Check

NOTE: Water/methanol check pressure tolerance is +7, −0 psi.

Corrections +6 psi/1000 feet BSL, −6 PSI/1000 feet ASL.

• Water/Methanol Check Press Data Plate	Left ______________psi	Right ______________psi
• Correction	Left ______________psi	Right ______________psi
• Corrected	Left ______________psi	Right ______________psi
• Observed	Left ______________psi	Right ______________psi

(1) Set fuel datum to 70% or less for ambient conditions.

(2) Place WATER/METHANOL switch to ARMED below 14,000 RPM

(3) Advance throttle lever slowly to 14,500 RPM and pause 10 seconds; no sudden increase in torque pressure and water/methanol should not come on.

NOTE: Water/methanol injection must occur between 14,500 and 14,900 RPM.

(4) Advance throttle lever slowly in increments of 100 RPM; Record RPM at which water/methanol cut-in (cut-in is noted by a substantial sudden increase in torque pressure).

- Observed Left ______________RPM Right ______________RPM

NOTE: Water/methanol should have cut-in before continuing further. If water/methanol does not cut-in, shut down engine and correct malfunction before continuing.

TGT shall not exceed 860°C with Water/Methanol.

(5) Open throttle to full and allow engine to stabilize; record torque pressure from master gage and cross check cockpit.

(6) Move throttle lever aft slowly to 14,500 RPM, pause for 15 seconds and check that water/methanol injection stops before 14,500 RPM; allow engine to stabilize.

- Observed Left _____________RPM Right _____________RPM

(7) Turn water/methanol off and observe torque pressure; no reduction in torque pressure is permissible.

- Observed Left _____________psi Right _____________psi

R. Electrical Operational Checks (at 11,000 RPM unless otherwise noted):

NOTE: Turn switches off after each check below.

- Windshield heat operation Checked_______
- Wing and tail de-ice operation Checked_______
- Propeller/Engine de-ice operation (at 12,000 rpm) Checked_______
- Temperature controls (using APU) Checked_______

S. Monitor bus actuation

(1) With both generators ON and main ac bus energized; L ENG 10,000 RPM or lower, R ENG 12,000 RPM or higher; turn monitor bus switch OFF; retard higher RPM engine and ensure following occurs:

- At approximately 10,500 RPM, main ac bus will drop off line.
- Advance throttle lever slowly; at approximately 10,400 to 11,000 RPM, main ac bus will come on line.

Observed Left _____________RPM Right _____________RPM

T. Gang Bar check.

(1) Select left and right generators and alternators and four boost pumps to ON.

(2) Pull gang bar firmly down then release, electrical system should be in EMER as follows:

- Both dc generators are tripped.
- Both ac alternators are tripped.
- Battery switch is in EMER position.

(3) Reset left and right generators and alternators.

(4) Select battery switch to NORM.

U. De-ice Boot Operation

(1) Perform Pneumatic De-ice System — Operational Test, see Section 30-1-0.

V. Shutdown Engine

(1) Close throttle and allow 2 minutes for engine conditions to stabilize.

(2) Propeller synchronizer system .... OFF

(3) Engine and prop deice switch .... OFF

(4) Water/methanol switch .... OFF

(5) Throttle lever position .... IDLE

(6) Generator switch .... OFF

(7) Alternator switch .... OFF

(8) Inverter switch .... OFF

(9) Select HP fuel cock .... FUEL OFF

(10) Fuel boost pump switch (After propeller stops)....................................OFF

WARNING: KEEP PERSONNEL CLEAR OF EXHAUST AREA AFTER ENGINE SHUTDOWN FOR AT LEAST 5 MINUTES, TO AVOID INJURY FROM POSSIBLE IGNITION OF RESIDUAL FUEL.

(11) If installed, remove temporary jumper installed across 106-1 and 106-2 of pedestal terminal panel to energize landing gear handle disabling switch.

(12) Inspect propellers, engine intakes and exhaust for condition and damage.

(13) Rotate engine compressor through one complete turn, inspecting for compressor damage.

(14) Check engine and accessory gearbox oil level and service if needed.

(15) If maintenance has been performed on accessory gearbox, check for leaks.

4. Engine — Removal / Installation

NOTE: The following is a list of special tools and equipment required for engine removal and installation. The Rolls-Royce tools and equipment listed below are for identification purposes only. Consult the Rolls-Royce manuals for details concerning tools and equipment for a particular engine. Tools equivalent to those listed may be used.

Part No.	Description	Part No.	Description
—	Ballast (as req.) 1500 Lbs.	159GT1058	Engine Sling
159GT1004-T1	Stand, Parking - Eng Change	—	Torque Wrench (0 - 1600 in-pounds)
CP.168213	Stand, Engine Transportation	159GT1009-5	Special Combination Wrench

WARNING: ELECTRICAL DISCHARGE FROM THE HIGH TENSION (H.T.) IGNITION UNITS IS POTENTIALLY LETHAL. THEREFORE THE LOW TENSION SUPPLY TO IGNITION UNITS MUST BE DISCONNECTED, AND 1 MINUTE ALLOWED TO ELAPSE BEFORE HANDLING ANY H.T. IGNITION CABLES.

BEFORE LOOSENING ENGINE MOUNTS IN THE AIRFRAME OR ATTEMPTING TO INSTALL AN ENGINE, ENSURE THAT ALL PERSONNEL EXCEPT THOSE ACTIVELY ENGAGED ON ENGINE LIFTING AND DISCONNECTING OR SECURING THE ENGINE MOUNTS ARE CLEAR OF THE AREA. UNDER NO CIRCUMSTANCES SHOULD PERSONNEL WORK UNDERNEATH THE ENGINE UNLESS ENGINE IS SECURE IN THE AIRFRAME.

CAUTION: MAINTAIN A PRECISE INVENTORY OF TOOLS, STORES, ETC., USED BY PERSONS ACTIVELY EMPLOYED ON, OR USED IN THE VICINITY OF THE ENGINE TO BE REMOVED OR INSTALLED. UNDER NO CIRCUMSTANCES MUST ANY ITEM BE ALLOWED TO ENTER THE ENGINE.

NOTE: For an engine being removed and prepared for shipment: ensure all components/accessories comprising the Quick Engine Change Kit (QEC) are removed and serial numbers are recorded.

A. Removal

(1) Remove engine cowl and top accessory section access panel.

(2) Apply electrical power and close main fuel shutoff valve (pull FIRE T-handle).

(3) Remove electrical power and disconnect batteries in nacelle battery compartments (or open all circuit breakers associated to engine systems/components).

(4) Distribute 1000 pounds of ballast in forward baggage compartment, entrance area, and cockpit to prevent adverse aircraft center of gravity movement; if both engines are to be removed while aircraft is jacked, distribute 1500 pounds. (Weight may be halved for single engine change with aircraft resting on main gear.)

(5) Remove propeller, see Propeller - Removal / Installation procedure, Section 61-0.

NOTE: Propeller may remain with engine in feathered position if sling hoist and engine parting stand can accommodate complete assembly.

(6) Disconnect firewall connections, with exception of engine mount bolts, (see Figure 202).

(7) Attach engine sling.

(8) Apply lifting force to relieve weight on mount bolts.

(9) Make final check on all disconnect points.

(10) Remove engine mount bolts.

CAUTION: WHILE REMOVING, GUIDE ENGINE ASSEMBLY TO MAINTAIN HORIZONTAL ATTITUDE AND PREVENT DAMAGE TO COMPONENTS. (EXHAUST CONE, ETC.)

(11) Remove engine.

(12) Inspect tailpipe blanket, see Section 78-0.

(13) Remove and inspect forward and aft tailpipe, see Section 78-0.

NOTE: If cracks are noted, see Section 78-0, for Tailpipe Cracks Limitations and Repair.

(14) Inspect inboard and outboard engine mount truss and diagonal truss tubes (Grumman Mounts) for chafing, corrosion, interference with tailpipe blanket, cracks, security and cleanliness.

NOTE: If indication of chafing is found on the diagonal truss tube (Aircraft not having ASC 180) refer to ASC 180.

If any damage is noted refer to CB 241C for wall thickness evaluation and possible repairs.

(15) Inspect chafe guard on diagonal truss tube (Aircraft having ASC 180 only). If chafing is found, replace chafe guard.

(16) Inspect fire sealing ring for indications of wear, damage and security.

(17) Remove Fuel Heater and Perform Fuel Heater - Pressure Check, see Section 73-1-0.

(18) Perform Firewall Receptacle - Inspection, see procedure in Section 54-0.

(19) Magnaflux inspect the following:

NOTE: If engine is removed for overhaul, the mount bolts, locking bolts and engine mount and struts are magnafluxed at the overhaul activity.

- Engine mount bolts.
- Engine special mount locking bolts.
- Mount attaching pins.
- Engine mount and struts.

B. Installation

NOTE: Installation procedure is for a complete Quick Engine Change (QEC) assembly.

If engine is newly overhauled, loaner or if switching engine from one side to the other, remove old engine calibration card from left circuit breaker panel cover.

The fuel heater, fuel temperature bulb and magnetic chip detector are not necessarily furnished with replacement engine. If units are not furnished, replacements should be installed before engine installation.

(1) Attach lifting sling to four attaching points on engine.

(2) Prepare engine for installation by removing all closures from fluid, air and electrical connectors.

CAUTION: GUIDE ENGINE CAREFULLY TO AVOID DAMAGE TO ALL FLUID LINES, AIR LINES, ELECTRICAL CONNECTORS AND TAILPIPE.

(3) Align engine assembly with nacelle firewall and move aft to attachment points. On aircraft not having ASC 220, water/methanol line should be aligned and inserted in respective fitting at this time.

(4) Lightly coat engine mount bolts with anti-seize compound, (Royco 81ms / MIL-T-83483).

(5) Install top engine mount bolts.

(6) Install lower engine mount bolts.

(7) Install special wrench P/N 159GT1009-5 (from combination wrench) onto torque wrench and torque all engine mount bolts to 1440 +120, −0 inch-pounds. Safety-wire bolts.

(8) Remove lifting sling.

(9) Connect following at firewall.

NOTE: All numbers in parentheses in below Steps are items in Figure 202.

(a) Accessory gearbox drive shaft (8).

(b) Thermocouple connector (2). Safety-wire and seal connector using Glyptol varnish or equivalent to prevent corrosion.

(c) Fuel drain line (3).

(d) Heat shield drain line (4).

(e) High tension ignition leads and safety-wire (13).

(f) Electrical connectors at nest. Connect starter and feathering pump cables to proper posts, auto feathering connector, pulse generator connector, pitch coarsening connector, flight safety lock indicator connector, increase pressure solenoid connector, third oil line solenoid connector, air intake and propeller de-icing connector, propeller hub switch connectors and left harness connector (16) to (28).

NOTE: Above electrical connectors are on low tension harness of QEC assembly, and must be safetied to one another.

(g) Fuel differential pressure and low pressure warning switch firewall electrical connector (15).

(h) Main fuel line. Connect flared tube nut of engine fuel line to firewall fitting; pressure check main fuel line assembly. Install hose shield halves, using clamp on forward end and nuts, bolts and washers at aft end (14).

NOTE: Seal between hose shield halves must be in a horizontal plane.

(i) Engine control rods at quick disconnects (12).

(j) Fire extinguisher line (11).

(k) Water/methanol line. Check for leaks after hookup (10).

(l) Fire sensing connectors (9).

(m) Engine breather (7).

(n) De-icer line (6).

(10) Lubricate engine controls, see procedure Section 12-0.

NOTE: Before installing propeller, inspect propeller hub switches.

(11) Install propeller, see Propeller - Removal / Installation procedure, Section 61-0.

(12) Install engine cowling.

(13) Fill out a new engine calibration card if the engine is a newly overhauled, loaner or if switching engine from one side to the other. The calibration card is located on the left circuit breaker panel cover. See Procedure for filling out Engine Calibration Card, Figure 207

NOTE: See Figure 205 and Figure 206 for example of completed card.

(14) Connect batteries.

(15) Open main fuel shutoff valve (push in FIRE T-handle).

(16) Bleed fuel system, see Section 28-0-1.

(17) Apply pressure to water/methanol system and check for leaks.

(18) Perform Engine Controls - Static Check, see Section 76-0.

(19) Perform Propeller - Inspection and Propeller System - Functional Test, see Section 61-0.

(20) Inspect area for foreign objects, security of all attachments.

(21) Secure cowl and close access openings.

(22) Perform Engine — Operational Test (leak check at idle RPM before maximum power check), see procedure this Section.

(23) Open cowl and visually check complete installation for leaks, connections, security of attachment and required safeties.

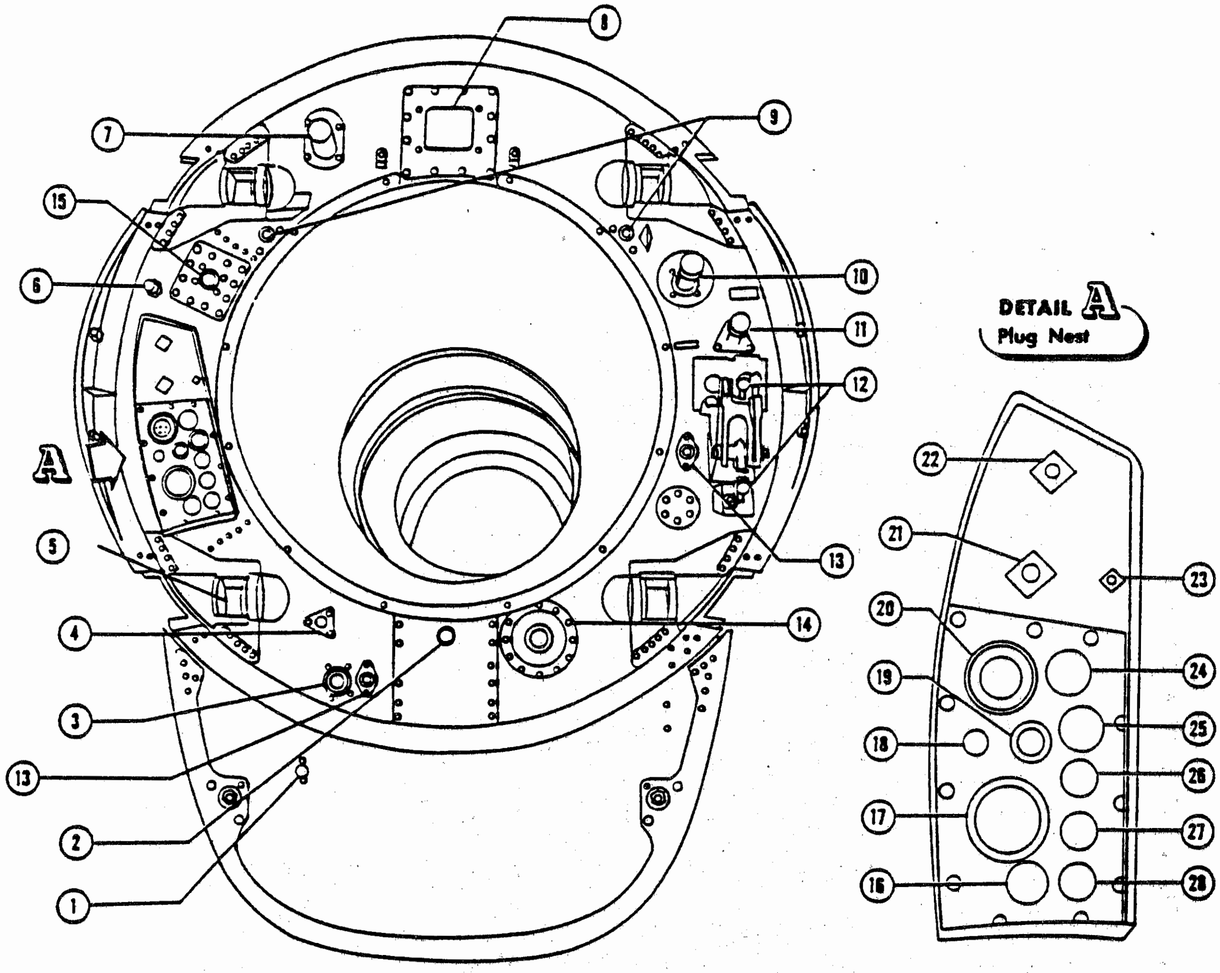
(24) Inspect for presence of foreign objects then close cowl.

PRESSURE ALTITUDE - FEET															
°C	-1000	-500	-250	SL	500	1000	2000	3000	4000	5000	6000	7000	8000	9000	10,000
21	3														
19	9														
17	15	0													
15	22	6	2	0											
13	28	12	8	6	2										
11	34	18	14	12	7	4									
9	40	23	20	17	13	9	1								
7	46	29	26	23	19	15	6	0							
5	52	35	32	29	24	20	12	4							
3	58	41	37	34	30	26	17	9	2						
1	65	47	43	40	36	31	23	15	7	1					
-1	71	53	49	46	41	37	28	20	12	6					
-3	77	58	55	52	47	42	33	25	17	11	4				
-5	83	64	61	58	53	48	38	30	22	15	9	4			
-7	89	70	67	63	58	53	44	35	27	20	14	8	2		
-9	95	76	72	69	64	59	49	40	32	25	18	13	6	2	
-11	101	82	78	75	70	64	54	45	37	30	23	17	11	6	1
-13	108	88	84	80	75	70	60	51	42	34	27	22	15	10	5
-15	114	94	90	86	81	76	65	56	47	39	32	27	20	14	9
-17	120	100	96	92	87	81	71	61	52	44	37	31	24	19	13
-19	126	105	102	98	92	87	76	66	57	49	42	36	29	23	17
-21	132	111	107	103	98	92	82	72	62	53	46	40	33	27	22
-23	138	117	113	109	104	98	87	77	67	58	51	45	38	32	26
-25	145	123	119	115	109	103	92	82	72	63	56	49	42	36	30
-27	151	129	125	120	115	109	97	87	77	68	60	54	47	40	34
-29	157	135	131	126	121	115	103	92	82	73	65	58	51	45	38
-31	163	141	136	132	126	120	108	97	87	77	70	63	55	49	42
-33	169	147	142	138	132	126	113	103	92	88	74	67	60	53	47
-35	175	153	148	144	137	131	119	108	97	87	79	72	64	57	51
-37	182	158	154	149	143	137	124	113	102	92	84	76	69	62	55
-39	188	164	160	155	148	142	129	119	107	96	88	81	73	66	59

SUBTRACT THESE AMOUNTS FROM DATA PLATE LIMITATIONS BEFORE COMPARING WITH OBSERVED RESULTS.

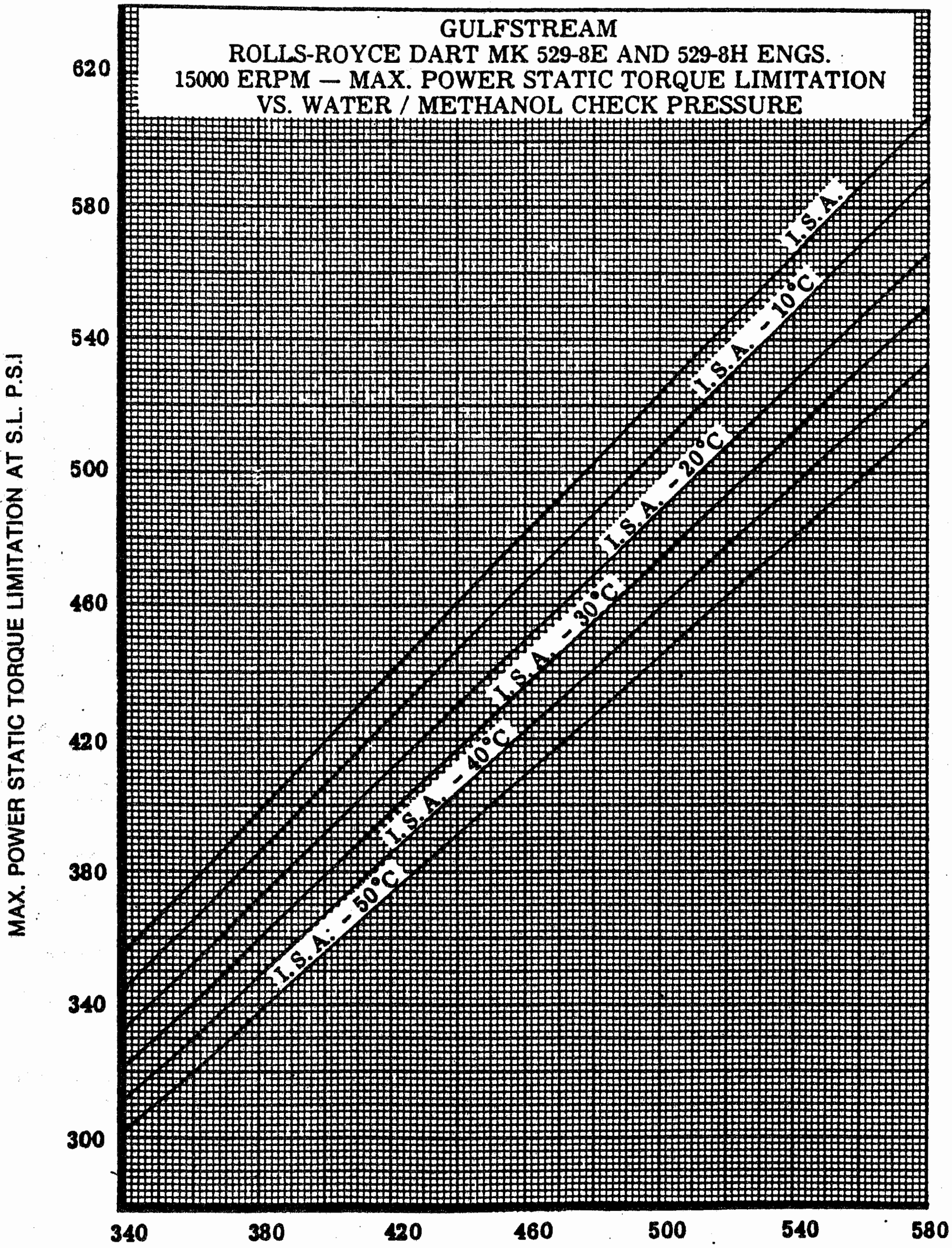
THE OBSERVED TGT MUST FALL BETWEEN THE ACTUAL DATA PLATE LIMITATIONS AT ATMOSPHERIC CONDITIONS IN THIS AREA.

Pressure Altitude — ft.
Figure 201.

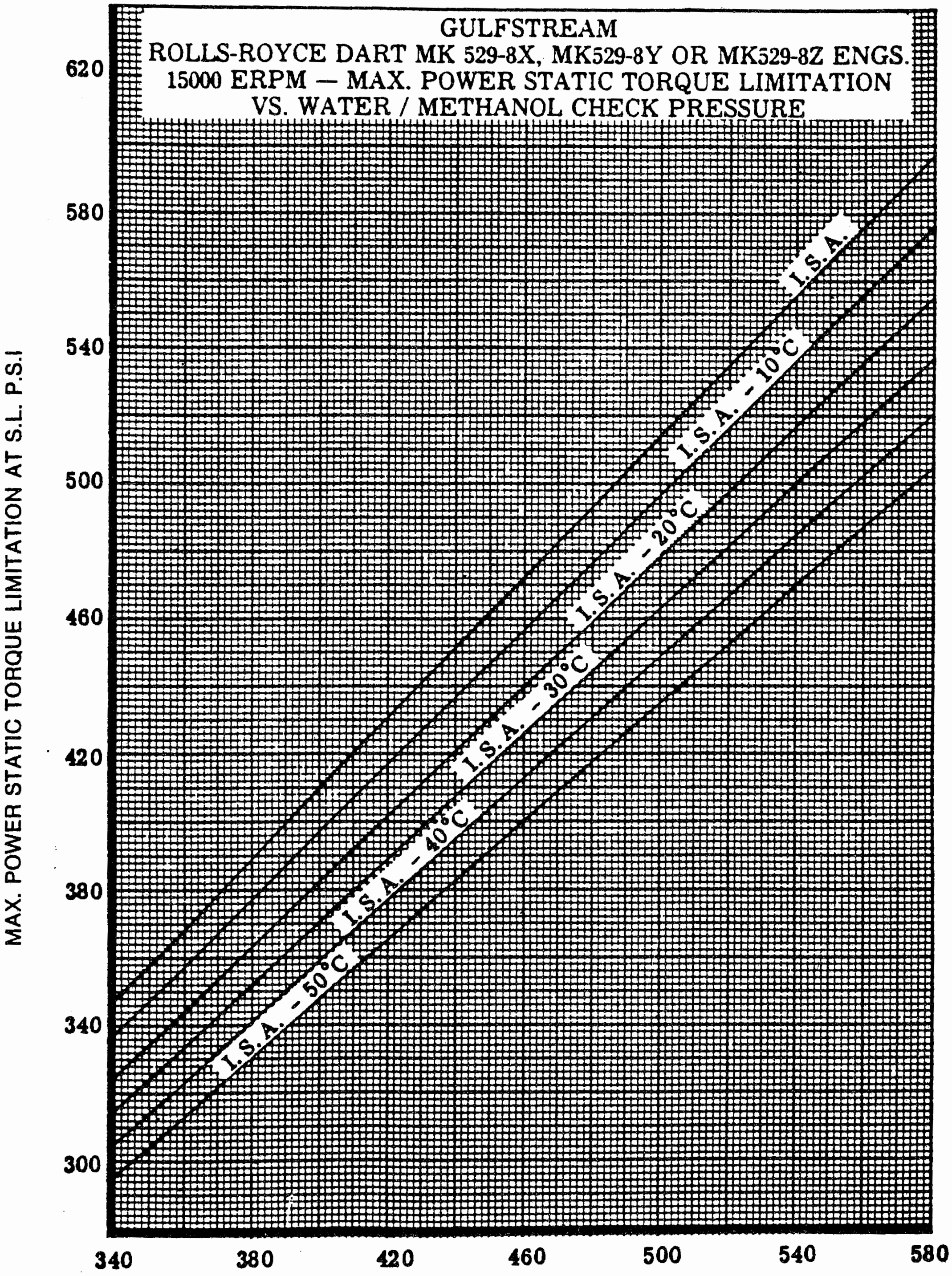


1. Engine Cowling Drain
2. Thermocouple
3. Main Engine Drain Line
4. Heat Shield Drain Line
5. Engine Mount (4)
6. De-icer
7. Engine Breather
8. Accessory Gearbox Drive Shaft
9. Fire Detection
10. Water/Methanol
11. Fire Extinguisher
12. Engine Controls
13. High Tension Ignition Lead
14. Main Fuel Line
15. Differential Pressure and Low Pressure Warning Switch Connector
16. 3rd Oil Line Connector
17. Air Intake De-icing and Propeller De-icing Connector
18. Propeller Hub Switch Connector (1 Lead)
19. Propeller Hub Switch Connector (2 Leads)
20. Port Harness, Oil Pressure Warning Light Switch, Hot Air Gate Valve, Oil Pressure Transmitter, Torque Pressure Transmitter, Oil Inlet Temperature Lead and Propeller Hub Switch (1 Lead) Connector
21. Positive Terminal Post-Starter Cable Connection
22. Positive Terminal Post-Feathering Pump Cable Connection
23. Negative Terminal Post-Starter Cable and Feathering Pump Cable Connection
24. Auto Feathering Switch Connector
25. Pulse Generator Connector
26. Pitch Coarsening Solenoid Connector
27. Flight Safety Lock Indicator Switch Connector
28. Increase Pressure Solenoid Connector

Firewall Connections
Figure 202.



Water/Methanol Check Pressure PSI
Figure 203.



Water/Methanol Check Pressure PSI
Figure 204.

GULFSTREAM ENGINE CALIBRATION CARD - ISSUE IV JUNE 1965						
ROLLS-ROYCE DART MARK 529-8E ENGINE						
TAKE-OFF AND MAX CONTINUOUS POWER @ I.S.A. SEA LEVEL						
ENGINE SERIAL NUMBERS (L) 16000 (R)						
RATING	**RPM**	**SHP**	**TGT °C**	**TORQUE PSI**	**TGT °C**	**TORQUE PSI**
NOMINAL DRY POWER	15000	1990	780	-		
MIN. GUAR. DRY POWER	15000	1910	761			
MIN.GUAR. WETPOWER	15000	1950	-	385	-	
ENG. PWR. CK. PRESS	15000	1950	-	370	-	
W/MCP VARIATION WITH ALT: MINUS 6 PSI/1000 FEET ABOVE S.L						
MAX. POWER STATIC TORQUE LIMITATIONS @ SEA LEVEL						
MAX. POWER @ ISA	15000	2040	-	402	-	
MAX. POWER @ ISA-10°C	15000	1975	-	389	-	
MAX. POWER @ ISA-20°C	15000	1905	-	376	-	
MAX. POWER @ ISA-30°C	15000	1845	-	364	-	
MAX. POWER @ ISA-40°C	15000	1790	-	353	-	
MAX. POWER @ ISA-50°C	15000	1735	-	342	-	
STATIC TORQUE CORRECTION: MINUS 5 PSI/1000 FEET ABOVE S.L.						

Card shown for -8E engine having:

Max. Power Temp.	= 780°C
Min. Power Temp.	= 761°C
Water/Methanol Check Pressure	= 385 PSI
Engine Power Check Pressure	= 370 PSI

EXAMPLE No.1 DART MK529-8E AND 8H
Figure 205.

GULFSTREAM ENGINE CALIBRATION CARD - ISSUE IV JUNE 1965						
ROLLS-ROYCE DART MARK 529-8X ENGINE						
TAKE-OFF AND MAX CONTINUOUS POWER @ I.S.A. SEA LEVEL						
ENGINE SERIAL NUMBERS (L) 16000 (R)						
RATING	RPM	SHP	TGT °C	TORQUE PSI	TGT °C	TORQUE PSI
NOMINAL DRY POWER	15000	1990	780	-		
MIN. GUAR. DRY POWER	15000	1910	761			
MIN. GUAR. WET POWER	15000	1990	-	393	-	
ENG. PWR. CK. PRESS	15000	1950	-	370	-	
W/MCP VARIATION WITH ALT: MINUS 6 PSI/1000 FEET ABOVE S.L.						
MAX. POWER STATIC TORQUE LIMITATIONS @ SEA LEVEL						
MAX. POWER @ ISA	15000	2040	-	403	-	
MAX. POWER @ ISA-10°C	15000	1975	-	390	-	
MAX. POWER @ ISA-20°C	15000	1905	-	375	-	
MAX. POWER @ ISA-30°C	15000	1845	-	364	-	
MAX. POWER @ ISA-40°C	15000	1790	-	353	-	
MAX. POWER @ ISA-50°C	15000	1735	-	343	-	
STATIC TORQUE CORRECTION: MINUS 5 PSI/1000 FEET ABOVE S.L.						

Card shown for -8X engine having:

Max. Power Temp.	= 780°C
Min. Power Temp.	= 761°C
Water/Methanol Check Pressure	= 393 PSI
Engine Power Check Pressure	= 370 PSI

EXAMPLE No.2 DART MK529-8X, 8Y AND 8Z
Figure 206.

GULFSTREAM ENGINE CALIBRATION CARD - ISSUE IV JUNE 1965						
ROLLS-ROYCE DART MARK 529-8 ENGINE						
TAKE-OFF AND MAX CONTINUOUS POWER @ I.S.A. SEA LEVEL						
ENGINE SERIAL NUMBERS			(L)		(R)	
RATING	RPM	SHP	TGT °C	TORQUE PSI	TGT °C	TORQUE PSI
NOMINAL DRY POWER	15000	1990		-		
MIN. GUAR. DRY POWER	15000	1910				
MIN. GUAR. WET POWER	15000	1990	-		-	
ENG. PWR. CK. PRESS	15000	1950	-		-	
W/MCP VARIATION WITH ALT: MINUS 6 PSI/1000 FEET ABOVE S.L.						
MAX. POWER STATIC TORQUE LIMITATIONS @ SEA LEVEL						
MAX. POWER @ ISA	15000	2040	-		-	
MAX. POWER @ ISA-10°C	15000	1975	-		-	
MAX. POWER @ ISA-20°C	15000	1905	-		-	
MAX. POWER @ ISA-30°C	15000	1845	-		-	
MAX. POWER @ ISA-40°C	15000	1790	-		-	
MAX. POWER @ ISA-50°C	15000	1735	-		-	
STATIC TORQUE CORRECTION: MINUS 5 PSI/1000 FEET ABOVE S.L.						

Line 1 - Enter correct designation for the engine, either MK529-8E, MK529-8H, or MK529-8X, (R-R Mod 1365), MK529-8X, (R R Mod A 1814), MK529-8Y or MK529-8Z.
Line 2 - Fill in engine serial numbers.
Line 3 & 4 - The "Nominal Dry Power" and the "Minimum Guaranteed Dry Power". ratings are the and maximum power temperature and the minimum power temperature, respectively for each engine. These are obtained from the interconnection data listed on the engine inspection and test certificates in the engine logbooks.
Line 5 - The "Minimum Guaranteed Wet Power" of 1950 shaft horsepower for the MK529-8E and MK529-8H or 1990 shaft horsepower for the MK529-8X, (RR Mod 1365), MK529-8X, (RR Mod No. 1814), MK529-8Y or MK529-8Z must be entered in the SHP column. The "Minimum Guaranteed Wet Power" rating is the water/methanol check pressure in pounds per square inch at 100°C oil inlet temperature. This is obtained from the engine inspection and test certificates in the engine logbook.
Line 6 - Fill in the "Engine Power Check Pressure". This is obtained from the engine inspection and test certificates in the engine logbook, and corresponds to the power check pressure at 1950 shaft horsepower and 70°C oil inlet temperature for the MK529-8E, MK529-8H, MK529-8X, (RR mod 1365), MK 529-8X (RR mod 1814), MK529-8Y and MK529-8Z engines.
Line 7 thru 12 - Maximum powers at ISA and below are obtained by the use of Figure 203 for the MK529-8E, through MK529-8H and Figure 204 for the MK529-8X (RR mod 1365 and RR mod No. 1814), MK529-8Y or MK529-8Z. Start with the known water/methanol check pressure for the given engine and read vertically to intersect the appropriate ISA line. Reading horizontally from this intersection will give the "Maximum Power Static Torque Limitation at Sea Level".

Blank Engine Calibration Card
Figure 207.

ENGINE MOUNTING SYSTEM — DESCRIPTION

1. General

The engine is mounted to the airframe structure by a welded steel W shaped truss (see Figure 1) which attaches to the engine at three points; at the top and each side engine mounting foot. Engine mounting feet are located between the first and second stage compressors at the three, nine and twelve o'clock positions. The four rear attaching points of the engine mount mate with four fittings on the firewall. There are no vibration dampening fittings incorporated between the engine and mount, or between the mount and airframe. Forward ends of the lugs and split rings. Engine mount lugs and split rings are secured by hollow retaining bolts, tab washers and locking bolts. Lower right and left single engine mount tubes each have attachment fitting welded at aft end. Upper engine mount tubes are joined in pairs by two Y shaped attachment fittings and form the upper right and left attachment points. Four rear eye attachment fittings mate with four firewall fork fittings and are secured by the four engine mount attaching pins (bolts). Pin assemblies are made up of three parts; the special nut, split bushing and tapered inner pin. With the holes in the eye and fork fittings aligned, the engine mount attaching pins are inserted (using special tool 159GT1009) and the special nuts are threaded to the fork fitting threads. When the special nut is drawn up, the tapered pin is moved, thus causing the split bushing to expand, tightening the attachment joint.

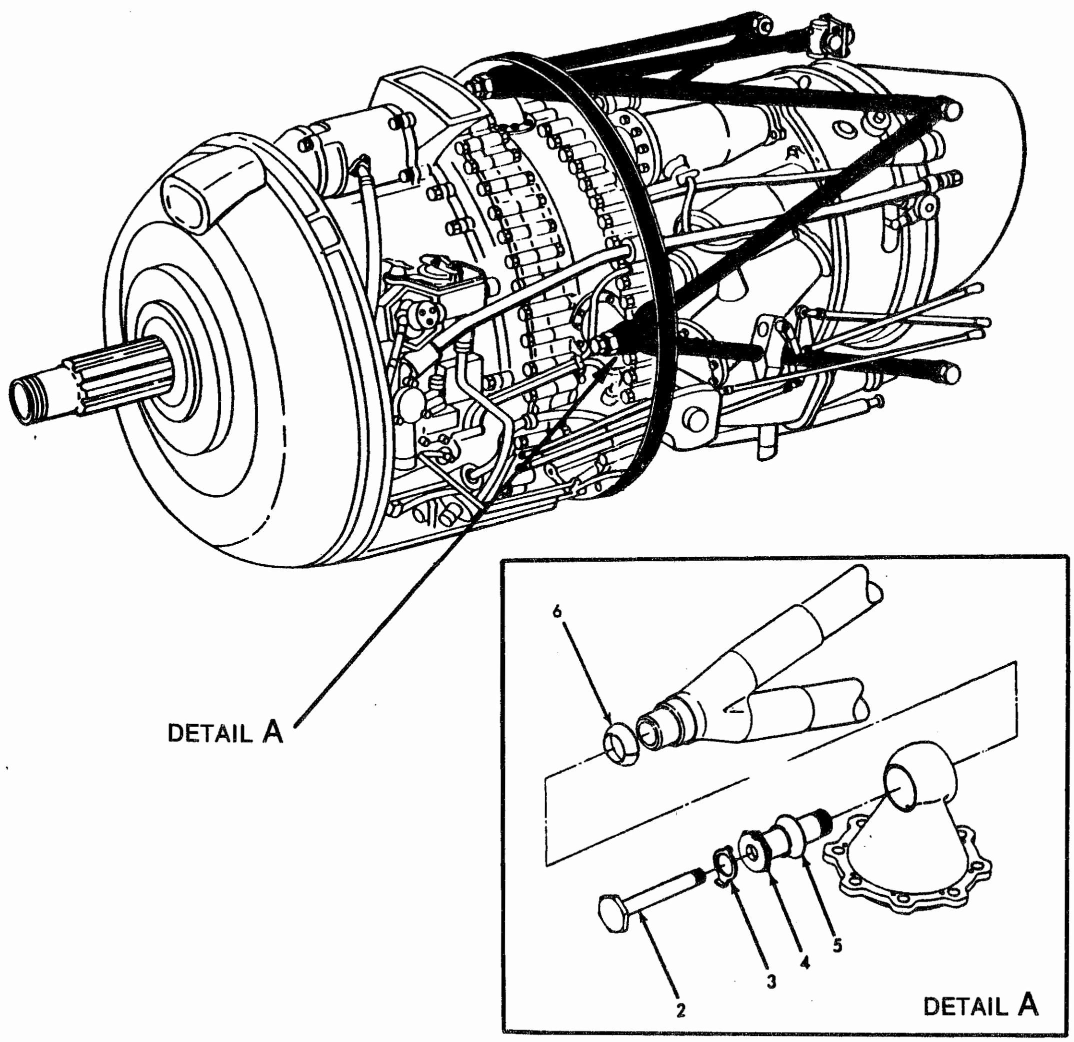


Figure 1

ENGINE MOUNTING SYSTEM — MAINTENANCE PRACTICES

1. **Engine Mount — Removal / Installation (Vickers)**
   A. Removal
      (1) Remove engine from aircraft.
      (2) With engine supported by sling, 159GT1058, perform the following:
         (a) Remove all fluid lines attached to engine mount tubes (water / methanol line at front of fire seal).
         (b) Remove all electrical harnesses and conduits attached to engine mount tubes.
         (c) Remove all air lines attached to engine mount tubes and disconnect de-icer lines attached to left hand side of second stage compressor.
         (d) Remove all fire sensing leads attached to engine mount tubes.
         (e) Remove engine breather line at forward hose connection.
         (f) Remove fuel heater clamp from engine mount tube.
         (g) Disconnect throttle rod and high-pressure fuel cock at engine control box.
         (h) Unlock tab washers on locking bolts at engine mount lugs.

         CAUTION: LOCKING BOLTS HAVE LEFT HAND THREADS.

         (i) Remove locking bolts and tab washers from hollow retaining bolts.

         CAUTION: WHEN HOLLOW BOLTS AND WASHERS ARE REMOVED, ENGINE MOUNT MUST BE HELD IN POSITION SO AS NOT TO DAMAGE ENGINE COMPONENTS.

         (j) Remove hollow retaining bolts.

         CAUTION: CARE SHOULD BE TAKEN WHEN REMOVING THE MOUNT FROM THE ENGINE SO THAT NO COMPONENTS, FLUID LINES OR ELECTRICAL LEADS ARE DAMAGED.

         (k) Remove engine mount from engine by pulling aft until the Y shaped fittings are clear of engine lugs.
         (l) Install engine in shipping stand.
         (m) Install protective covers where applicable.
   B. Installation
      (1) Install engine sling, 159GT1058 and lift engine clear of shipping stand.
      (2) Install split rings (dip in MIL-G-3278A) in grooves of engine mount Y shaped fittings.
      (3) Position engine mount Y-shaped fittings in engine lugs.
      (4) Place concave washers on hollow retaining bolts so that boltheads mate with concave surfaces.
      (5) Install hollow retaining bolts (dip in Turbo Oil) and washers and torque to 600 inch-pounds.
      (6) Place tab washers so that one tab is in locking hole of each hollow bolt.

      CAUTION: LOCKING BOLTS HAVE LEFT HAND THREADS.

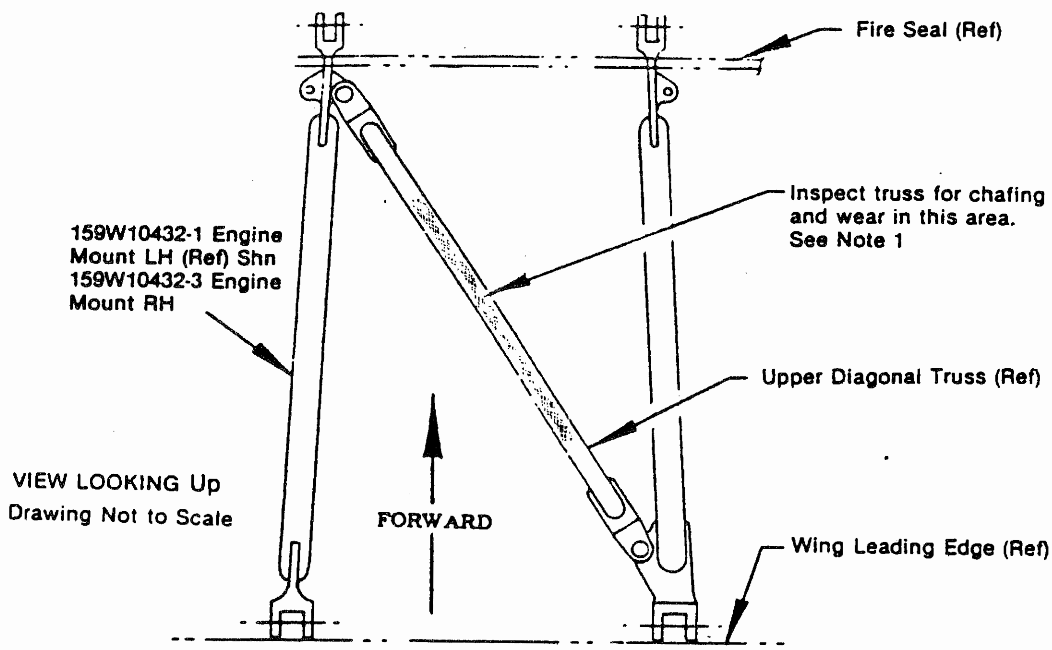
      (7) Install locking bolts (dip in Turbo Oil) and torque to 300 – 420 inch-pounds. Bend tab to safety.
      (8) Mount engine on QEC stand, 159GT1004-T1.

3. **Engine Mount Chafe — Inspection**
   A. For aircraft having ASC 180.
      (1) Inspect engine mount for signs of chafing, damage and security.
      (2) Replace chafe guard if chafing is evident.

      NOTE: This Step complies with requirements of AD 67-17-5.

   B. For aircraft not having ASC 180, (see Figure 201).
      (1) Perform a thorough inspection of engine mount for evidence of chafing or interference with engine tailpipe blanket.

      NOTE: If chafing is evident, see ASC 180 for recommended instructions / repairs.



Inspection of Upper Diagonal Truss for Chafing
Figure 201

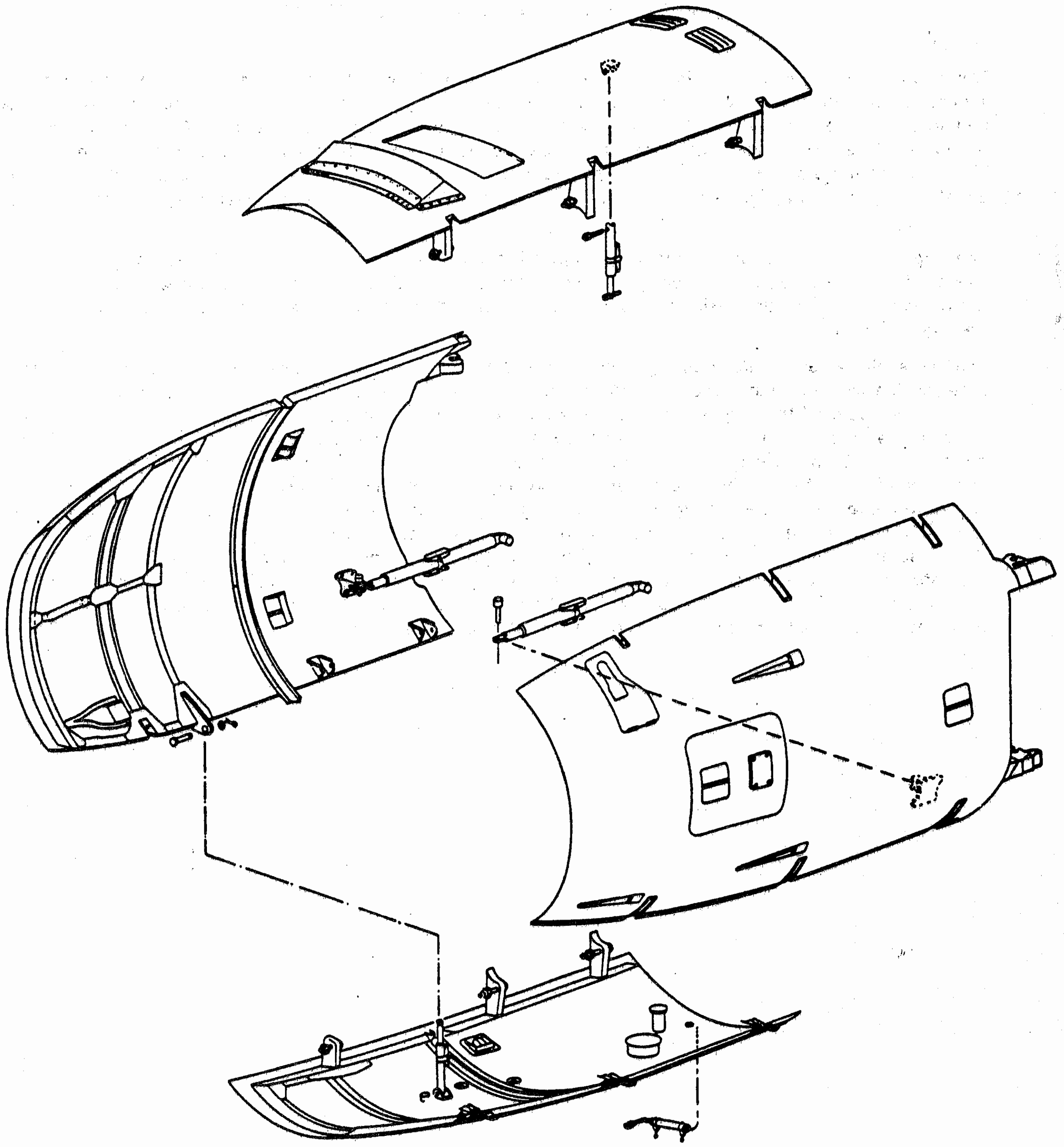
ENGINE COWLING — DESCRIPTION / OPERATION

1. Description

The engine cowling is of the clamshell type (See Figure 1), with the top and left side panels forming one section of the shell and the bottom and the right side panels forming the other. The forward edge of the engine cowl panels mates with the aft edge of the air intake casing. The aft edges of the engine cowl panels extend over the firewall and mate with the nacelle skin to form a smooth unrestricted surface. The top panel is hinged to the left side panel at three points and is held in the open position by a telescoping stay. The bottom panel is hinged to the right side panel in the same manner and also has a telescoping stay to hold it in the open position.

The engine cowl panels incorporate Vee shaped fireseal channels which mate with and compress the engine fireshield's fire-resistant rubber seal gasket when the cowl panels are closed and secured. The arrangement of the annular shield, the fire-resistant rubber seal gasket and the fireseal channels of the cowl panels form a fire and liquid tight seal.

The two shell halves are secured together when the cowling is closed by means of six push-pull latches, three between the left side panel and the bottom panel, and three between the right side panel and the top panel. The forward edge of the engine cowl panels is secured to the air intake casing by means of six lock fasteners, three for each shell half. The two side panels are attached to the nacelle welded truss which has four hinge points just aft of the firewall. When either half of the cowling is opened, it is held in position by a telescoping stay which is attached to the side panel and the engine mount tube. The entire shell half, either right or left, can be removed by disconnecting the telescoping stay at the side panel (remove pip pin at outboard end of rod), and then by lifting up on the shell half until the hinge points part. The top cowl panel has a small fairing which mates with the oil cooler air intake, and an opening for the oil cooler exit duct. The right side panel incorporates an access door for the oil filler cap and dipstick. both right and left cowl side panels have cooling air inlets for fire zone No. 1 and zone No. 2. The lower cowl incorporates a drain system which collects fluids from engine. Fluid is then routed to the drain collector tank in aft nacelle.



Engine Cowling Assembly
Figure 1.

ENGINE COWLING — MAINTENANCE PRACTICES

1. Engine Air Intake Cowling — Removal / Installation

A. Refer to Rolls-Royce Dart Maintenance Manual M-Da7-G.

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POWER PLANT FLUID DRAINS — DESCRIPTION / OPERATION

1. Description

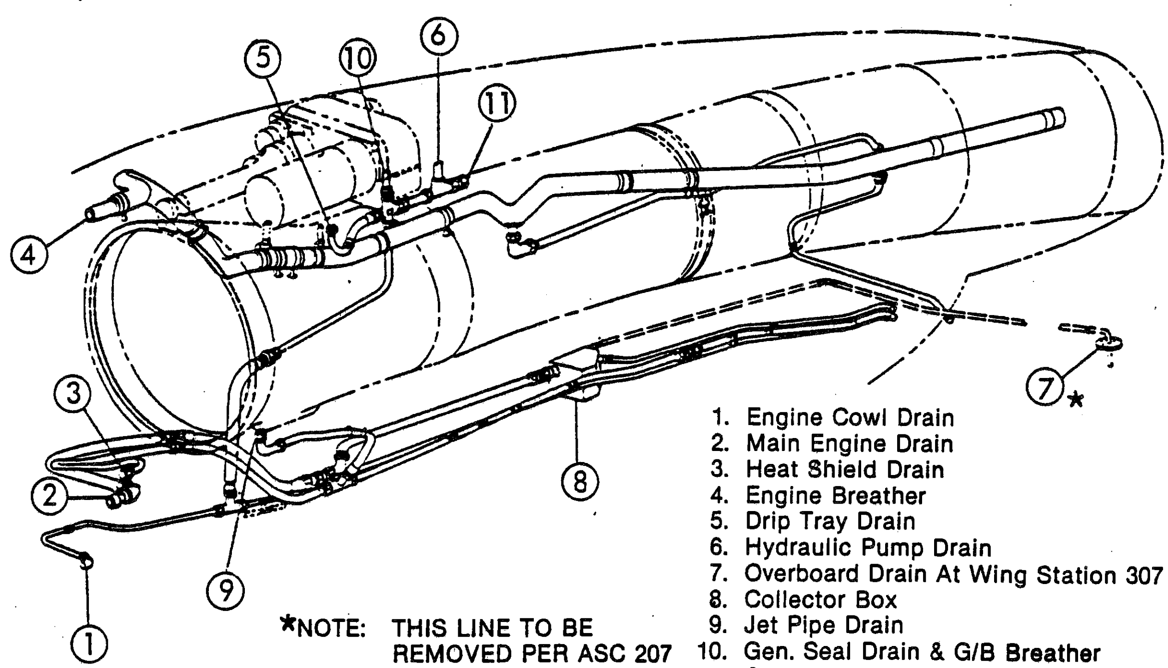
Fluid from the flow control unit, fuel pump, water methanol control unit, compressor casing, combustion chambers and nozzle box is discharged through individual lines to the drain valve on each engine. This drain valve is located on the No. 5 combustion chamber just aft of the engine fireshield. A single line from the drain valve runs aft to the fuel collector tank in the respective wheel well. See Figure 1 and Figure 2. On Aircraft not having ASC 207, this fluid drains through a line running parallel to the nacelle longeron and then outboard at rear spar to be discharged under the surface of the wing, inboard of Wing Station 307. On Aircraft having ASC 207, the fluid is stored in the tank until drained by maintenance personnel. A valve is provided in the bottom of each tank by ASC 207, for the controlled draining and removal of fluids drained into this tank.

Teed to the engine cowl drain aft of the firewall are the following:

- Hydraulic Pump Drain
- Gearbox Drip Tray Drain
- DC Generator Seal Drain
- Gearbox Breather

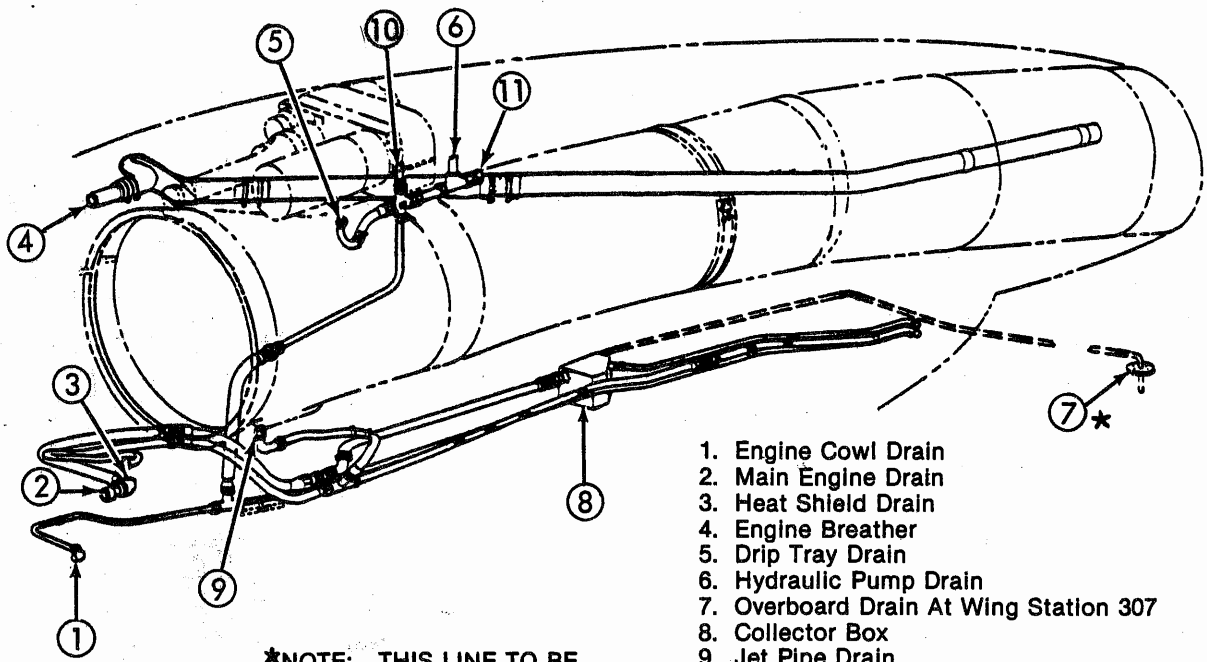
The cowl drain runs parallel to the heat shield drain and discharges overboard at the rear of the nacelle.

On Aircraft 1 - 144, 322 and 323 the engine breather line has a Vee sump drain that discharges overboard at the rear of the nacelle. On Aircraft 145 - 200 and Aircraft 1 - 144, 322 and 323 having ASC 173, the Vee sump and sump drain have been eliminated.



1. Engine Cowl Drain
2. Main Engine Drain
3. Heat Shield Drain
4. Engine Breather
5. Drip Tray Drain
6. Hydraulic Pump Drain
7. Overboard Drain At Wing Station 307
8. Collector Box
9. Jet Pipe Drain
10. Gen. Seal Drain & G/B Breather
11. Cabin Blower Drain (R. Side Only)

Power Plant Fluid Drain Lines and Breather
Aircraft 1 - 144, 322 and 323 Not Having ASC 173
Figure 1.



1. Engine Cowl Drain
2. Main Engine Drain
3. Heat Shield Drain
4. Engine Breather
5. Drip Tray Drain
6. Hydraulic Pump Drain
7. Overboard Drain At Wing Station 307
8. Collector Box
9. Jet Pipe Drain
10. Gen. Seal Drain & G/B Breather
11. Cabin Blower Drain (R. Side Only)

*NOTE: THIS LINE TO BE REMOVED PER ASC 207

Power Plant Fluid Drain Lines and Breather
Aircraft 145 - 200 and 1 - 144, 322 and 323 Having ASC 173
Figure 2.

FUEL TRIMMER — DESCRIPTION

1. **Description**

A fuel trimmer motor is installed in each nacelle and is connected to the engine control box through a series of push-pull rods and bellcranks. A fuel trimmer switch for each engine is installed on the center pedestal console panel. A dual gage fuel trimmer datum indicator is installed on the lower portion of the center instrument panel. On hot day, it is necessary to lessen (or trim) the amount of fuel delivered to the engine to prevent the turbine gas temperatures from exceeding the allowable limits. The loss of power accompanying this operation is regained by use of water / methanol injection. The amount that the fuel is trimmed is governed by atmospheric conditions. After engines are started and before taxi, compensation is made for ambient conditions by means of a computer or a fuel trimmer position chart.

The purpose of this system is control and indication of the fuel to air mixture as suppled to the engines. A switch for each actuator mounted in the pedestal allows the selection of a weak or rich mixture. The actuator in addition to trimming the fuel mixture, mechanically acts upon the transmitter. The transmitter (two, one mounted on left side of each nacelle) consists of a toroidal resistor, over which two contacts move. The toroidal resistor is tapped at 3 places, 120° apart and is supplied with 24V dc. The three taps are routed to the indicator which comprises a two pole permanent magnet motor, pivoted to rotate within a soft iron stator. When the actuator (two, one mounted on left side of each nacelle), acts upon the transmitter, it causes the two contacts to move around the toroid. This in turn causes the currents through the toroid to change. The change in current causes a change in the magnetic field of the indicator stator, which is manifested by a change in the indicator needle position. Both fuel trim and fuel trim indication are available in emergency dc operation.

A. Effect of Ambient Air Temperature

The DART power output is effected by ambient temperature, with the power developed being a function of the temperature drop across the turbines. As ambient temperature decreases engine efficiency, power and fuel flow increase while maintaining the same turbine inlet temperature. In the case of the Dart, however, an ambient temperature is reached where the flow control unit delivers its maximum calibrated flow, with the result being a decrease in engine power output at a constant fuel flow below that temperature. For takeoff power, this change occurs at sea level on a standard day and at 10,000 feet on an ISA 12°C day. To determine whether the engine is developing its correct takeoff power at ambient temperatures where the maximum fuel flow has been reached, (100% fuel trim), TGT corrections are provided which are applied to the maximum power temperature of the engine. These TGT corrections are a function of ambient temperature and pressure and are covered in Chapter 77.

At ambient temperatures where the maximum calibrated fuel flow would result in excessive turbine temperatures a fuel trimer is provided which allows fuel to be reduced, thus maintaining TGT at acceptable limits without altering rpm. Under these conditions engine efficiency, fuel flow and power are reduced. The fuel trimmer is operated from the pedestal by an electric fuel datum control in conjunction with a position indicator. To ensure that satisfactory takeoff power is obtained at ambient conditions where fuel trim position in percent as a function of ambient temperature and pressure.

With the fuel datum set at FULL INCREASE on the indicator gage, there is no reduction in fuel flow. When the fuel datum is set at some lower figure, the fuel flow is proportionately reduced or trimmed, at all settings of the pilots throttle lever. At fuel datum FULL DECREASE (0 indicator reading), fuel flow at any given throttle setting is reduced by approximately 20 percent from that at fuel datum FULL INCREASE (100 indicator reading).

With the fuel datum preset to a position appropriate to the ambient conditions a satisfactory TGT is automatically obtained on opening the pilots throttle lever for takeoff and initial climb. In flight, the fuel datum is adjusted to maintain a TGT approaching the maximum permissible, for the reasons given previously.

The fall in air mass flow under hotter than ISA conditions, also means that less power is produced by the turbine. It is a characteristic of a compressor matched to a turbine that a reduction in mass flow is accompanied by a reduction in the pressure rise achieved by the compressor. It follows that with the same turbine entry temperature there is on a hot day less pressure and temperature drop across the turbine. This results in a further reduction in turbine power output. While power absorbed by the compressor also falls, this fall is less than the reduction in turbine power and there is, therefore, a decrease in propeller shaft horsepower.

For takeoff purposes, nominal power can be restored by injecting water / methanol. This mixture cools the air at the inlet to the compressor, restoring density and mass flow and enables the methanol burnt in the combustion chambers to restore power.

B. Setting the Fuel Datum

When ambient conditions require fuel trimming to keep the TGT within limits, use is made of the fuel datum position indicator to set the trimmer after start up and before opening up the engine. For ground running, ascertain the fuel datum position appropriate to the ambient conditions. The Rolls-Royce computer is available for pilot use only. With fuel datum set to the appropriate position for the prevailing conditions, the engine will give an acceptable TGT and power when throttle is opened fully.

To enable fuel datum indicator to be used for controlling engine performance at widely varying ambient conditions, it is essential that the controls should have been correctly adjusted. For consistent operation, trimmer should always be set from decrease towards increase. Atmospheric conditions may change during a ground run; it is important that frequent checks of the prevailing temperature and pressure altitude are made to ensure that fuel datum position selected is correct for the ambient conditions.

C. Fuel Trim Computer

Rolls-Royce computer R.K. 28677 for Pre Mod 1371 engines or computer R.K. 40916 for Mod 1371 engines, is furnished to the pilot for determining his fuel trim datum position for takeoff and for landing (set for ambient condition). This computer consists of three scales, a temperature scale and a pressure altitude scale which, when aligned, give a resultant in fuel datum position in tenths, which is the third scale.

D. Fuel Trimmer Position Chart (See Figure 1 for Pre Mod 1371 or Figure 2 for Mod 1371 engines)

The procedure for reading the fuel trimmer position chart is as follows:

Example 1: A pressure altitude of 1000 feet and a temperature of +27°C. Start at bottom at 1000 feet and continue vertically upward until we intercept a horizontal line from the centigrade temperature (+27°C) line going toward the right. At this intersection read the trimmer position figure, which is 70.

Example 2: Sea level (SL) and a temperature of +25°C. Start at SL and proceed up until we intercept a horizontal line from +25°C. At that point of intersection our figure is 79.

Example 3: 3000 foot pressure altitude and a temperature of +1°C. Start at 3000 feet and proceed up until we intercept a horizontal line from +1°C. At the intersection we are in the full increase area (100°), consequently, no trim-off is necessary.

NOTE: Rule of thumb: At sea level (SL), temperatures below ISA (+15°C), do not require trim off.

E. Emergency DC Operation

Both fuel trim and fuel trim indication are available in emergency dc operation.

F. Use of TGT Correction Chart (See Figure 3)

The procedure for using the TGT correction chart is as follows:

Example 1: A 3000 foot pressure altitude and a temperature of +1°C. Start at 3000 feet and proceed upward until we intercept a horizontal line from +1°C. At the intersection we read the TGT correction of +14°C. This correction is then subtracted from the engine maximum power temperature to give the corrected maximum power temperature for the prevailing ambient conditions.

Example 2: At sea level and a temperature of +19°C. At the intersection the reading is in the zero range. Therefore, fuel trimming will be necessary to maintain the engine at its maximum power temperature. For these ambient conditions refer to either Figure 1 (Pre Mod 1371) or Figure 2 (Having Mod 1371) will determine the correct fuel trim position.

Figure 1, Pre Mod 1371 — 92%
Figure 2, Having Mod 1371 — 93%

Ground checks should not be carried out at atmospheric conditions in this area.

FULL INCREASE

PREVAILING AIR TEMPERATURE — °C	− 1000	− 500	− 250	SL	+ 500	+ 1000	+ 2000	+ 3000	+ 4000	+ 5000	+ 6000	+ 7000	+ 8000	+ 9000	+ 10,000
+ 51	40	33													
+ 49	44	37	34	30	28										
+ 47	49	41	38	34	32	29									
+ 45	53	45	42	38	36	33	29								
+ 43	57	50	46	42	40	37	33	28							
+ 41	61	54	51	47	44	42	37	32	27						
+ 39	66	58	55	51	48	46	41	36	31	26					
+ 37	70	62	59	55	52	50	45	40	34	29	24				
+ 35	74	66	63	59	57	54	49	44	38	33	28	23			
+ 33	78	71	67	63	61	58	53	47	42	37	32	26	21		
+ 31	82	75	71	67	65	62	57	51	46	41	35	30	25	20	
+ 29	87	79	75	71	69	66	61	55	50	45	39	34	28	24	18
+ 27	91	83	80	75	73	70	65	59	54	49	43	38	32	27	22
+ 25	95	87	84	79	77	74	69	63	58	53	47	42	36	31	25
+ 23	100	91	88	83	81	78	73	67	62	56	51	45	40	35	29
+ 21		95	92	87	85	82	77	71	65	60	54	49	43	38	33
+ 19		100	96	92	89	86	81	75	69	64	58	53	47	42	36
+ 17			100	96	93	90	84	79	73	68	62	57	51	46	40
+ 15				100	97	94	88	83	77	72	66	60	55	49	44
+ 13					100	98	92	87	81	75	69	64	58	53	47
+ 11						100	96	91	85	79	73	68	62	57	51
+ 9							100	95	88	83	77	72	66	60	55
+ 7								98	91	87	81	75	70	64	58
+ 5								100	94	91	85	79	73	68	62
+ 3									98	94	88	83	77	71	66
+ 1									100	98	92	87	81	75	69
− 1										100	96	90	84	79	73
− 3											100	94	88	82	76
− 5												98	92	86	80
− 7												100	96	90	84
− 9													100	93	87
− 11														97	91
− 13														100	95
− 15															98
− 17															100

PRESSURE ALTITUDE – ft.

Fuel Trimmer Position Chart Pre Mod 1371
Figure 1

Engine power checks should not be carried out in ambient conditions in excess of ISA +35 deg. C.

AMBIENT TEMPERATURE — °C	−1000	−500	−250	SL	+500	+1000	+2000	+3000	+4000	+5000	+6000	+7000	+8000	+9000	+10,000
+51	48	41													
+49	52	45	42	39	38										
+47	56	49	46	43	41	39									
+45	59	53	50	47	44	42	38								
+43	63	56	53	50	48	46	43	36							
+41	67	60	57	54	52	49	45	40	35						
+39	71	64	61	58	55	53	49	43	39	35					
+37	75	67	64	61	59	57	52	47	42	38	34				
+35	78	71	68	65	62	60	56	50	46	42	37	32			
+33	82	75	72	69	66	64	59	54	49	45	40	35	31		
+31	86	78	75	72	70	67	63	57	53	48	44	39	34	30	
+29	89	82	79	76	73	71	66	61	56	52	46	42	37	33	29
+27	93	85	82	79	77	74	70	64	60	55	50	46	41	36	32
+25	96	89	86	83	80	78	73	68	63	59	54	49	44	40	35
+23	100	92	89	86	83	81	76	71	67	62	56	52	47	43	38
+21		96	92	90	87	85	80	75	70	65	60	56	51	46	41
+19		99	96	93	90	88	83	78	73	69	64	59	54	49	45
+17		100	99	96	94	91	86	81	77	72	67	62	57	53	48
+15			100	100	96	95	90	85	80	75	70	66	61	56	51
+13					100	98	93	88	83	79	74	69	64	59	54
+11						100	96	92	87	82	77	72	67	62	58
+ 9							100	95	90	85	80	76	71	65	61
+ 7								98	93	88	84	79	74	69	64
+ 5								100	97	92	87	82	77	72	67
+ 3									100	95	90	85	80	75	70
+ 1										98	93	89	84	78	74
− 1										100	96	92	87	82	77
− 3											99	95	90	85	80
− 5											100	98	93	88	83
− 7												100	96	91	86
− 9													100	94	89
−11														97	93
−13														100	96
−15															99
−17															100

FULL INCREASE

PRESSURE ALTITUDE – ft.

Fuel Trimmer Position Chart Having Mod 1371
Figure 2

Subtract these amounts from data plate limitations before comparing with observed results.

The observed TGT must fall between the actual data plate limitations at atmospheric conditions in this area.

PREVAILING AIR TEMPERATURE — °C	−1000	−500	−250	SL	+500	+1000	+2000	+3000	+4000	+5000	+6000	+7000	+8000	+9000	+10,000
+21	5														
+19	11	0													
+17	17	7	0												
+15	23	12	6	0											
+13	28	18	12	6	2										
+11	34	23	17	11	7	3									
+ 9	39	29	23	17	13	9	0								
+ 7	44	34	28	22	19	14	6								
+ 5	49	39	34	27	24	20	11	3							
+ 3	54	44	39	33	30	25	17	9	0						
+ 1	59	49	44	38	35	31	22	14	6						
− 1	64	54	49	43	40	36	27	20	11	3					
− 3	69	59	54	48	45	41	32	25	16	8	0				
− 5	74	64	59	54	50	46	37	30	22	14	6				
− 7	79	69	64	59	55	51	43	35	26	19	11	3			
− 9	84	74	69	64	60	56	48	40	32	24	16	8			
−11	88	79	74	69	65	61	53	45	37	29	21	13	5		
−13	93	84	79	73	70	66	58	50	42	34	26	18	10	3	
−15	98	89	84	78	75	70	63	55	47	39	31	24	15	8	
−17	103	93	88	83	79	75	68	60	52	44	36	29	20	13	4
−19	107	98	93	88	84	80	72	64	57	49	41	34	26	18	9
−21	112	103	97	93	89	85	77	69	62	54	46	39	31	23	14
−23	116	107	102	97	93	89	82	74	66	59	51	44	36	28	19
−25	121	112	107	102	98	94	87	79	71	64	56	49	41	33	24
−27	126	116	111	107	103	99	91	84	76	69	61	53	46	38	29
−29	131	121	115	111	107	103	96	88	81	73	65	58	50	42	33
−31	136	125	120	115	111	107	100	92	85	78	70	62	55	47	38
−33	140	130	124	119	115	112	104	97	89	82	74	67	59	51	42
−35	145	134	129	124	120	116	108	101	93	86	78	71	63	55	46
−37	149	138	133	128	123	120	112	104	97	90	82	75	67	59	50
−39	154	142	137	132	128	124	116	108	101	93	85	79	70	62	54

PRESSURE ALTITUDE – ft.

TGT Correction Chart
Figure 3

POWER PLANT FLUID DRAINS

1. Description

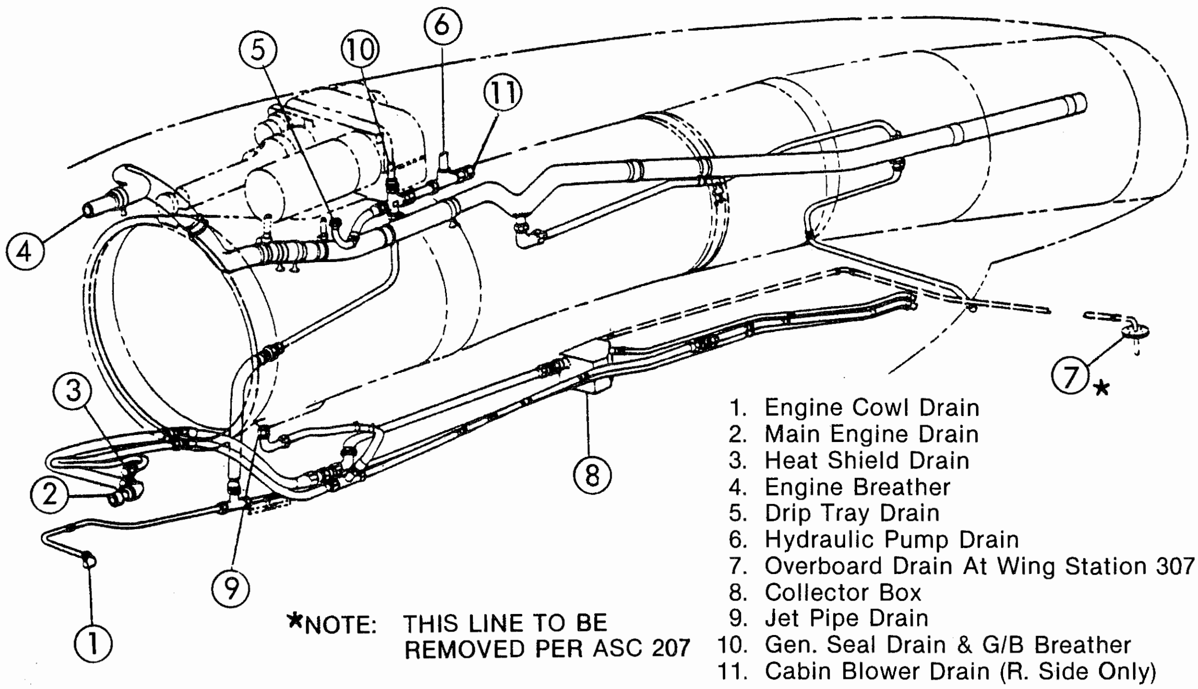
Fluid from the flow control unit, the fuel pump, water methanol control unit, the compressor casing, the combustion chambers and the nozzle box is discharged through individual lines to the drain valve on each engine. This drain valve is located on the No. 5 combustion chamber just aft of the engine fireshield. A single line from the drain valve runs aft to the fuel collector box in the respective wheel well. See Figures 1 and 2. This fluid then is stored in this tank until drained by maintenance personnel. A valve is provided in the bottom of each tank by ASC 207, for the controlled draining and removal of fluids drained into this tank.

Teed to the engine cowl drain aft of the firewall are the following:

Hydraulic Pump Drain
Gearbox Drip Tray Drain
DC Generator Seal Drain
Gearbox Breather

The cowl drain runs parallel to the heat shield drain and discharges overboard at the rear of the nacelle.

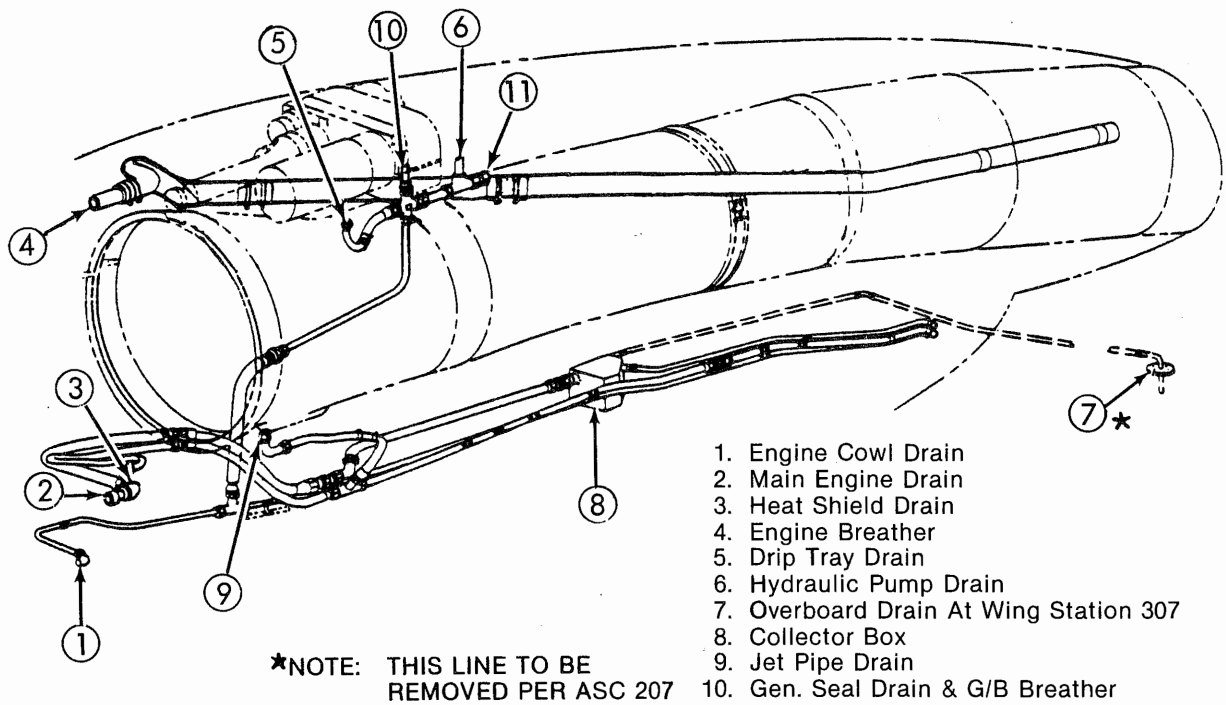
On Aircraft 1 - 144 including 322 and 323 the engine breather line has a V sump drain that discharges overboard at the rear of the nacelle. On Aircraft 145 - 200 and Aircraft 1 - 144 including 322, 323 having ASC 173, the V sump and sump drain have been eliminated.



1. Engine Cowl Drain
2. Main Engine Drain
3. Heat Shield Drain
4. Engine Breather
5. Drip Tray Drain
6. Hydraulic Pump Drain
7. Overboard Drain At Wing Station 307
8. Collector Box
9. Jet Pipe Drain
10. Gen. Seal Drain & G/B Breather
11. Cabin Blower Drain (R. Side Only)

*NOTE: THIS LINE TO BE REMOVED PER ASC 207

Power Plant Fluid Drain Lines and Breather
Aircraft 1 - 144 Including 322, 323 Not Having ASC 173
Figure 1



1. Engine Cowl Drain
2. Main Engine Drain
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9. Jet Pipe Drain
10. Gen. Seal Drain & G/B Breather
11. Cabin Blower Drain (R. Side Only)

*NOTE: THIS LINE TO BE REMOVED PER ASC 207

Power Plant Fluid Drain Lines and Breather
Aircraft 145 - 200 and Aircraft 1 - 144 Including 322, 323 Having ASC 173
Figure 2

LIST OF EFFECTIVE PAGES

CHAPTER 72

ENGINE

SECTION	PAGE	DATE
*72	i	July 10/92
*72	ii	July 10/92
*72	1	July 10/92
*72	2	July 10/92
*72-0	1	July 10/92

* Pages revised by Revision 47
** Pages added by Revision 47
*** Pages deleted by Revision 47

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ENGINE

ENGINE — GENERAL

Refer to Rolls-Royce Dart Maintenance Manual (M-Da7-G).

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ENGINE FUEL AND CONTROL

* Pages revised by Revision 47
** Pages added by Revision 47
*** Pages deleted by Revision 47

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CHAPTER 73

ENGINE FUEL AND CONTROL

ENGINE FUEL FILTER DEICING SYSTEM — DESCRIPTION / OPERATION

1. Description

Each engine fuel feed line is equipped with a fuel filter deicing system consisting of a differential pressure switch, solenoid operated hot air gate valve, and a hot air line to the fuel heater. The purpose of the system is to detect and remove ice crystals which form in the fuel and collect at the filter of the Flow Control Unit (FCU). Power for the system is supplied from the 28 volt essential dc bus through 5 ampere circuit breakers located on the left circuit breaker panel. A differential pressure switch is installed on a shock-mounted shelf which is located on the left side of each engine between the first and second stage compressors. The line from the low pressure side of the differential pressure switch is tapped into the top of the flow control unit downstream of the filter and the line from the high pressure side of the differential pressure switch is tapped into the fuel inlet connection of the flow control unit to enable the switch to sense a differential pressure across the filter. A two-position FUEL FILTER HEAT ON-AUTO switch is provided in the cockpit upper overhead panel for each fuel filter heater. (See Figure 1) Fuel filter heater indicator lights are located below the fuel filter deicing switches. The left and right differential pressure switches are accessible when the cowlings on each engine are opened. (See Figure 2) Aircraft 1 - 99 and 114, having ASC 117 or 117A and Aircraft 100 - 200, 322 and 323 have fuel filter heat timers added. The fuel filter heat timer circuit adds timers, relays, and contacts to nutcracker relay number one. All these components are contained in the fuel filter heat timer box assembly which is installed under FLR No. 8 (Fuselage Station 232 1/2).

NOTE: ASC 117A changes the timing cycle from two minutes (ASC 117) to five minutes 45 seconds and use of fuel filter heat while on the ground.

Aircraft 1 - 200, 322 and 323 having ASC 114 added fuel temperature bulbs and fuel temperature indicating gages to the system. These added parts permit visual monitoring of the fuel temperature at the inlet (low pressure fuel filter bowl) of each engine. The ability to monitor this temperature serves as a check of the effectiveness of the fuel filter deicing system. (Operation of the fuel heater should increase fuel temperature by approximately 500C.) The fuel temperature bulbs are installed in the fuel filters. The gages normally take the positions formerly occupied by the fuel flow indicators on the right center instrument panel. (The fuel flow indicators are moved to the locations of the OAT and W/M QUANTITY indicators which are in turn relocated to the copilots flight instrument panel if space is available.)

2. Operation

If ice crystals form and collect on a fuel filter during auto operation, the pressure drop across the filter causes the associated differential pressure switch to close and completes a circuit to the solenoid controlled hot air gate valve. The differential pressure switch operates when a differential of 1.5 to 1.9 psi exists between the sensing lines. The solenoid controlled hot air gate valve opens and permits hot air from the engine second stage compressor to flow into the heat exchanger unit. When the pressure differential drops to 1.5 psi, the switch returns to its original position, thus breaking the circuit to the solenoid at the hot air gate valve. (See Figure 3)

Placing the FUEL FILTER HEAT switches in the "ON" position open the solenoid valve (regardless of the pressure drop across the filter) and illuminates the amber lights located beneath the FUEL FILTER HEAT switches. (See Figure 3)

NOTE: Illumination of either of these lights does not indicate that the hot air gate valve solenoid is open, but only that it is receiving power.

On Aircraft 1 - 99 having ASC 117 or 117A and Aircraft 100 - 200, 322 and 323, the solenoid controlled hot air gate valve opens and permits hot air from the engine second stage compressors to flow into the heat exchanger unit for two minutes (ASC 117) or five minutes 45 seconds (ASC 117A) timed cycle once instituted by a pressure drop. When the timed cycle is complete, the circuit to the solenoid controlled hot air gate valve is broken. (See Figure 4 and Figure 5)

During the period in which the engine compressor is supplying hot air to the heat exchanger unit, a seven percent loss of engine power occurs. As a result, it is not desirable to have the AUTO portion of the fuel filter

deicing system operative while on the ground, since a differential across the filter can develop without ice or contamination momentarily while starting engines or performing certain ground fuel checks prior to flight. Consequently, the AUTO section of the fuel filter deicing system is inoperative while on the ground through the ground position of the nutcracker system. The manual ON position, however, is not effected. As the aircraft becomes airborne, the nutcracker system makes the AUTO system operative for flight deicing of the filter if selected.

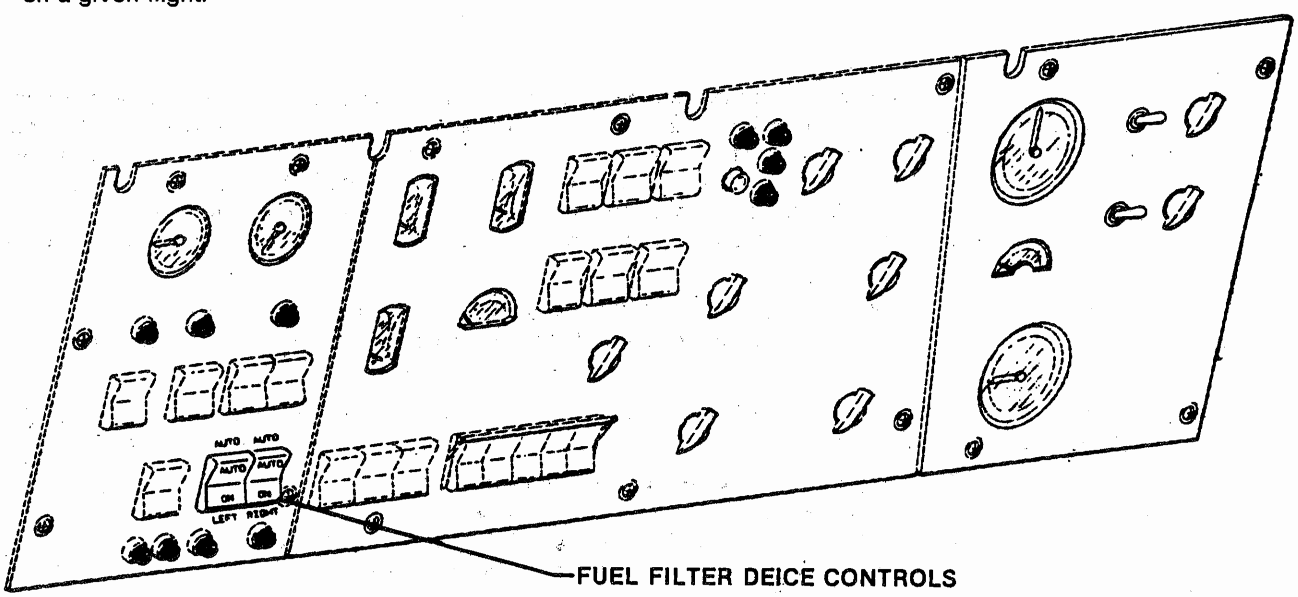
NOTE: Aircraft 1 - 99 having ASC 117A and Aircraft 100 - 200, 322 and 323, the auto section of the fuel filter deicing system is not timed on the ground. However, heat will be supplied as long as there is a differential pressure across the fuel filter. The continued application of heat, as indicated by the indicator light, indicates the presence of solid contaminants in the fuel rather than ice.

When the aircraft or fuel has been exposed to temperature below 20°C prior to takeoff or landing, the fuel filter heaters should be switched to ON for two minutes immediately prior to takeoff and landing, then switched to AUTO. Even though filter icing condition exist during the takeoff roll, the aircraft is airborne and climbing before the filter ices up again and causes the hot air valve to open. As a result the power loss which accompanies the opening of the hot air valve does not occur during the crucial period of takeoff.

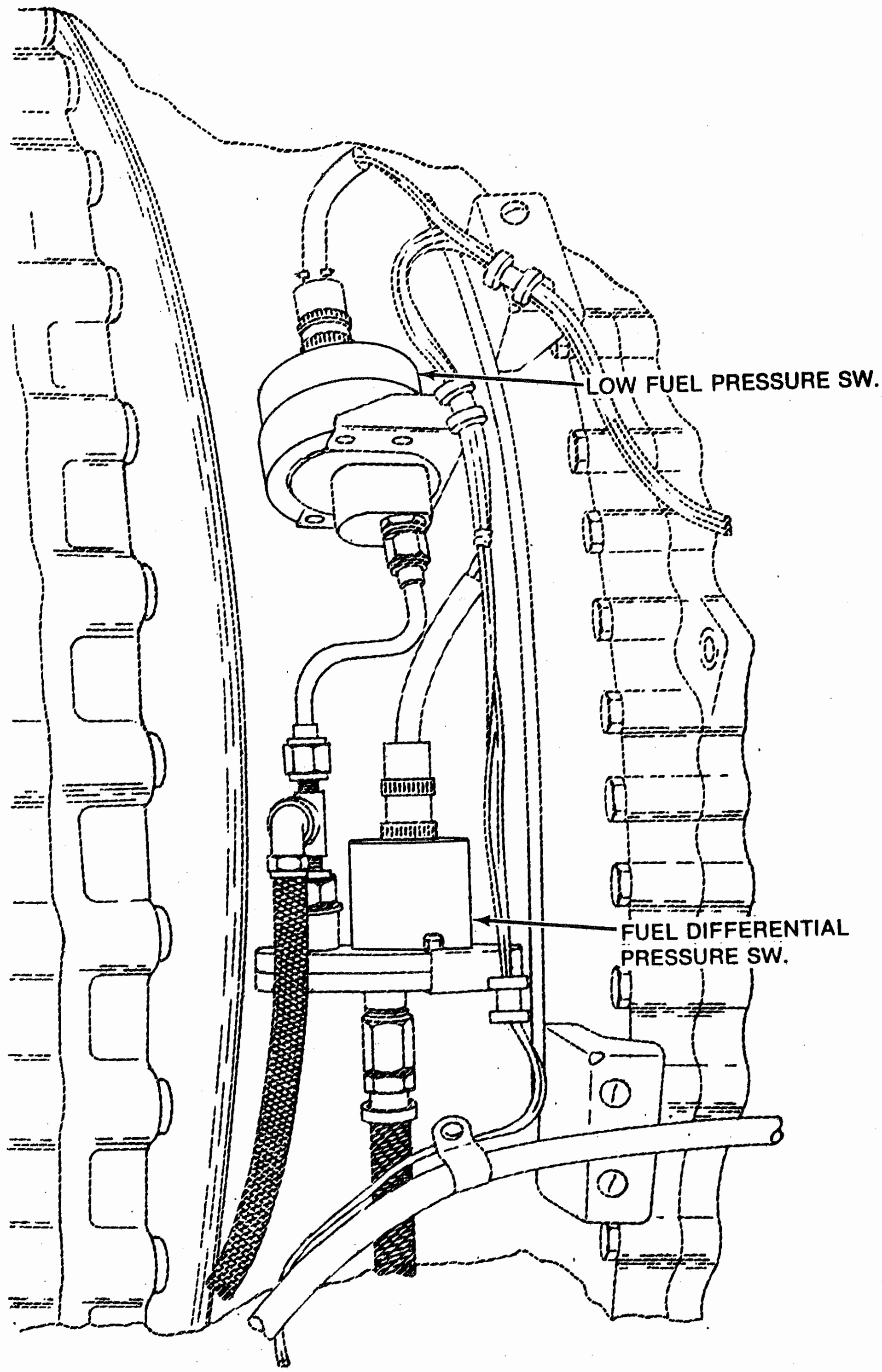
Under conditions of prolonged use of the fuel heaters in flight, there is a loss of engine power up to approximately 50 psi torque pressure and 40°C TGT. If sufficient fuel trimmer datum range is available, this loss of TGT, can be restored by trimming up to limit TGT. If full increase is reached before limit TGT, partial restoration of power is achieved. The fuel heaters increase fuel temperature approximately 50°C.

NOTE: Engines that have R.R. Dart Modification 773 react differently. (Modification 773 is a lengthened servo pillar on the FCU, it is intended to compensate for fuel temperature differences.) This modification overcompensates and results in an increase in TGT of up to 40°C when fuel heat is turned ON. It is then necessary to trim down to limiting TGT if sufficient trim range is available. If not, retardation of the power lever is necessary to maintain TGT within established limits. A problem of this type can normally be resolved by reinterconnection of the engine controls.

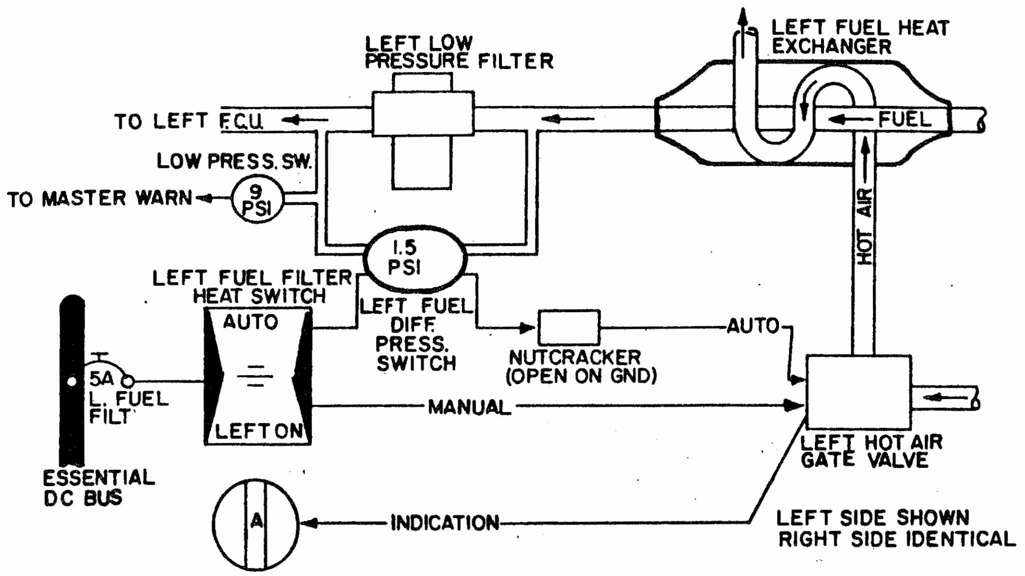
While this system is basically one of fuel deicing, it also serves to indicate filter blockage caused by contaminants other than ice. Indication will manifest itself in the normal manner when AUTO is selected. Application of heat will not dissipate the contaminant and the light will remain on. This will be more noticeable at higher fuel flows, particularly at takeoff flows. A dirty filter will also manifest itself by the frequency of deicing cycles on a given flight.



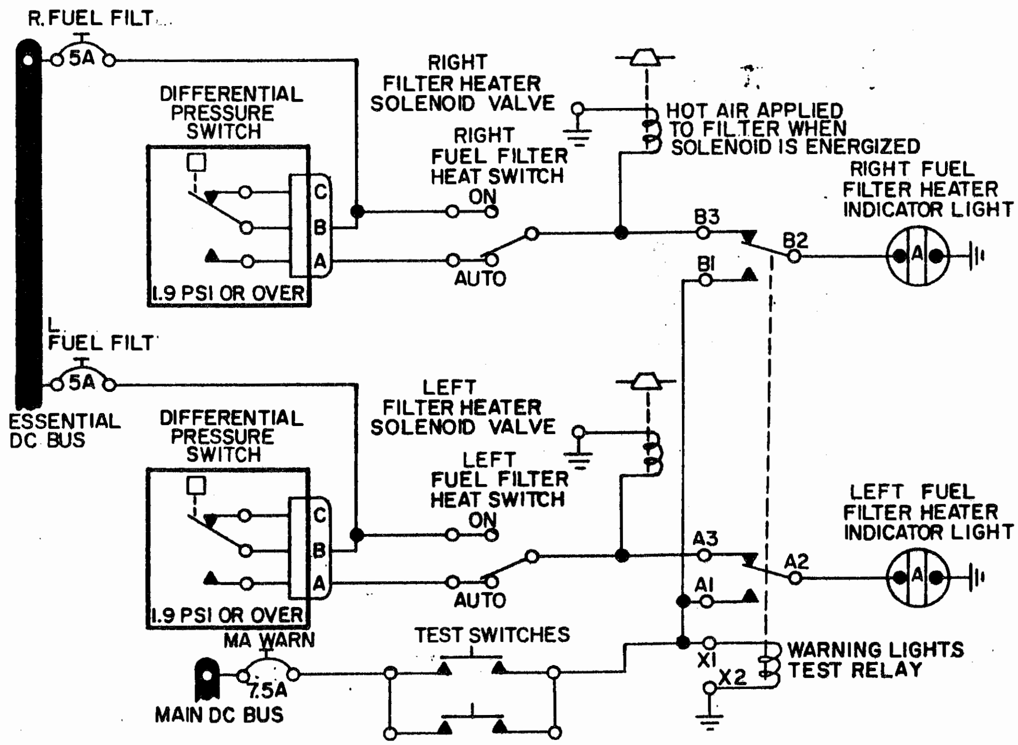
Fuel Filter Heat Controls
Figure 1.



Fuel Filter Heat System Engine — Mounted Components
Figure 2.

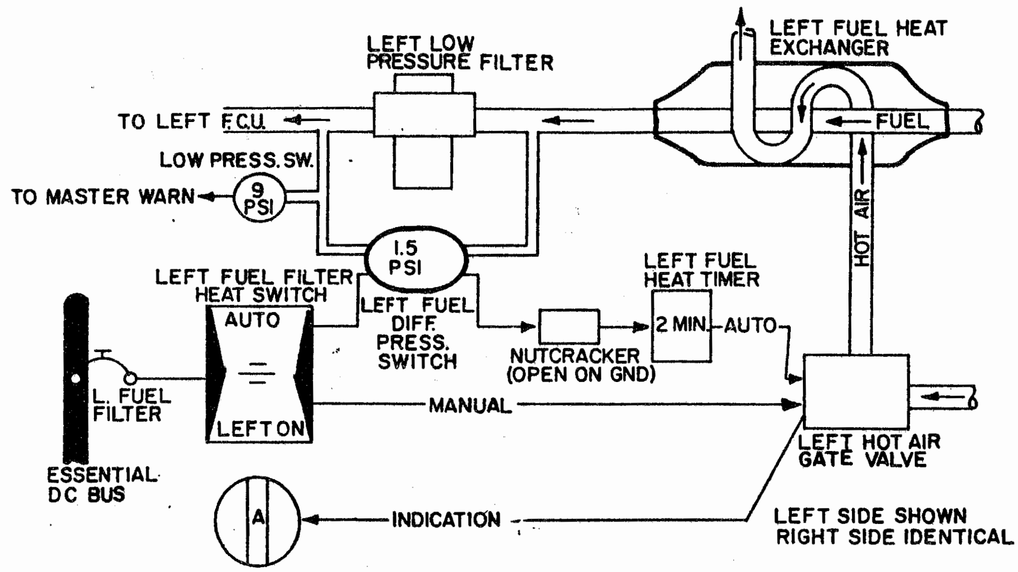


FUEL FILTER HEAT–SYSTEM SCHEMATIC

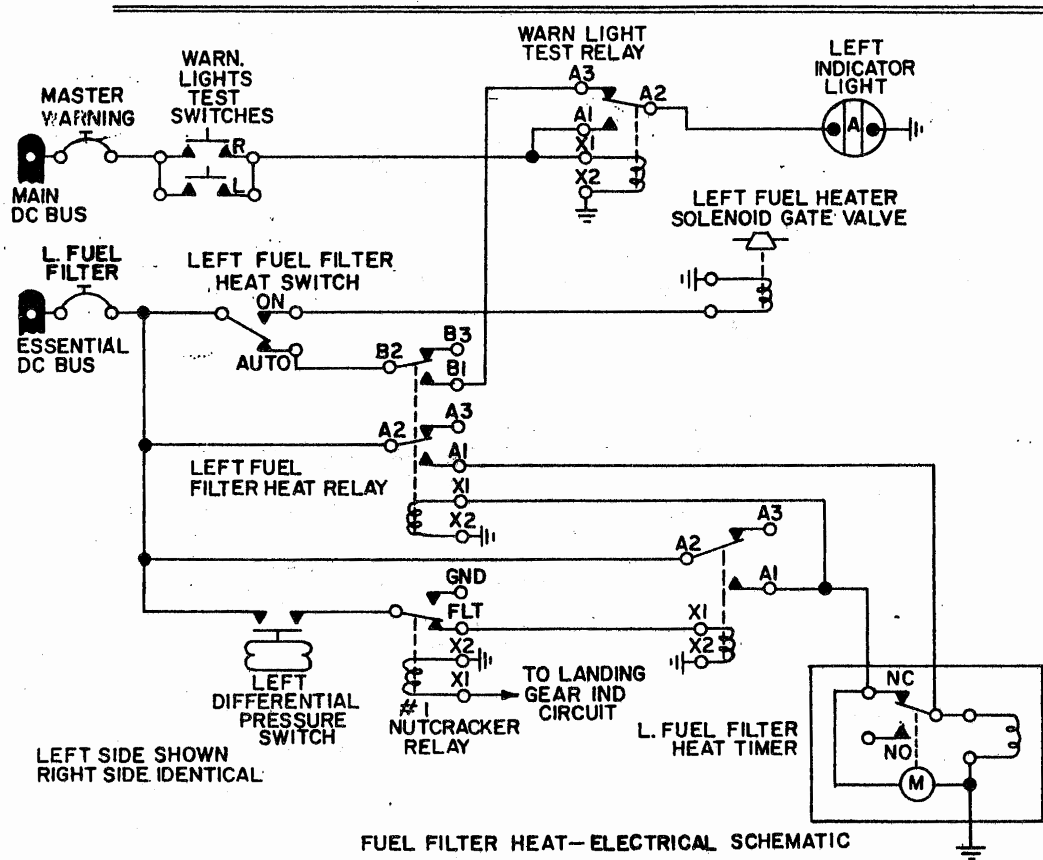


FUEL FILTER HEAT-ELECTRICAL SCHEMATIC

Fuel Filter Heat System — Aircraft 1 - 99, and 114, not having ASC 117 or 117A)
Figure 3.

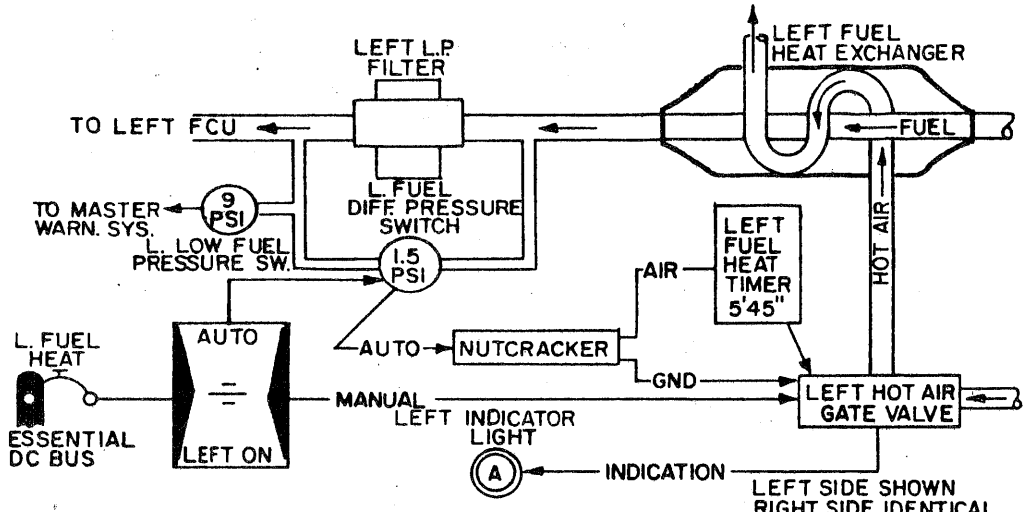


FUEL FILTER HEAT–SYSTEM SCHEMATIC

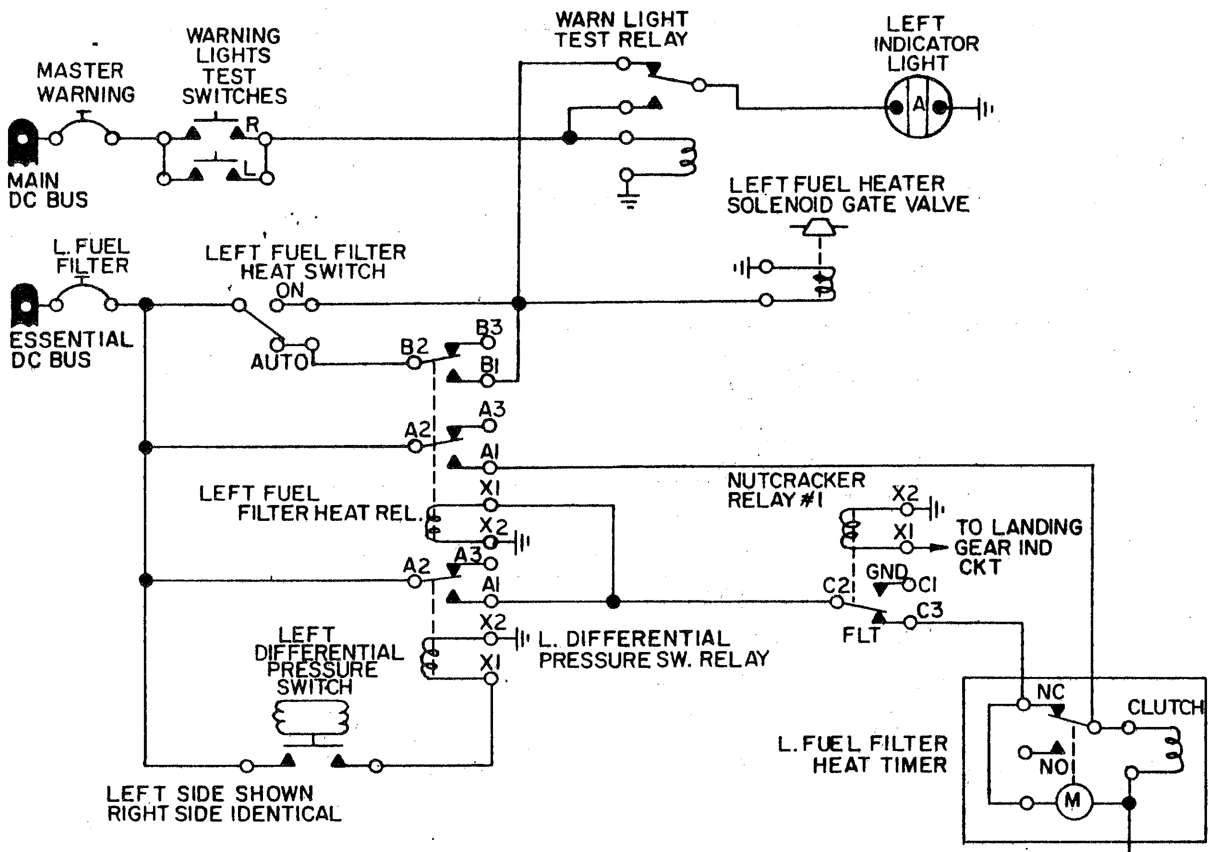


FUEL FILTER HEAT—ELECTRICAL SCHEMATIC

Fuel Filter Heat System — Aircraft 1 - 82, and 114, having ASC 117)
Figure 4.



FUEL FILTER HEAT-SYSTEM SCHEMATIC



FUEL FILTER HEAT—ELECTRICAL SCHEMATIC

Fuel Filter Heat System — Aircraft 1 - 99, and 114, having ASC 117A
and Aircraft 100 - 200, 322 and 323)
Figure 5.

ENGINE FUEL FILTER DEICING SYSTEM — MAINTENANCE PRACTICES

1. **Fuel Filter Deicing System - Functional Test**

(Aircraft 1 - 99 and 114, not having ASC 117 or 117A)

A. Energize main and essential dc buses (BATT switch in NORM.)

B. Close main fuel shutoff valve by pulling FIRE PULL T-handle.

C. Disconnect 3/16 inch pressure sensing line at LOW side of differential pressure switch.

D. Disconnect 1/4 inch pressure sensing line at HIGH side of differential pressure switch.

E. Connect an air pressure source and a 0-30 psi direct reading gage, or a manometer, to HIGH side of switch.

F. Place fuel filter heat switch to AUTO position.

G. Slowly increase air pressure until hot air solenoid valve mounted on underside of engine clicks open. Note that pressure gage reads between 1.5 and 1.9 psi and that amber warning light below switch is on.

H. Slowly decrease air pressure until hot air solenoid valve mounted on underside of engine clicks closed. Note that pressure gage reads between 1.5 and value obtained in Step G, and that amber warning light below switch goes off.

I. Set fuel filter heat switch to ON and listen for the hot air solenoid valve to click open. Amber warning light below switch should come on.

J. Set fuel filter heat switch back to AUTO and listen for the solenoid valve to click closed. Amber warning light below switch should go off.

K. Reconnect lines disconnected in Steps 1.C and D.

L. Perform bleed procedures.

2. **Fuel Filter Deicing System - Functional Test**

(Aircraft 1 - 99 and 114, having ASC 117 or 117A and Aircraft 100 - 200, 322 and 323.)

A. Connect external power to aircraft.

B. Energize main and essential dc buses. (EXT PWR switch ON, BATT switch NORM.)

C. Close main fuel shutoff valve by pulling FIRE T-handle.

D. Disconnect 3/16 inch pressure sensing line at LOW side of differential pressure switch.

E. Disconnect 1/4 inch pressure sensing line at HIGH side of differential pressure switch.

F. Connect an air pressure source and a 0-30 psi direct reading gage, or a manometer, to HIGH side of switch.

G. Place fuel filter heat switch to AUTO.

H. With stop watch or suitable clock available perform the following:

(1) Pull NUTCRACKER circuit breaker.

(2) Slowly increase air pressure until solenoid valve clicks. Start clock at this point.

(3) Note pressure when click was heard. Should be between 1.5 to 1.9 psi.

(4) Note that amber warning light below switch in cockpit comes on simultaneous with actuation of solenoid.

(5) Decrease pressure immediately.

(6) Warning light should continue to glow in cockpit and solenoid should remain energized for 2 minutes $\pm$10 seconds on Aircraft 1 - 82 and 114, having ASC 117. Aircraft 1 - 99 and 114, having ASC 117A and Aircraft 100 - 200, 322 and 323 the time is 5 minutes 45 seconds.

NOTE: If more than 27.5 volts are applied, the timer motor may run faster. If less than 27.5 volts are applied it may run slower. Compensations to above time will have be made. If for any reason the applied voltage is removed the motor will stop and the time cycle will start over again when the voltage is reapplied.

(7) At end of timing cycle solenoid should deenergize (click) and light below switch in cockpit should go out simultaneously.

(8) Reset NUTCRACKER circuit breaker.

I. Place fuel filter heat switch to on and listen for the hot air solenoid valve to click open. Amber light below the switch should come on as long as the switch is ON.

J. Place fuel filter heat switch to AUTO and listen for the hot air solenoid valve to click closed. Warning light in cockpit should go off.

K. With fuel filter heat switch in AUTO apply pressure to 2.0 psi. Solenoid valve should click. Warning light in cockpit should come on.

L. Release pressure. Solenoid valve should click and light should go off.

M. Place fuel filter heat switch to ON. Valve should click and warning light should come on in cockpit.

N. Place fuel filter heat switch in AUTO. Valve should click closed and light should go off.

O. Reconnect lines disconnected in Steps C. and D. above.

P. Perform Fuel System — Bleed, See Chapter 28.

3. Solenoid Hot Air Gate Valve — Operational Test

A. Start left engine.

B. Place left fuel filter heat switch to AUTO.

C. Carefully observe engine tachometer and, while doing so, place left fuel heat switch to ON.

NOTE: After the switch has been placed to the ON position, a momentary drop of engine rpm will occur. This is an indication that the hot air gate valve has opened and is supplying heat to the fuel line.

On Aircraft 1 - 200, 322 and 323, having ASC 114, a rise in fuel temperature will be seen which is an indication of hot air gate valve opening.

D. Return switch to AUTO.

4. Fuel Differential Pressure Switch — Removal / Installation

A. Removal

(1) Open engine cowl.

(2) With electrical power on aircraft pull FIRE PULL T-handle.

(3) De-energize electrical power.

(4) Disconnect electrical connector from pressure switch.

(5) Disconnect sensing line at HP port.

(6) Disconnect lines from T-fitting at low pressure port.

(7) Cap lines.

(8) Remove screws securing switch to shelf assembly and remove switch.

B. Installation

(1) Replace O-ring.

(2) Install switch on shelf and secure with screws.

(3) Remove caps and connect lines to HP port and T-fitting.

(4) Connect electrical connector pressure switch.

(5) Perform Fuel Differential Pressure Switch — Functional Test, this Section.

(6) Perform Fuel System — Bleed, See Chapter 28.

(7) Inspect area for presents of foreign objects, security of all attachments.

(8) Close engine cowl.

5. **Fuel Differential Pressure Switch — Functional Test**

A. Open engine cowl.

B. Energize main and essential dc buses.

C. Pull FIRE PULL T-handle to close main fuel shutoff valve, ensure fuel heater switch is in AUTO.

D. On aircraft not having ASC 108A, pull LANDING GEAR WARNING HORN and WARNING LIGHTS circuit breakers. On aircraft having ASC 108A, pull LG NUTCRACKER circuit breaker.

E. Disconnect pressure sensing line at high side of pressure switch.

F. Connect air pressure source and pressure gage to high side of switch.

G. Slowly increase air pressure until hot air solenoid valve, on underside of engine clicks open.

NOTE: Use a mercury manometer to observe switch actuation pres sure.

H. Observe pressure when valve actuates. Pressure readings should be:

(1) Switch P/N 32028-1

(a) Increasing pressure - 3.87 in Hg (1.9 psig maximum).

(b) Decreasing pressure - 2.04 in Hg (1.5 psig minimum).

(2) Switch P/N 32028-3

(a) Increasing pressure - 5.09 in Hg (2.5 psig maximum).

(b) Decreasing pressure - 4.07 in Hg (2.0 psig minimum).

I. Note that amber warning light below switch in cockpit comes on simultaneously with actuation of solenoid.

NOTE: Aircraft with 2 minute $\pm$10 seconds or 5:45 $\pm$30 seconds timer, valve will remain energized and amber light will continue to glow after pressure is released due to timer operation. Untimed systems (not having ASC 117A or 117B) valve will deenergize and light will go out when pressure is lowered.

J. Reset circuit breakers.

K. Perform Fuel System — Bleed, See Chapter 28.

L. Remove electrical power from aircraft.

M. Inspect area for presents of foreign objects, security of all attachments.

N. Close engine cowl.

6. **Fuel Heater Assembly — Removal / Installation**

A. Removal

(1) With electrical power on essential dc bus, pull FIRE PULL T-handle for the side involved.

(2) Disconnect fuel heater assembly from engine mount support bracket.

(3) Remove clamps securing split shroud at aft firewall connection to gain access to aft fuel feed line B-nut.

(4) Disconnect aft fuel feed line B-nut.

(5) At low-pressure fuel filter, forward side of fireseal bulkhead, disconnect safety wire from forward fuel feed line B-nut and disconnect B-nut.

(6) Lower aft end of fuel heater assembly and carefully separate assembly from engine hot air inlet line and discard O-ring.

B. Installation

(1) Install new O-ring on engine hot air line. Apply compound to O-ring (MS4 or equivalent).

(2) Install forward fuel feed line through engine fireseal bulkhead. Position and insert the engine hot air inlet line.

(3) Connect forward and aft fuel feed line B-nuts and hand tighten.

(4) Connect and secure fuel heater assembly support bracket to engine mount.

(5) Tighten forward and aft fuel feed line B-nuts and safety wire forward B-nut.

(6) Perform Fuel System — Bleed, See Chapter 28.

(7) Check system installation for leaks.

(8) Inspect area for presents of foreign objects, security of all attachments and close cowl.

7. Fuel Heater Assembly — Pressure Test

A. Open engine cowling.

B. Remove fuel heater assembly, see this Section.

C. Cap fuel heater outlet line (See Figure 201)

NOTE: The above connection contains a British thread. Gulfstream Aerospace P/N 159PM10007-1 may be used as an adapter fitting with an AN929-16D pressure cap.

D. Assemble a test rig consisting of a suitable adapter fitting, pressure regulator, shutoff valve, direct reading pressure gage (0-100 psi recommended) and a pressure bleed valve, connect to fuel heater inlet line and air pressure source. (See Figure 201)

E. Completely submerge the fuel heater assembly in water.

F. Close pressure bleed valve and apply a 50 psi of air pressure to the fuel heater assembly and close shutoff valve.

G. Allow to remain pressurized for 30 minutes. No leakage is permitted.

H. Open pressure bleed valve to depressurize the fuel heater assembly.

I. Remove all test adapters.

J. Remove cap from heater outlet line.

K. Install fuel heater assembly, see this Section.

L. Inspect area for presents of foreign objects, security of all attachments and close cowling.

8. Fuel Heater — Leak Test

A. Open engine cowling.

NOTE: Fuel heaters are on lower left side of engine adjacent to combustion chambers.

B. Remove clamps securing split shroud at aft firewall.

C. Apply dc power to essential bus.

CAUTION: ENSURE FUEL HP COCKS ARE IN FUEL OFF POSITION.

D. Place fuel boost pump switch on.

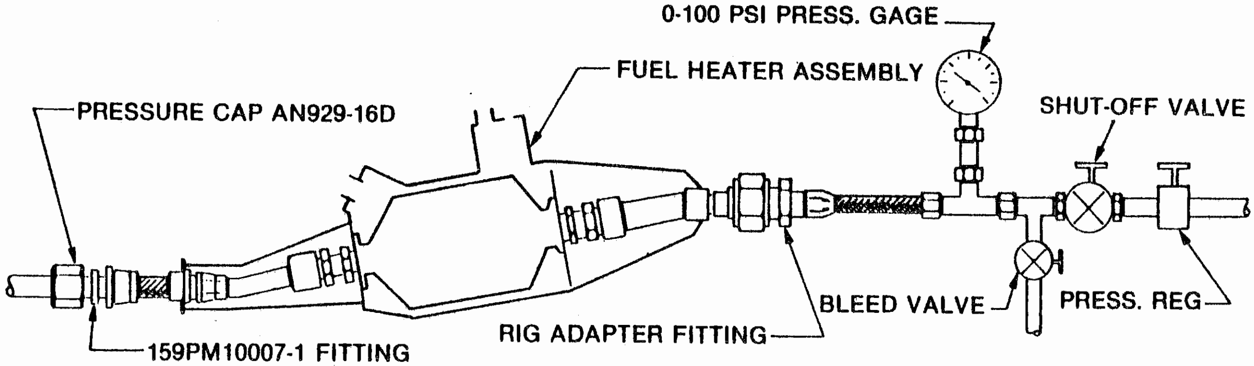
E. Check fuel heater drain, forward and aft B-nut for leaks. (Pressurize heater for 3 minutes minimum)

F. Place fuel boost pump switch off.

G. Remove electrical power.

H. Install shroud.

I. Inspect area for presents of foreign objects, security of all attachments and close cowling.



Fuel Heater Pressure Check Rig
Figure 201.

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ENGINE FUEL LOW PRESSURE WARNING SYSTEM — DESCRIPTION / OPERATION

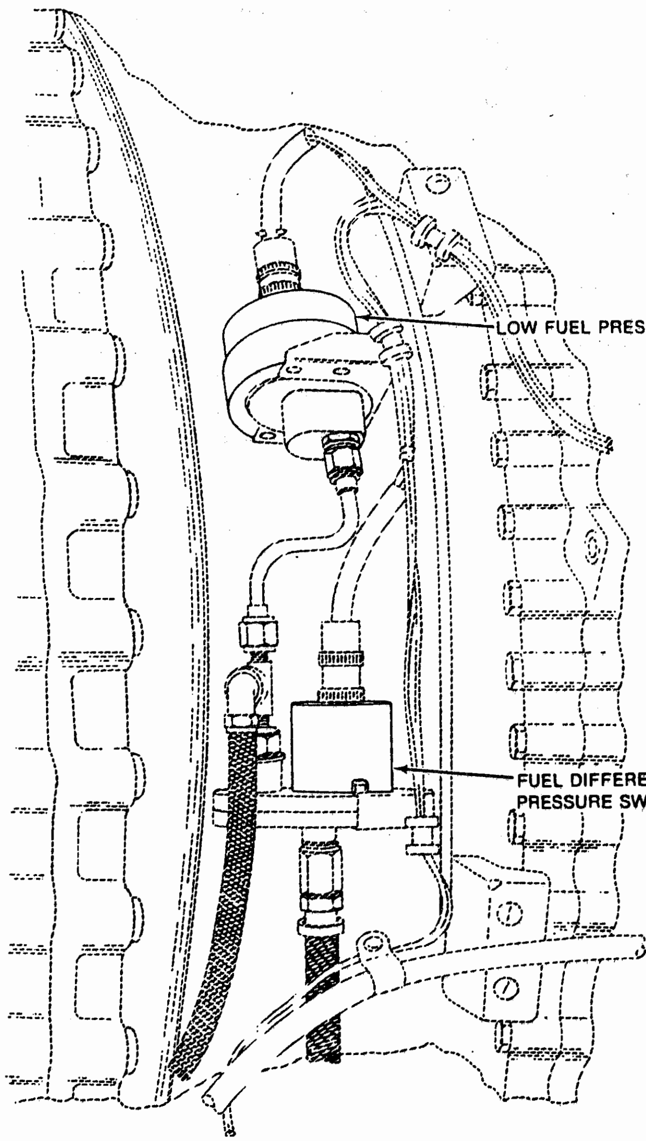
1. Description

The fuel low pressure warning system provides a visible indication of low fuel pressure being delivered to the low pressure inlet port of the engine driven fuel pump on each engine. The system consists of two pressure switches (one for each engine), and all necessary tubing. Power is supplied to the system from the 28-volt essential dc bus through two circuit breakers located on the right circuit breaker panel. A low pressure warning switch is installed on a shock-mounted shelf which is located on the left side of each engine between the first and second stage compressors. (See Figure 1) The pressure line for each system extends from the switch to the low pressure (downstream) side of the differential pressure switch.

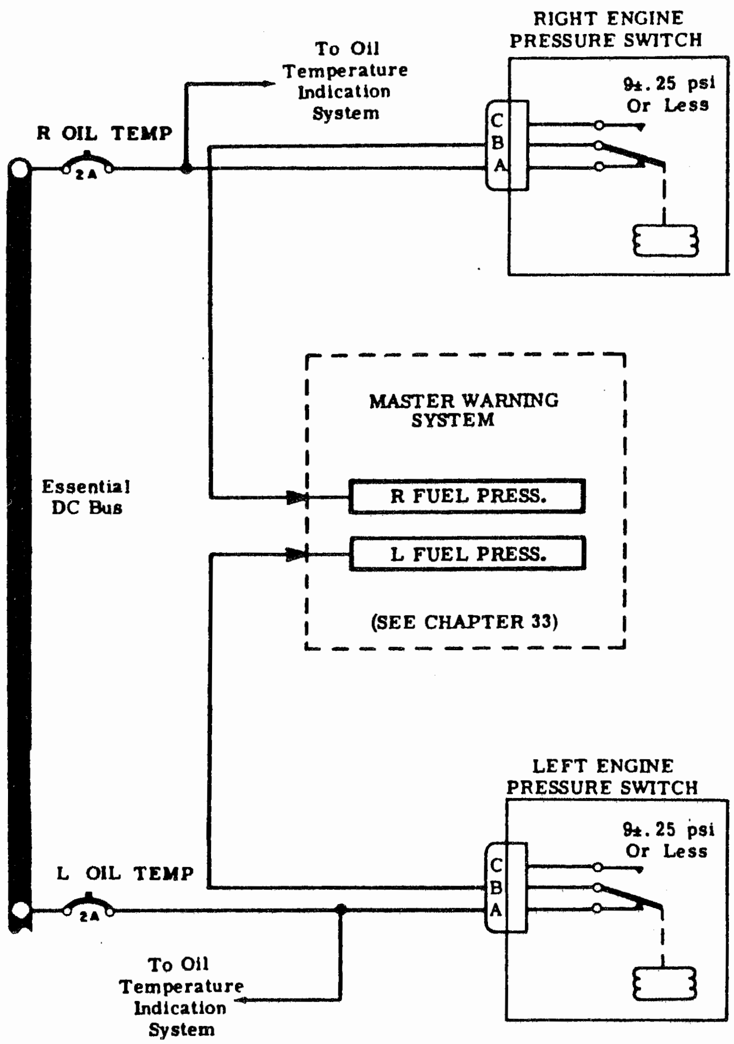
The left and right low pressure warning switches are accessible when the cowlings on each engine are opened. Two low pressure warning lights (L and R FUEL PRESS) are provided as part of the master warning system.

2. Operation

The low fuel pressure switch provides a warning in the cockpit that low fuel pressure condition exists at the inlet side of the engine driven fuel pumps. Pressure is tapped from the engine feed line downstream of the filter and is directed to the pressure switch. The fuel pressure warning lights should be on if the aircraft's electrical system is turned on and the boost pumps are inoperative. Operation of the boost pumps causes pressure to build up in the engine feed line. When pressure reaches approximately 11 psi, the pressure switches are actuated and the lights will go out. The pressure switches are released and the lights in the cockpit go on again at any time pressure in the engine feed line downstream of the filter reaches 9 $\pm 1/4$ psi, or less. (See Figure 2)



Location of Fuel Low Pressure Warning System Switch
Figure 1.



Fuel Low Pressure Warning System — Schematic
Figure 2.

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ENGINE FUEL LOW PRESSURE WARNING SYSTEM — MAINTENANCE PRACTICES

1. **Low Pressure Switch — Pressure Check**

   A. Energize main and essential dc buses (BATT. switch at NORM.).

   B. Close main fuel shutoff valve by pulling Fire T-handle.

   C. Disconnect 1/4 inch sensing line at switch.

   D. Connect an air pressure source and a 0-30 psi direct reading gage or manometer to the switch.

   E. Slowly increase air pressure until approximately 13 psi is reached.

   NOTE: When increasing pressure, warning light should go out at or before reading of 11 psi.

   F. Slowly decrease pressure. Record pressure as warning light comes on.

   NOTE: When decreasing pressure, warning light should come on at 9 $\pm$1/4 psi.

   G. Return system to normal.

   H. Inspect area for presents of foreign objects, security of all attachments.

   I. Perform Fuel System — Bleed, See Chapter 28.

   J. Inspect area for presents of foreign objects, security of all attachments and close cowling.

2. **Low Pressure Switch — Removal / Installation**

   A. Removal

      (1) Open engine cowl.

      (2) With electrical power on, pull FIRE PULL T-handle to close main fuel shutoff valve.

      (3) Remove electrical power.

      (4) Disconnect pressure sensing line at elbow.

      (5) Disconnect electrical plug at pressure switch.

      (6) Remove screws holding switch to mount shelf.

      (7) Remove switch.

   B. Installation

      (1) Install pressure switch with mounting screws.

      (2) Connect electrical plug at pressure switch.

      (3) Connect pressure sensing line at elbow.

      (4) Perform Fuel System — Bleed, See Chapter 28.

      (5) Inspect area for presents of foreign objects, security of all attachments and close cowling.

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ENGINE FUEL TEMPERATURE INDICATION — DESCRIPTION / OPERATION

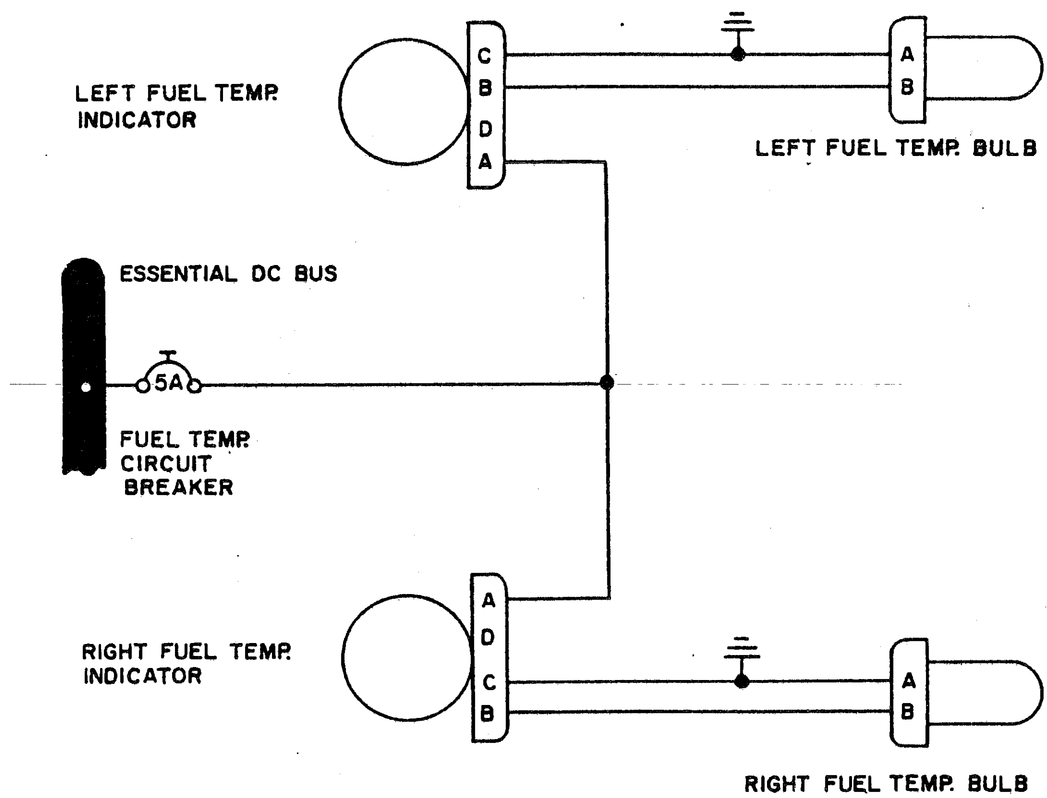
(Aircraft 1 - 200, 322 and 323 having ASC 114)

1. **Description**

Aircraft having ASC 114, add fuel temperature bulbs and fuel temperature indicating gages to the system. These added parts permit visual monitoring of the fuel temperature at the inlet (low pressure bowl) of each engine. The ability to monitor this temperature serves as a check of the effectiveness of the fuel filter de-icing system. (Operation of the fuel heater should increase fuel temperature by approximately 50°C) The fuel temperature bulbs are installed in the fuel filters. The gages normally take the positions formerly occupied by the fuel flow indicators on the right center instrument panel. The fuel flow indicators are moved to the locations of the OAT and W/M QUANTITY indicators which are in turn relocated to the copilots flight instrument panel if space is available.

2. **Emergency DC Operation — Effect On System**

The fuel temperature indication is operative in emergency.



Engine Fuel Temperature Indication Schematic
Figure 1.

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ENGINE FUEL TEMPERATURE INDICATION — MAINTENANCE PRACTICES

1. **Fuel Temperature Bulb — Removal / Installation**

   A. Removal

   (1) Energize aircraft electrical system.

   (2) Close main fuel shutoff valve by pulling FIRE PULL T-handle.

   (3) De-energize aircraft electrical system.

   (4) Disconnect batteries and external electrical power.

   (5) Open engine cowlings on left side of nacelle.

   (6) Cut safety wire to temperature bulb on fuel filter bowl.

   (7) Disconnect electrical plug running to temperature bulb. (See Figure 201

   (8) Turn temperature bulb and remove temperature bulb is bayonet type.)

   B. Installation

   (1) Install temperature bulb with new O-ring. (See Figure 201)

   (2) Connect electrical plug to temperature bulb and safety wire.

   (3) Energize aircraft electrical system.

   (4) Push in FIRE PULL T-handle.

   (5) Disconnect electrical power.

   (6) Reconnect batteries.

   (7) Inspect area for presents of foreign objects, security of all attachments and close cowling.

2. **Fuel Temperature Bulb — Resistance Check**

   NOTE: The fuel temperature bulb(s) must be removed from the fuel filter bowl(s) prior to performing the resistance check.

   A. Remove fuel temperature bulb(s). (See this Section)

   B. Connect ohmmeter to bulb.

   C. Using chart below perform resistance check.

   NOTE: The following information is supplied for fault isolation purposes. It gives the approximate resistance of the bulb at surrounding temperature ranges from − 35 to + 120°F, also°C.

OHMS C	TEMP.	OHMS F
78.97	-35	78.25
80.57	-30	79.13
82.17	-25	80.01
83.77	-20	80.90
85.39	-15	81.79
87.04	-10	82.68
88.69	- 5	83.58
90.38	0	84.48
92.08	5	85.39
93.80	10	86.30
95.55	15	87.22
97.31	20	88.14

OHMS C	TEMP.	OHMS F
99.11	25	89.07
100.91	30	90.00
101.65	32	90.38
102.75	35	90.94
104.60	40	91.89
106.49	45	92.84
108.39	50	93.80
110.33	55	94.76
112.28	60	95.73
114.27	65	96.71
116.27	70	97.70
118.31	75	98.70
119.13	77	99.11
120.36	80	99.70
122.45	85	100.71
124.55	90	101.73
126.70	95	102.75
128.85	100	103.78
131.05	105	104.82
133.26	110	105.86
135.51	115	106.91
137.78	120	107.97

D. Disconnect ohmmeter from bulb.

E. Install fuel temperature bulb(s) in fuel filter bowl(s). (See this Section)

F. Inspect area for presents of foreign objects, security of all attachments and close cowling.

3. **Fuel Temperature Indicator — Removal / Installation**

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Disconnect electrical connector from indicator.

(2) Remove screws that secure indicator to panel.

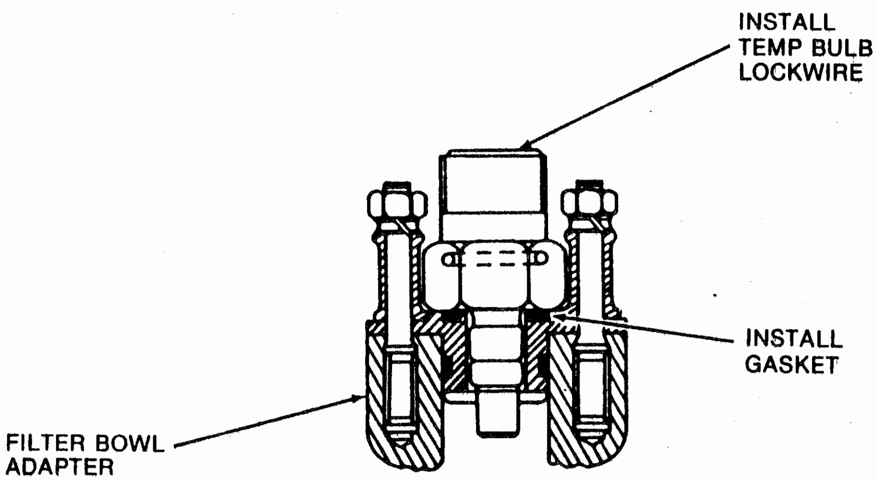
(3) Remove indicator.

B. Installation

(1) Install indicator in panel.

(2) Connect electrical connector.

(3) Energize essential dc bus and check indicator for reading of fuel temperature.



Fuel Temperature Bulb
Figure 201.

LIST OF EFFECTIVE PAGES

CHAPTER 74

IGNITION

SECTION	PAGE	DATE
*74	i	May 8/92
*74	ii	May 8/92
*74	1	May 8/92
*74	2	May 8/92
*74-0	1	May 8/92

* Pages revised by Revision 46
** Pages added by Revision 46
*** Pages deleted by Revision 46

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CHAPTER 74

IGNITION

IGNITION — GENERAL

Refer to Chapter 80, Starting, for information on the Ignition System

LIST OF EFFECTIVE PAGES

CHAPTER 75

POWERPLANT GENERAL

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*75	i	May 8/92
*75	ii	May 8/92
*75	1	May 8/92
*75	2	May 8/92
*75-0	201	May 8/92
*75-0	202	May 8/92

* Pages revised by Revision 46
** Pages added by Revision 46
*** Pages deleted by Revision 46

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CHAPTER 75

POWERPLANT GENERAL

HOT AIR GATE VALVES — MAINTENANCE PRACTICES

1. **Hot Air Gate Valve — Operational Test**

WARNING: OBSERVE ALL SAFETY PRECAUTIONS WHILE STARTING AND RUNNING ENGINES.

A. Energize essential dc bus.

B. Place left or right FUEL FILTER HEAT switch (overhead panel) on and listen for audible operation of hot air gate valve solenoid. Ensure applicable amber indicating light comes on.

C. Place switch to AUTO. Ensure amber indicating light goes out.

D. Start applicable engine.

CAUTION: FUEL FILTER HEAT SWITCH CAN BE LEFT ON FOR 2 MINUTES ONLY ON AIRCRAFT NOT HAVING ASC 117A AND B INCORPORATED. RPM MUST NOT EXCEED 11,000 RPM.

NOTE: Aircraft having ASC 117A and B are equipped with timers.

On Aircraft having ASC 114, a rise in temperature will be observed which indicates opening of hot air gate valve.

E. While observing tachometer, place applicable FUEL FILTER HEAT switch on. A momentary drop of engine rpm should occur indicating hot air valve is open and supplying heat.

F. Place switch to AUTO.

G. Shut down engine.

2. **Hot Air Gate Valve — Removal / Installation**

(See Figure 201)

CAUTION: ENSURE ELECTRICAL POWER IS OFF.

A. Removal

(1) Open bottom engine cowl.

NOTE: Remove outlet adapter on rear side of engine bulkhead together with fuel heater tube connection, before removing hot air gate valve.

(2) Remove safety-wire and disconnect electrical connector.

(3) Remove nuts from D-head bolts that pass through bulkhead and secure outlet adapter to valve outlet connection and separate fuel manifold clip.

(4) Remove outlet adapter and delivery tube aft and downward.

(5) Remove hardware from bolts attaching valve outlet connection to front of bulkhead; then remove bolts with friction washers from each side of bulkhead.

(6) Remove cotter pins and nuts (Mod. 775 self-locking) that secure valve body to support plate on engine. Remove bolts.

(7) Move valve body downward until rubber ferrules are free, and remove ferrules. Push valve forward to clear bulkhead flange and downward to release seals on inlet tube.

(8) Remove valve being careful not to damage seals, remove inlet tube.

B. Installation

NOTE: On Pre-Mod. 664 (Piston-ring type seals), coat graphite grease on inlet tube seals. On Mod. 664 (aluminum and rubber sealing rings). Coat new rings with silicone compound.

(1) Insert inlet tube into valve body.

(2) Move assembly up to engine at an oblique angle to clear bulkhead while exerting a steady upward pressure to permit upper seal to seat correctly.

(3) Center assembly and fit rubber ferrules on each side of valve mounting flange.

(4) Install bolts and nuts (Mod. 775 self-locking) and cotter pins.

(5) Install friction washers on both sides of bulkhead when replacing bolts that make valve connection. until connection is snug. Back off three castellations before installing cotter pins.

(6) Inspect area for presents of foreign objects, security of all attachments.

(7) Coat outlet tube seals with graphite grease and install.

(8) Install D-head bolts through bulkhead and secure outlet adapter to valve OUTLET connection.

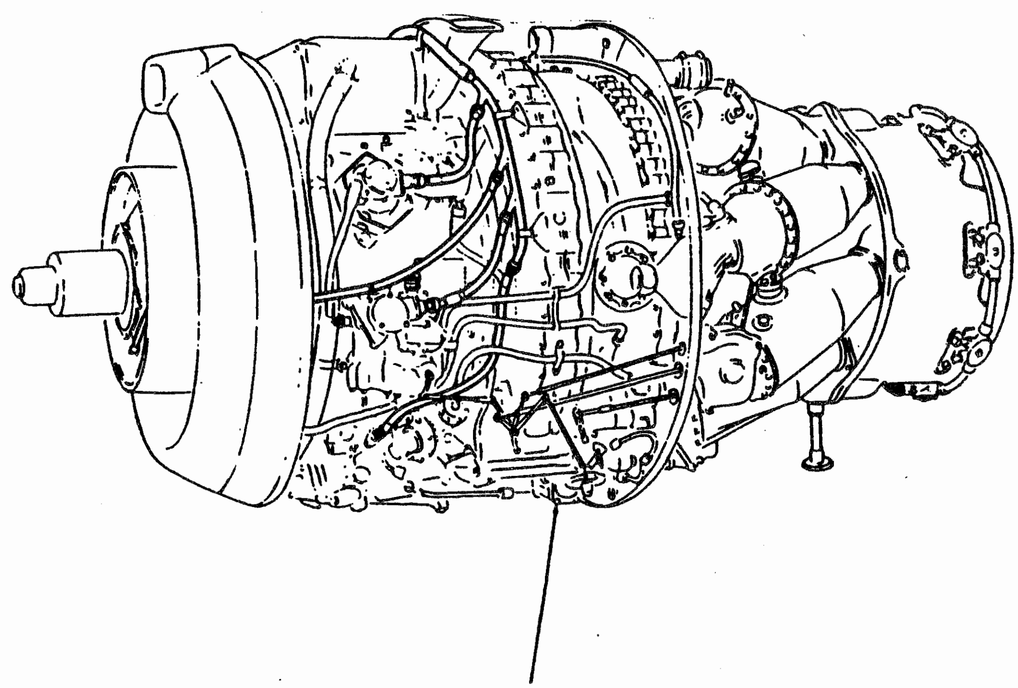
(9) Install fuel manifold clip on studded D-head bolt.

(10) Install and tighten nuts.

(11) Connect electrical connector and safety-wire.

(12) Perform Hot Air Gate Valve — Operational Test.

(13) Close bottom engine cowl.



Hot Air Gate Valve
Figure 201.

LIST OF EFFECTIVE PAGES

CHAPTER 76

ENGINE CONTROLS

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*76	i	September 11/92
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76-1-0	6	May 8/92
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76-1-0	202	May 8/92

* Pages revised by Revision 48
** Pages added by Revision 48
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CHAPTER 76

ENGINE CONTROLS

ENGINE CONTROL SYSTEM — DESCRIPTION / OPERATION

1. Description

NOTE: Power plant control rods and levers mounted on the engine are installed by engine manufacturer and terminate at engine control box. The description, operation and rigging information pertaining to these components is covered in the Rolls-Royce Dart Maintenance Manual M-Da7-G. Information presented in the Dart Manual must be used with this chapter to complete the description of the power plant control system. The engine control box, located on the left side of the engine, is the pickup point at which engine and aircraft throttle lever systems are joined together.

Power plant control lever installed on the center pedestal console. Outboard levers are the left and the right high pressure fuel cock levers and inboard levers are the throttle levers. High-pressure fuel cock and throttle levers are connected to the control pedestal sectors by push-pull rods. High-pressure fuel cock and throttle cable terminating sectors are installed on the left side of each nacelle. Nacelle sectors are connected to their related fittings on the engine control box by a series of push-pull rods and levers. Movement of any engine lever will therefore, be transmitted to move the corresponding engine control component. In addition to basic engine controls, a fuel trim motor is installed in each nacelle and is connected to the engine control box through push-pull rods and levers. The right engine throttle linkage differs from the left in that it contains the synchronizer corrector motor. The synchronizer motor adjusts the position of the right engine throttle linkage so that the rpm of the right engine equals that of the opposite engine. A throttle lever interlocking device can be manually operated by means of the propeller fine pitch selector and the flight controls gust lock control lever.

A. High-Pressure Fuel Cock Levers

High-pressure fuel cock levers, one for each engine, are provided to open or close the high-pressure fuel cocks, to control propeller manual feathering and to manually remove the cruise pitch lock through the propeller control unit. Levers are spring loaded outboard and must be moved inboard to clear the detent stops. (See Figure 1) Detent stop positions, from extreme aft to full forward, are as follows: FEATHER, FUEL OFF, FUEL ON and CRUISE LOCKOUT. Levers have a travel of 68°, with fixed stops located on the pedestal sectors to permit 2° of overtravel at each end, thus limiting handle travel to 72°. A push-pull rod connects each lever to the pedestal sectors.

B. Throttle Levers

Throttle levers (one for each engine) are connected to the engine fuel controls by cables and push-pull rods. Levers have a travel of 60°, with fixed stops located on the pedestal sectors to permit 5° overtravel at each end, thus limiting the total travel to 70°. The extreme aft position is IDLE, while the full forward position is maximum power, cruise power is monitored between the two extremes. These levers are attached to the pedestal sectors by push-pull rods. A throttle lever friction lock is incorporated in the control pedestal console to lock the levers in any desired position and prevent creeping.

C. Fine Pitch Lock Selector Levers

Propeller fine pitch lock selector lever is located in the center aft slot of the control pedestal console. It is spring loaded in either the forward or aft position and locks in the forward or FLIGHT position. An integral latch must be released to move the lever aft to the GROUND INTERLOCK position. The functions of this control lever are to select the removal or return of the propeller flight fine pitch stop and to form a link in the throttle lever/gust lock interlock. This lever, with attached push-pull rods, actuates electrical switches mounted in the pedestal.

D. Gust Lock Lever

The gust lock lever, located in the center forward slot of the control pedestal console, primarily is a flight control lock, but it also forms a link in the interlock system. It is a two-position lever, with the forward position placarded OFF and the aft position placarded ON.

E. Throttle Lever Interlocking System

A throttle lever interlocking device can be operated manually by means of the propeller fine pitch lock selector lever and the flight control gust lock lever. With the propeller fine pitch lock selector lever in the

GROUND INTERLOCK position, and the gust lock lever in the ON position, the following conditions prevail:

- Both throttle levers cannot be advanced beyond the 30° position at the same time.
- Advancing one throttle lever toward maximum automatically will retard the other below its 30° setting.
- The propeller fine pitch lock selector lever is locked in GROUND INTERLOCK position when the gust lock lever is in ON position.

When the propeller fine pitch lock selector lever is in the FLIGHT position:

- It is impossible to move the gust lock lever to the ON position.
- Throttle lever movement is unrestricted between IDLE to MAXIMUM POWER.
- The propeller fine pitch lock selector lever cannot be moved to the GROUND INTERLOCK position when the throttle levers are advanced to the 30° setting.

With the gust LOCK lever in the OFF position, advancement of both throttle levers past the 30° setting engages a mechanism which moves the propeller fine pitch lock selector lever from the GROUND INTERLOCK position to the FLIGHT position. This movement operates a latching device on the fine pitch lock selector lever, locking it in the FLIGHT position.

NOTE: The throttle lever and the high pressure fuel cock cable systems are similar to each other, therefore, the following description of the throttle lever system also may be applied to the high pressure fuel cock system.

F. Cable System

The throttle lever cable system consists of a single run from each throttle lever at the center pedestal console to a sector located on the left side of each engine. However, the left engine cables are routed through the left main landing gear wheel well because the sectors are located on the outboard side of the engine nacelles. Aircraft 94 - 200, 322 and 323 excluding 114 and Aircraft having ASC 149 have included a four cable, two turnbuckle throttle control cable for the left engine. This change is to provide improved capability for removing worn throttle control cables in the left engine. The cables to the right engine are not routed through the wheel well because the sectors are located on the inboard side of the engine nacelles. The cables used are standard 3/32-inch diameter stainless steel 7 X 7 cables. One leg of each cable run connects the cockpit console sector to the sector in the related nacelle. A second cable connects the nacelle sector back to the cockpit console sector, thereby forming a loop between the two sectors. Any movement of the throttle levers in the cockpit is transmitted by these cables to the sectors in the nacelles.

NOTE: The following description of the throttle lever-to-nacelle interconnection applies to both engines.

The throttle lever in the cockpit is attached to a sector that is located within the center pedestal console. A short length of cable is connected to the throttle lever pedestal sector and is routed under the cockpit floorboard, terminating in the entrance door compartment. A second, longer length of cable is joined to the short cockpit pedestal sector cable by means of an adjuster. The longer cable is routed aft under the cabin floorboards and is directed outboard through the skin of the fuselage in the area of the front wing beam. A cable pressurization seal is provided on the outboard side of the fuselage to prevent the loss of cabin pressure air. The cable runs along the front beam of the wing, and is attached to the sector in the nacelle by means of a standard terminal ball end that fits into the mating cavity of the sector. A second cable run connects the opposite sides of the pedestal and nacelle sectors and follows the same general path as other cable. Therefore, any movement of the throttle lever will result in one cable being pulled, thus causing the nacelle sector to change to a new position corresponding with that of throttle lever. Cables are routed around pulleys that are located in the nacelle sector box, along the front wing beam and in the fuselage. Two inboard sets of cables are the throttle LEVER cables. Cable tension is adjusted by means of the special adjusters that are located under the floorboards in entrance door compartment.

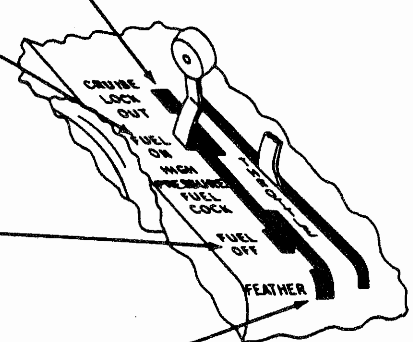
Sector alignment pin holes are provided at the cockpit console pedestal and at the nacelle sectors. The cockpit console pedestal alignment hole is located at the left side of the console pedestal. The left engine nacelle sectors can be aligned by opening the access cover on the left side of the nacelle. An alignment pin can be inserted from the outside of the nacelle so that it passes through the alignment hole in the nacelle structure and the throttle lever and/or high pressure fuel cock nacelle sectors. When aligning the

right engine nacelle sector, an alignment pin is inserted from the inside of the nacelle so that it passes through the alignment hole in the sector box and the alignment hole of the sectors.

The throttle lever system has been provided with a friction unit which ensures that the synchronizing motor operates the engine throttle lever and not the pilots throttle lever, by enabling additional friction to be applied to the throttle system. This unit is installed adjacent to the throttle lever sector at the nacelle sector box. On initial installation, the friction adjustment nut is slowly tightened until 7 pounds of friction is produced in the system after initial rigging of the control cables directly to the engine.

CAUTION: WHEN ADJUSTING FRICTION PLATE IN THE NACELLE SECTOR BOX, ENSURE COCKPIT FRICTION KNOB IS FULLY RELEASED.

FUEL	PROPELLER
ON	CONSTANT SPEED- 34-1/2° LOCK MANUALLY REMOVED
ON	CONSTANT SPEED
OFF	CONSTANT SPEED
OFF	MANUAL FEATHER



High Pressure Fuel Cock Positions
Figure 1.

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ENGINE CONTROL SYSTEM — MAINTENANCE PRACTICES

1. **Engine Control System — Inspection**

   A. Inspect controls for distortion, security and locking.

   B. Inspect all throttle control rods for excessive play caused by working rivets. Replace any rod with working rivets.

   C. Inspect security of ball and socket fittings while moving controls through their full range. Replace any worn ball sockets.

NOTE: Lubricate system bolts, bearings, bushings and bellcranks with water displacing lubricant (e.g,. LPS 1 OR LPS 3). Bushings and bolts may be lubricated with MIL-L-25681 (Royco 81 MS). Lubricate specific components of engine control system in accordance with procedures listed in Chapter 12.

2. **High-Pressure Fuel Cock Control System — Rigging**

(See Figure 201)

NOTE: Rigging of the high-pressure fuel cock control system is divided into four parts; CRUISE LOCKOUT position, FEATHER position, FUEL OFF position, and FUEL ON position.

CAUTION: ACCURATE RIGGING OF THIS SYSTEM IS ESSENTIAL IN ORDER TO ENSURE THE HIGHEST DEGREE OF SERVICE RELIABILITY.

A. Rigging for CRUISE LOCKOUT position (Gust Lock OFF, all throttle and HP cock levers forward).

(1) Ensure low pressure fuel shutoff valve is closed.

(2) The high-pressure fuel cock lever is now in the CRUISE LOCKOUT position.

(3) Insert a 3/8 inch x 11 1/2 inch rigging pin through cockpit control pedestal and high-pressure fuel cock sector alignment holes.

NOTE: The cockpit pedestal alignment hole is located on the left side of the pedestal assembly.

(4) Position high-pressure fuel cock nacelle sector so that rigging pin holes in sector and sector box are aligned. Insert a 3/16 inch x 4 1/4 inch rigging pin through holes.

NOTE: The high-pressure fuel cock alignment hole in the left engine nacelle sector box is made accessible by opening the access cover on the left side of the nacelle. The alignment hole for the right engine high-pressure fuel cock is located at the aft end of the sector box and is accessible from inside the nacelle only.

(5) Remove center floor boarding of the entrance door compartment to gain access to cable adjusters (FLR No. 2).

(6) Adjust cable tension to 40-50 pounds. Cable tension may be measured in area adjacent to cable adjusters. Adjusters should be safety-wired.

NOTE: Each cable loop should be adjusted with the same amount of tension on each cable. If this is not accomplished, the sectors will rotate after the rigging pins are removed and the position of the control pedestal and nacelle sectors will not be in proper relationship with each other.

(7) Move engine control box (pickup) lever to CRUISE LOCKOUT (LO) position and secure.

(8) Insert rigging pin through alignment holes located in bellcrank and synchronizer motor bracket on right engine; on bellcrank support bracket on left engine.

(9) Connect all high-pressure fuel cock linkage between engine control box and nacelle sector box.

CAUTION: WHEN ADJUSTING ROD ENDS, CARE MUST BE TAKEN TO ENSURE THAT ROD END IS NOT THREADED OUT PAST THE INSPECTION HOLES.

(10) Adjust rod ends of push-pull rods so that bolts may be freely inserted.

(11) Remove rigging pins at cockpit control pedestal, nacelle sector box and bellcrank.

(12) Operate high-pressure fuel cock lever through its full range several times, then place forward and check insertion of pedestal rigging pin. if it can be replaced without forcing, it is rigged correctly.

(13) Remove pins.

B. Rigging for the FEATHER position.

(1) Move high-pressure fuel cock lever aft to FEATHER position, and insert a rigging pin in control pedestal to lock high-pressure fuel cock sector in FEATHER position.

(2) Check that there is no clearance between engine control box high-pressure fuel cock (pickup) lever and its stop.

NOTE: This may be changed at discretion of the engine manufacturer; therefore, always check latest instructions in the engine manufacturer handbook.

(3) Adjust vertical terminals on bellcranks located just forward and aft of firewall, to obtain required clearance at stops.

(a) To decrease clearance, thread terminal ends into bellcranks.

(b) To increase clearance, thread terminal ends out of bellcranks.

NOTE: Adjustments should first be made at the forward bellcrank. If further adjustments is required, adjust aft bellcrank terminal ends.

(4) Remove rigging pin at cockpit control pedestal.

C. Adjusting high-pressure fuel cock lever to O (Open).

NOTE: Red position markings are scribed on the top inboard side of the engine control box. These may be seen by looking down between the engine mount tubes. The forward stop is scribed with the letter F (feather) and the aft stop with the letters LO (lockout). The letter S (shutoff) and O (open) are scribed between the two stops. Indication marks are scribed adjacent to the letters S, O.

(1) Move high-pressure fuel cock lever on cockpit control pedestal forward until rear of the stop on engine control box aligns with scribe mark at O position.

NOTE: This operation requires two men, one in the cockpit and one at the engine control box who must signal when the stop and the scribe mark are aligned.

(2) Adjust movable stop in control pedestal to lock high-pressure fuel cock lever in detent. Secure movable stop.

D. Adjusting high-pressure fuel cock lever to S (Shutoff).

(1) Move high-pressure fuel cock lever on cockpit control pedestal aft until rear of stop on engine control box aligns with scribe mark at S position.

NOTE: This operation requires two men, one in the cockpit and one at the engine control box who must signal when the stop and the scribe mark are aligned.

(2) Adjust movable stop in control pedestal to lock high-pressure fuel cock lever in detent. Secure movable stop.

NOTE: The high-pressure fuel cock lever should be moved several times through its complete range of travel. Another clearance check should be taken in the FEATHER position to ensure that the proper clearance did not change. In addition the engine control box should be checked to ensure that the stop and scribe marks are aligned to correspond with the position of the high-pressure fuel cock handle. The aircraft high-pressure fuel cock linkage is properly adjusted if these checks are successful. If a check of the engine clearance (forward of the engine control box) is made and found to be incorrect and the aircraft system is checked as described above and found to be correct then further adjustments should be made on the engine linkage.

3. **Throttle Lever Control System — Rigging**

(See Figure 201)

A. Rig and adjust throttle lever control cables between cockpit pedestal and nacelle sectors. This is accomplished in a manner similar to that described for high-pressure fuel cock system.

(1) Ensure low pressure fuel shutoff valve is closed.

(2) Move gust lock off, all throttle and HP cock levers to its forward position. Insert rigging pin through cockpit pedestal alignment hole, so that it passes through alignment holes in cockpit pedestal throttle lever sectors.

(3) Move throttle lever sectors in nacelle sector box to position where rigging pin can be inserted through the alignment hole of nacelle sector box and throttle control sector.

(4) Adjust cable tension to 40-50 pounds. The same cautions and notes apply to these cables as applied to the high-pressure fuel cock cable system.

B. Using special tool, 159GT1034-11, lock synchronizer motor in datum position. This step applies to right engine installation only. (See Figure 201)

NOTE: Use the forked end of the tool to align the B slot, then reverse the tool and position it on the two studs which are threaded in the motor casing. The key end of the tool must fit in the B slot.

C. Insert rigging pin through alignment hole in throttle lever bellcrank and alignment hole in locking tool on right engine only. For left engine, insert rigging pin through throttle lever bellcrank and bellcrank support bracket.

D. Connect throttle lever control linkage between pinned bellcrank and nacelle sector box.

E. Adjust rod ends so that bolts may be freely inserted.

F. Install all throttle control linkages from engine control box to engine firewall. Move push-pull rods so that engine control box throttle lever linkage is held against maximum rpm stop.

NOTE: This stop can be seen by looking up at the bottom of the engine control box.

G. Remove rigging pins from cockpit control pedestal sector, nacelle sector and bellcrank.

H. Move throttle lever to aft retarded position, and insert rigging pin in cockpit control pedestal to pin throttle lever sector in retarded position.

I. Check at engine control box for position of throttle lever stop and of minimum rpm control box stop.

J. To adjust for proper seating of throttle lever stop against minimum rpm stop, proceed as follows:

(1) Turn vertical terminal of aft bellcrank out to increase clearance between stops.

(2) Turn vertical terminal of aft bellcrank in to decrease clearance between stops.

(3) If further adjustment is necessary, vertical terminal ends of bellcrank forward of firewall should be adjusted in a similar manner.

K. Move throttle lever throughout its complete travel several times and check that engine control linkage makes contact with stop on engine control box.

L. To ensure that throttle linkage can be moved to fully open position when synchronizer motor has failed in fully retarded position, proceed as follows:

(1) Move throttle lever to fully open position and hold it there for all subsequent steps.

NOTE: This operation requires two men, one in the cockpit and one at the engine control box who must force the linkage aft.

(2) Force throttle linkage at engine control box aft.

NOTE: This rotates the synchronizing motor to the retarded position and moves the engine control box linkage away from the maximum rpm stop.

(3) Move throttle lever to fully forward position.

(4) Check that control box linkage touches maximum rpm stop.

(5) Adjust vertical terminals at bellcranks located forward and aft of the firewall if linkage does not reach stop.

4. Engine Control — Static Checks

A. The Static Checks must be carried out (with engine cold or at least 20 minutes after shut down) in any of the following conditions:

(1) After engine installation.

(2) Before an engine serviceability ground run.

(3) Following unit change or other change or disturbance to control system.

(4) As required when investigating a defect.

B. Refer to Rolls-Royce Dart Maintenance Manual (M-Da7-G).

5. Throttle Levers / HP Cock Cable — Tension Check

CAUTION: REMOVE ALL ELECTRICAL POWER FROM AIRCRAFT.

NOTE: Adjust each cable loop to same tension or sectors will rotate after rigging pins are removed resulting in improper relationship between control pedestal and nacelle sectors.

3/16 x 4 1/4 inch rigging pin will be referred to as "short" rigging pin and 3/8 x 11 1/2 inch rigging pin will be referred to as "long" rigging pin for the remainder of this procedure.

A. Power Lever Cables (See Figure 201)

(1) Ensure low pressure fuel shutoff valve is closed.

(2) Position gust lock off and throttle/HP cock levers to forward position.

(3) Insert "long" rigging pin through cockpit pedestal alignment hole (left side pedestal assembly) so it passes through alignment holes in cockpit pedestal power lever sectors.

(4) Remove access cover on left side of left nacelle just above wing leading edge.

(5) Insert "short" rigging pin through alignment hole of nacelle sector box and throttle lever control sector in each engine nacelle.

NOTE: Right sector box is located through right wheel well and left sector box through opening on left side of left nacelle.

(6) Remove FLR No. 2.

(7) Check for cable tension of 40-50 pounds in area adjacent to cable adjusters. (Ensure cable adjusters are safety wired).

(8) inspect for presence of foreign objects then replace FLR No. 2 and remove rigging pins.

(9) Inspect for presence of foreign objects then replace nacelle access cover.

B. HP Cock Cables (See Figure 201)

(1) Position gust lock control to OFF, all throttle levers forward and HP fuel cock to CRUISE LOCKOUT.

(2) Insert "long" rigging pin through cockpit control pedestal (see Step A.(2)) and HP fuel cock sector alignment holes.

(3) Position HP fuel cock nacelle sector so rigging pin holes in sector and sector box are aligned. Insert rigging pin through holes.

NOTE: HP fuel cock alignment hole in left engine nacelle sector box is accessible beneath access cover on left side of nacelle. Alignment hole for right engine HP fuel cock is located at aft end of sector box and is accessible from inside nacelle only.

(4) Gain access to HP fuel cock cables beneath FLR No. 2.

(5) Check for cable tension of 40-50 pounds in area adjacent to cable adjusters. (Ensure cable adjusters are safety wired.)

(6) Inspect for presence of foreign objects then replace FLR No. 2.

(7) Inspect for presence of foreign objects then replace all access covers.

6. Throttle / HP Cock Control Cables — Wear Limits

A. Two broken wires per strand are allowed in any running inch of cable, with a maximum of three broken wires per inch. A maximum of three broken wires is allowed in any two consecutive inches of cable run.

B. A maximum of three broken wires per inch is allowed in any length of cable passing over pulleys and drums or through fairleads.

C. Any wire worn through in excess of 50% of its diameter shall be considered a broken wire.

D. No kinking, untwisting or bird caging of the cables is allowed.

E. if cable damage is in excess of the limits specified, it is recommended that the damaged cable be replaced rather than repaired. The cause for damage (misaligned or damaged pulleys, drums and fairleads or structural interference) should be found and corrected immediately.

7. Throttle Lever and HP Cock Cables - Removal / Installation

A. Removal

(See Figure 201)

(1) Remove appropriate access covers and floorboards.

(2) Remove inbd wing leading edge.

(3) Remove safety wire. Loosen nuts off cable adjusters and remove adjusters and nuts.

(4) Remove all pulleys restricting cable removal.

(5) Remove pressure seal clips, pressure seals and throttle lever cables.

B. Installation

(1) Route cable assemblies in position. Install pressure seals and pressure seal clips.

(2) Install nuts, adjuster and pulleys.

CAUTION: ENSURE ELECTRICAL POWER IS OFF.

C. Adjust throttle lever/HP cock cable tension as follows:

(1) Ensure low pressure fuel shutoff valve is closed.

(2) Move throttle lever, HP cock and gust lock to forward position, insert a 3/8 X 11 1/2 inch rigging pin through cockpit pedestal alignment hole and throttle lever sector.

(3) Insert a 3/16 x 4 1/4 inch rigging pin through alignment holes of nacelle sector box.

NOTE: Left control sector is accessible through access plate on left side of left nacelle, just above wing leading edge. Right control sector is accessible through right wheel well.

(4) Adjust tensions 40 to 50 pounds, ensure holes in nuts align with slots in barrel.

NOTE: Cable tension may be taken at either wing leading edge or under FLR No.2 in entrance compartment.

(5) Remove rigging pins.

D. Safety-wire each adjuster as follows: pass 0.32 safety wire through one hole of the nut. Bring wire up through slot and double twist wire around the outside of barrel to opposite hole in nut. Pass one wire through slot and out through hole in nut. Twist ends into pigtail approximately 6 twist and bend back pigtail, See Figure 201.

E. Operate throttle lever and inspect full length of cable run to ensure that cables are fully seated in pulleys and free of obstruction.

F. Inspect for foreign objects then secure all access covers, floorboards and leading edge.

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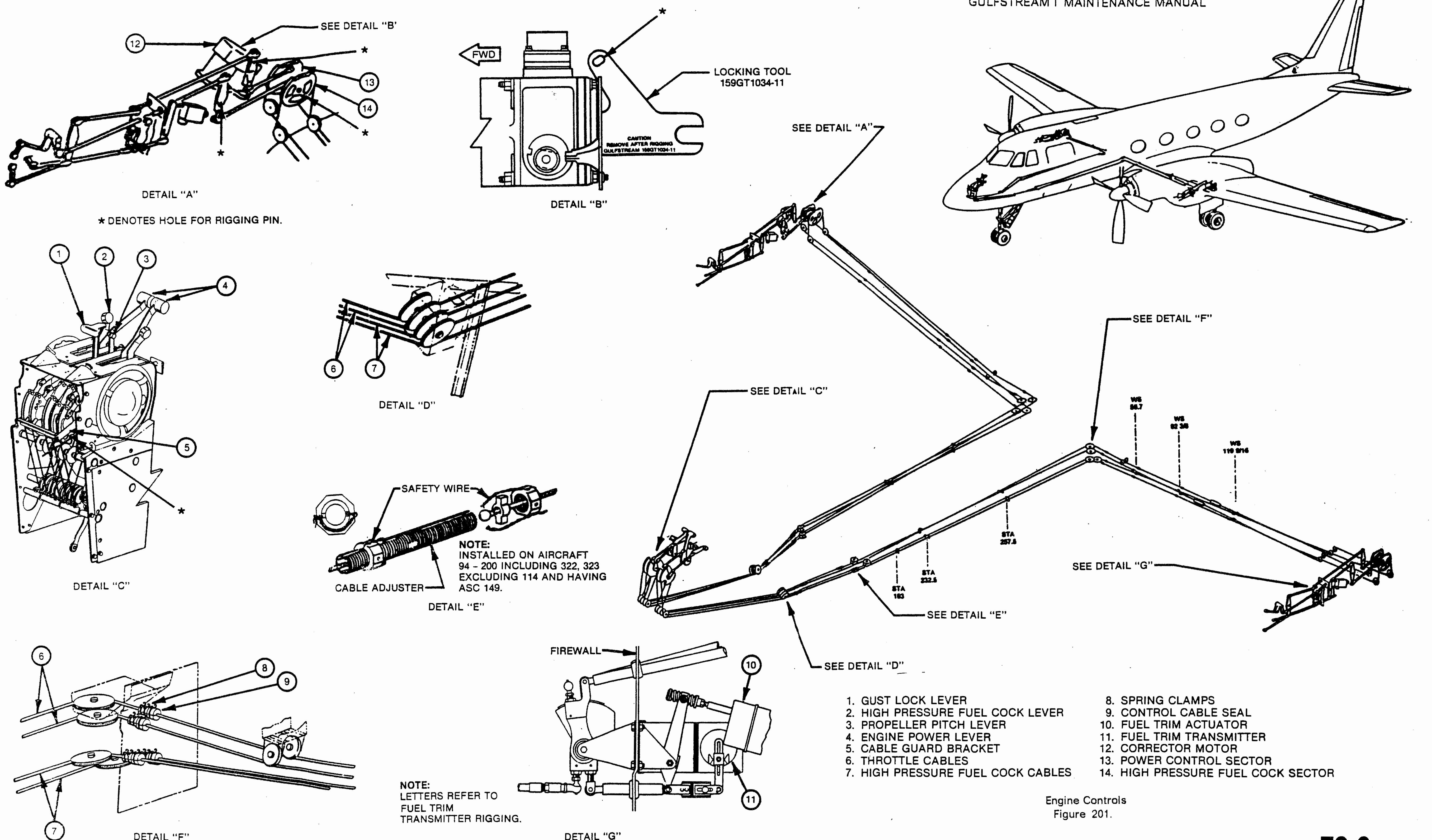


1. GUST LOCK LEVER
2. HIGH PRESSURE FUEL COCK LEVER
3. PROPELLER PITCH LEVER
4. ENGINE POWER LEVER
5. CABLE GUARD BRACKET
6. THROTTLE CABLES
7. HIGH PRESSURE FUEL COCK CABLES
8. SPRING CLAMPS
9. CONTROL CABLE SEAL
10. FUEL TRIM ACTUATOR
11. FUEL TRIM TRANSMITTER
12. CORRECTOR MOTOR
13. POWER CONTROL SECTOR
14. HIGH PRESSURE FUEL COCK SECTOR

Engine Controls
Figure 201.

FUEL TRIM — DESCRIPTION / OPERATION

1. Description

A fuel trim motor is installed in each nacelle and is connected to the engine control box through a series of push-pull rods and bellcranks. A fuel trim switch for each engine is installed on the center pedestal console panel. A dual gage fuel trim datum indicator is installed on the lower portion of the center instrument panel. On hot day, it is necessary to lessen (or trim) the amount of fuel delivered to the engine to prevent the turbine gas temperatures from exceeding the allowable limits. The loss of power accompanying this operation is regained by use of water/methanol injection. The amount that the fuel is trimmed is governed by atmospheric conditions. After engines are started and before taxi, compensation is made for ambient conditions by means of a computer or a fuel trim position chart.

The purpose of this system is control and indication of the fuel to air mixture as suppled to the engines. A switch for each actuator mounted in the pedestal allows the selection of a weak or rich mixture. The actuator in addition to trimming the fuel mixture, mechanically acts upon the transmitter. The transmitter (two, one mounted on left side of each nacelle) consists of a toroidal resistor, over which two contacts move. The toroidal resistor is tapped at 3 places, 120° apart and is supplied with 24V dc. The three taps are routed to the indicator which comprises a two pole permanent magnet motor, pivoted to rotate within a soft iron stator. When the actuator (two, one mounted on left side of each nacelle), acts upon the transmitter, it causes the two contacts to move around the toroid. This in turn causes currents through toroid to change. The change in current causes a change in the magnetic field of the indicator stator, which is manifested by a change in the indicator needle position. Both fuel trim and fuel trim indication are available in emergency dc operation. For more information see Chapter 77.

A. Effect of Ambient Air Temperature

The DART power output is effected by ambient temperature, with the power developed being a function of the temperature drop across the turbines. As ambient temperature decreases engine efficiency, power and fuel flow increase while maintaining the same turbine inlet temperature. In the case of the Dart, however, an ambient temperature is reached where the flow control unit delivers its maximum calibrated flow, with the result being a decrease in engine power output at a constant fuel flow below that temperature. For takeoff power, this change occurs at sea level on a standard day and at 10,000 feet on an ISA 12°C day. To determine whether the engine is developing its correct takeoff power at ambient temperatures where the maximum fuel flow has been reached, (100% fuel trim), TGT corrections are provided which are applied to the maximum power temperature of the engine. These TGT corrections are a function of ambient temperature and pressure and are covered in Chapter 77.

At ambient temperatures where the maximum calibrated fuel flow would result in excessive turbine temperatures a fuel trim is provided which allows fuel to be reduced, thus maintaining TGT at acceptable limits without altering rpm. Under these conditions engine efficiency, fuel flow and power are reduced. The fuel trim is operated from the pedestal by an electric fuel datum control in conjunction with a position indicator. To ensure that satisfactory takeoff power is obtained at ambient conditions where fuel trim position in percent as a function of ambient temperature and pressure.

With the fuel datum set at FULL INCREASE on the indicator gage, there is no reduction in fuel flow. When the fuel datum is set at some lower figure, the fuel flow is proportionately reduced or trimmed, at all settings of the pilots throttle lever. At fuel datum FULL DECREASE (0 indicator reading), fuel flow at any given throttle setting is reduced by approximately 20 percent from that at fuel datum FULL INCREASE (100% indicator reading).

With the fuel datum preset to a position appropriate to the ambient conditions a satisfactory TGT is automatically obtained on opening the pilots throttle lever for takeoff and initial climb. In flight, the fuel datum is adjusted to maintain a TGT approaching the maximum permissible, for the reasons given previously.

The fall in air mass flow under hotter than ISA conditions, also means that less power is produced by the turbine. It is a characteristic of a compressor matched to a turbine that a reduction in mass flow is accompanied by a reduction in the pressure rise achieved by the compressor. It follows that with the

same turbine entry temperature there is on a hot day less pressure and temperature drop across the turbine. This results in a further reduction in turbine power output. While power absorbed by the compressor also falls, this fall is less than the reduction in turbine power and there is, therefore, a decrease in propeller shaft horsepower.

For takeoff purposes, nominal power can be restored by injecting water/methanol. This mixture cools the air at the inlet to the compressor, restoring density and mass flow and enables the methanol burnt in the combustion chambers to restore power.

B. Setting the Fuel Datum

When ambient conditions require fuel trimming to keep the TGT within limits, use is made of the fuel datum position indicator to set the trim after start up and before opening up the engine. For ground running, ascertain the fuel datum position appropriate to the ambient conditions. The Rolls-Royce computer is available for pilot use only. With fuel datum set to the appropriate position for the prevailing conditions, the engine will give an acceptable TGT and power when throttle is opened fully.

To enable fuel datum indicator to be used for controlling engine performance at widely varying ambient conditions, it is essential that the controls should have been correctly adjusted. For consistent operation, trim should always be set from decrease towards increase. Atmospheric conditions may change during a ground run; it is important that frequent checks of the prevailing temperature and pressure altitude are made to ensure that fuel datum position selected is correct for the ambient conditions.

C. Fuel Trim Computer

Rolls-Royce computer R.K. 28677 for Pre Mod 1371 engines or computer R.K. 40916 for Mod 1371 engines, is furnished to the pilot for determining his fuel trim datum position for takeoff and for landing (set for ambient condition). This computer consists of three scales, a temperature scale and a pressure altitude scale which, when aligned, give a resultant in fuel datum position in tenths, which is the third scale.

D. Fuel Trim Position Chart (See Figure 1 for Pre Mod 1371 or Figure 2 for Mod 1371 engines)

The procedure for reading the fuel trim position chart is as follows:

Example 1: A pressure altitude of 1000 feet and a temperature of 27°C. Start at bottom at 1000 feet and continue vertically upward until we intercept a horizontal line from the centigrade temperature (+ 27°C) line going toward the right. At this intersection read the trim position figure, which is 70.

Example 2: Sea level (SL) and a temperature of + 25°C Start at SL and proceed up until we intercept a horizontal line from + 25°C. At that point of intersection our figure is 79.

Example 3: A 3000 foot pressure altitude and a temperature of + 1°C. Start at 3000 feet and proceed up until we intercept a horizontal line from + 1°C. At the intersection we are in the full increase area (100°), consequently, no trim-off is necessary.

NOTE: Rule of thumb: At sea level (SL), temperatures below ISA (+ 15°C) do not require trim off. E. Emergency DC Operation

Both fuel trim and fuel trim indication are available in emergency dc operation.

E. Use of TGT Correction Chart

(See Figure 3)

The procedure for using the TGT correction chart is as follows:

Example 1: A 3000 foot pressure altitude and a temperature of + 1°C. Start at 3000 feet and proceed upward until we intercept a horizontal line from + 1°C. At the intersection we read the TGT correction of 14°C. This correction is then subtracted from the engine maximum power temperature to give the corrected maximum power temperature for the prevailing ambient conditions.

Example 2: At sea level and a temperature of + 19°C. At the intersection the reading is in the zero range. Therefore, fuel trimming will be necessary to maintain the engine at its maximum power temperature. For these ambient conditions refer to either See Figure 1 (Pre Mod 1371) or See Figure 2 (Having Mod 1371) will determine the correct fuel trim position. Figure 1 Pre Mod 1371 - 92% Figure 2 Having Mod 1371 - 93%

° C	PRESSURE ALTITUDE - FT.														
	-1000	-500	-250	SL	500	1000	2000	3000	4000	5000	6000	7000	8000	9000	10,000
51	40	33													
49	44	37	34	30	28										
47	49	41	38	34	32	29									
45	53	45	42	38	36	33	29								
43	57	50	46	42	40	37	33	28							
41	61	54	51	47	44	42	37	32	27						
39	66	58	55	51	48	46	41	36	31	26					
37	70	62	59	55	52	50	45	40	34	29	24				
35	74	66	63	59	57	54	49	44	38	33	28	23			
33	78	71	67	63	61	58	53	47	42	37	32	26	21		
31	82	75	71	67	65	62	57	51	46	41	35	30	25	20	
29	87	79	75	71	69	66	61	55	50	45	39	34	28	24	18
27	91	83	80	75	73	70	65	59	54	49	43	38	32	27	22
25	95	87	84	79	77	74	69	63	58	53	47	42	36	31	25
23	100	91	88	83	81	78	73	67	62	56	51	45	40	35	29
21		95	92	87	85	82	77	71	65	60	54	49	43	38	33
19		100	96	92	89	86	81	75	69	64	58	53	47	42	36
17			100	96	93	90	84	79	73	68	62	57	51	46	40
15				100	97	94	88	83	77	72	66	60	55	49	44
13					100	98	92	87	81	75	69	64	58	53	47
11						100	96	91	85	79	73	68	62	57	51
9							100	95	88	83	77	72	66	60	55
7								98	91	87	81	75	70	64	58
5								100	94	91	85	79	73	68	62
3									98	94	88	83	77	71	66
1									100	98	92	87	81	75	69
-1										100	96	90	84	79	73
-3											100	94	88	82	76
-5												98	92	86	80
-7												100	96	90	84
-9													100	93	87
-11														97	91
-13														100	95
-15															98
-17															100

Fuel trim Position Chart Pre Mod 1371
Figure 1.

°C	PRESSURE ALTITUDE - FT.														
	-1000	-500	-250	SL	500	1000	2000	3000	4000	5000	6000	7000	8000	9000	10,000
51	48	41													
49	52	45	42	39	38										
47	56	49	46	43	41	39									
45	59	53	50	47	44	42	38								
43	63	56	53	50	48	46	43	36							
41	67	60	57	54	52	49	45	40	35						
39	71	64	61	58	55	53	49	43	39	35					
37	75	67	64	61	59	57	52	47	42	38	34				
35	78	71	68	65	62	60	56	50	46	42	37	32			
33	82	75	72	69	66	64	59	54	49	45	40	35	31		
31	86	78	75	72	70	67	63	57	53	48	44	39	34	30	
29	89	82	79	76	73	71	66	61	56	52	46	42	37	33	29
27	93	85	82	79	77	74	70	64	60	55	50	46	41	36	32
25	96	89	86	83	80	78	73	68	63	59	54	49	44	40	35
23	100	92	89	86	83	81	76	71	67	62	56	52	47	43	38
21		96	92	90	87	85	80	75	70	65	60	56	51	46	41
19		99	96	93	90	88	83	78	73	69	64	59	54	49	45
17		100	99	96	94	91	86	81	77	72	67	62	57	53	48
15			100	100	96	95	90	85	80	75	70	66	61	56	51
13					100	98	93	88	83	79	74	69	64	59	54
11						100	96	92	87	82	77	72	67	62	58
9							100	95	90	85	80	76	71	65	61
7								98	93	88	84	79	74	69	64
5								100	97	92	87	82	77	72	67
3									100	95	90	85	80	75	70
1										98	93	89	84	78	74
-1										100	96	92	87	82	77
-3											99	95	90	85	80
-5											100	98	93	88	83
-7												100	96	91	86
-9													100	94	89
-11														97	93
-13														100	96
-15															99
-17															100

Fuel Trim Position Chart Having Mod 1371
Figure 2.

°C	PRESSURE ALTITUDE - FT														
	-1000	-500	-250	SL	500	1000	2000	3000	4000	5000	6000	7000	8000	9000	10,000
21	5														
19	11	0													
17	17	7	0												
15	23	12	6	0											
13	28	18	12	6	2										
11	34	23	17	11	7	3									
9	39	29	23	17	13	9	0								
7	44	34	28	22	19	14	6								
5	49	39	34	27	24	20	11	3							
3	54	44	39	33	30	25	17	9	0						
1	59	49	44	38	35	31	22	14	6						
- 1	64	54	49	43	40	36	27	20	11	3					
- 3	69	59	54	48	45	41	32	25	16	8	0				
- 5	74	64	59	54	50	46	37	30	22	14	6				
- 7	79	69	64	59	55	51	43	35	26	19	11	3			
- 9	84	74	69	64	60	56	48	40	32	24	16	8			
-11	88	79	74	69	65	61	53	45	37	29	21	13	5		
-13	93	84	79	73	70	66	58	50	42	34	26	18	10	3	
-15	98	89	84	78	75	70	63	55	47	39	31	24	15	8	
-17	103	93	88	83	79	75	68	60	52	44	36	29	20	13	4
-19	107	98	93	88	84	80	72	64	57	49	41	34	26	18	9
-21	112	103	97	93	89	85	77	69	62	54	46	39	31	23	14
-23	116	107	102	97	93	89	82	74	66	59	51	44	36	28	19
-25	121	112	107	102	98	94	87	79	71	64	56	49	41	33	24
-27	126	116	111	107	103	99	91	84	76	69	61	53	46	38	29
-29	131	121	115	111	107	103	96	88	81	73	65	58	50	42	33
-31	136	125	120	115	111	107	100	92	85	78	70	62	55	47	38
-33	140	130	124	119	115	112	104	97	89	82	74	67	59	51	42
-35	145	134	129	124	120	116	108	101	93	86	78	71	63	55	46
-37	149	138	133	128	123	120	112	104	97	90	82	75	67	59	50
-39	154	142	137	132	128	124	116	108	101	93	85	79	70	62	54

TGT Correction Chart
Figure 3.

FUEL TRIM — MAINTENANCE PRACTICES

1. Fuel Trim Actuator — Removal / Installation

(See Figure 201)

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Remove left accessory section access panel. (29/30-LH-2)

(2) Remove safety wire and disconnect electrical plug.

(3) Disconnect trim rod.

(4) Remove actuator.

B. Installation

(1) Install actuator.

(2) Connect electrical plug and safety wire.

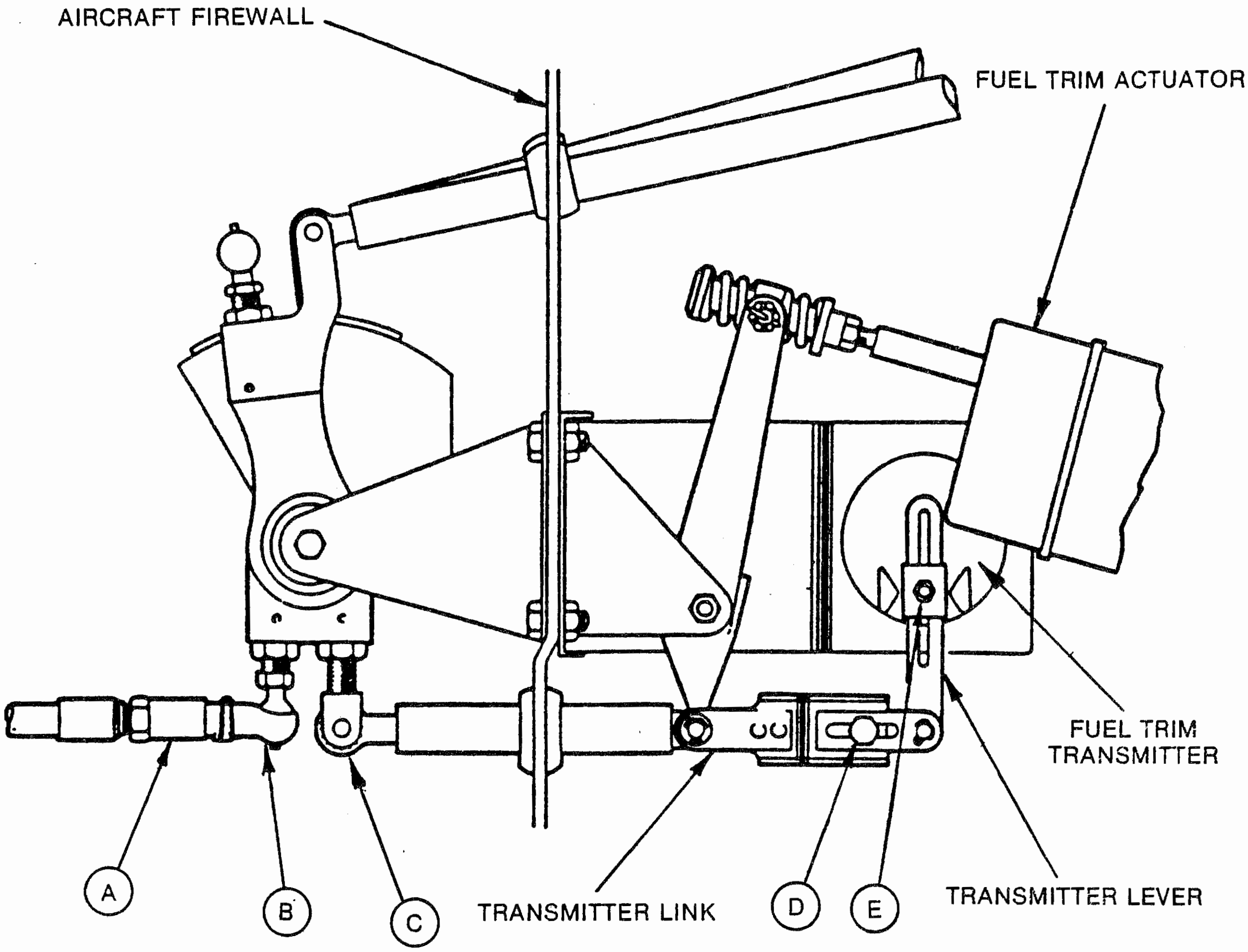
(3) Connect trim rod.

C. Perform Fuel Trim System — Adjustment and Fuel Trim System — Operational Test, Chapter 77.

NOTE: Fuel trim actuator system should be checked and adjusted for range after power lever on HP fuel cock has been rigged and checked.

D. Inspect area for presents of foreign objects, security of all attachments.

E. Install left accessory section access panel. (29/30-LH-2)



Engine Fuel Trim System
Figure 201.

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CHAPTER 77

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CHAPTER 77

ENGINE INDICATING

TACHOMETER INDICATING SYSTEM — DESCRIPTION / OPERATION

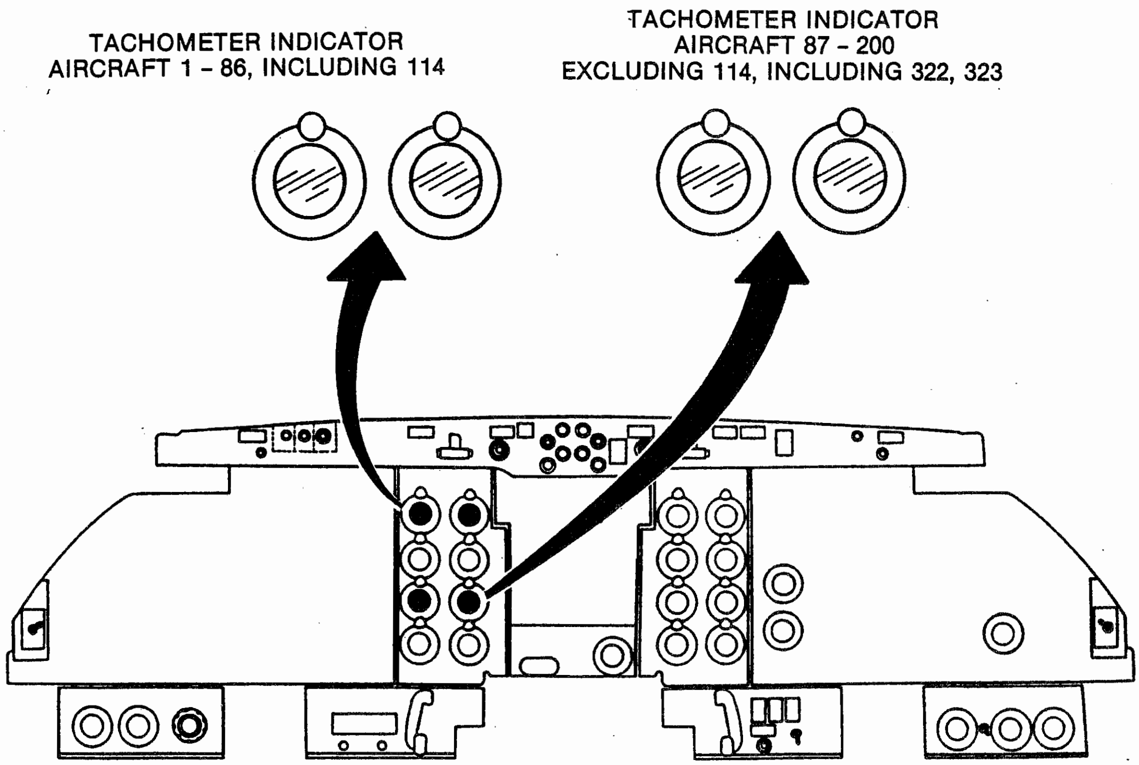
1. Description

The tachometer indicating system provides cockpit indications of engine rpm. Two tachometer indicators, one for each engine, are mounted in the left center instrument panel. (See Figure 1). These are double pointer instruments capable of indicating up to 20,000 rpm. A tachometer generator is mounted on each engine-driven accessory gearbox. Each indicator is connected to its respective tachometer generator by three wires. The electric tachometer system operates on low frequency alternating current. (See Figure 2). Since the tachometer generator is geared to the engine-driven accessory gearbox, it is driven at a speed proportional to the engine. The tachometer generator will generate three-phase alternating current at a frequency proportional to engine speed. The three-phase alternating current is supplied to the tachometer indicator whose speed is dependent on the frequency of the input voltage and not on the load. With any change in engine speed, there will be a like change in motor speed. The motor drives a magnetic assembly which actuates the pointer, causing it to move over a calibrated dial to provide a visual indication of the speed of the engine as measured in revolutions per minute. Since the tachometers do not depend on dc power, they will be operative in emergency dc operation.

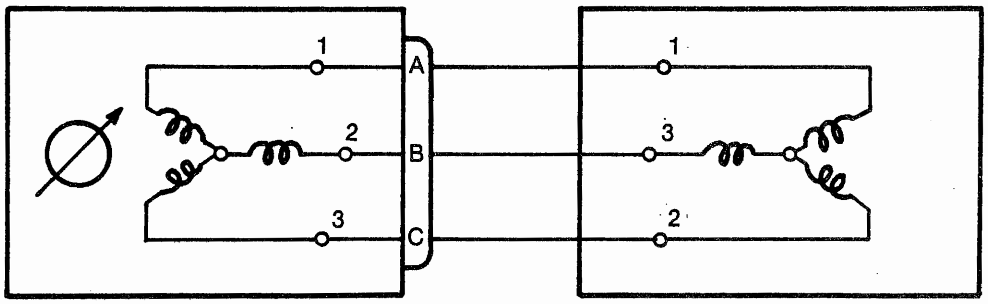
The instruments are marked as follows:

- 15,000 rpm - red line
- 5500 rpm - red line
- 5500 to 15,000 rpm - green arc

On Aircraft 1 - 86 and 114 having ASC 115 and Aircraft 87 - 200, 322 and 323 positions of the tachometer indicators and TGT indicators are interchanged.



Tachometer Location in Cockpit
Figure 1.



RIGHT SHOWN, LEFT IDENTICAL

Tachometer System Circuit — Schematic
Figure 2.

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TACHOMETER INDICATING SYSTEM — FAULT ISOLATION

FAULT	PROBABLE CAUSE	CORRECTION
Pointer turns in wrong direction.	Reversal of any two leads.	Check and connect leads in proper position.
Indicator fails to operate.	Defective or broken tachometer generator.	Replace tachometer generator.
	Open circuit between generator and indicator.	Repair or replace wiring.
	Defective indicator mechanism.	Replace indicator.
Pointer oscillates.	Loose or defective connections and leads.	Tighten or replace connections and leads.
	Defective indicator or generator.	Check and replace defective unit.

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TACHOMETER INDICATING SYSTEM — MAINTENANCE PRACTICES

1. **Tachometer Indicator — Modification No. .06**

   (Applies only to indicators having Smith Service Bulletin B-018)

   A. Some tachometer indicators have been modified in accordance to Smith's Service Bulletin B-018, Modification No. .06. This change extensively modifies units and among other items installs sealed ball bearings. With incorporation of this change, lubrication becomes unnecessary and undesirable.

   B. Sealed bearing instruments may be recognized by observation of these factors:

      (1) The .06 Mod. plate on nameplate will be marked.

      (2) There will be no oiling record nameplate (or tag).

      (3) There will be no front bearing oiling screw. On some instruments with ball bearings, the outer case hole remains, but there is no screw at bottom of it.

   C. This modification may be done by any certified instrument overhaul agency.

   CAUTION: DO NOT LUBRICATE THIS INSTRUMENT.

2. **Lubrication — Tachometer Indicator**

   (Applies only to indicators not having Smith Service Bulletin B-018, Mod. No. .06)

   A. **In Service** - Refer to Maintenance Manual, Chapter 5

   B. **In Storage** - Lubrication is required every 12 months. If indicator has been in storage more than 4 months, it must be lubricated before installation in aircraft.

   C. **Front Bearing** - Front bearing is lubricated through a small tube aligned with a hole in side of case. This hole is located approximately 1 1/2 inches to rear of mounting flange and utilizes a headless screw and a sealing washer as a plug.

      (1) Remove headless screw and washer.

      (2) Insert a hypodermic (No. 23 needle) filled with Aircraft instrument oil (Spec MIL-L-6085A) into small tube as far as possible. Deposit 1 drop of oil.

      (3) Install washer and headless screw.

      NOTE: This is an instrument and requires careful handling. All screws should be snugly seated. Do not overtighten. All hardware has British threads and replacement is difficult.

   D. Rear Bearing

      (1) Remove rear cover from case.

      (2) Remove large screw and washer from center of housing.

      (3) Using hypodermic, deposit 1 drop of oil directly to bearing.

      (4) Replace screw and washer.

      (5) Attach rear cover to case (keyway up).

      CAUTION: DO NOT SQUEEZE WIRES BETWEEN CASE AND COVER.

      (6) Mark date and aircraft hours of lubrication on tape and place it on instrument case. Do not obscure nameplate.

3. Lubrication — Tachometer Generator Quill

A. Refer to Maintenance Manual Chapter 5 for lubrication time periods.

(1) Remove tachometer generator and quill.See procedures this section.

(2) Lubricate quill by immersing it in clean Dowty-Rotol approved gearbox oil.

NOTE: Ensure serrations at both ends are well lubricated.

(3) Refit quill while still wet.

(4) Dip tachometer generator shaft serrations in clean Dowty-Rotol approved gearbox oil.

(5) Install tachometer generator while shaft is still wet. See procedures this section.

4. Engine Tachometer Generator — Removal / Installation

A. Removal

(1) Remove gearbox access cover.

(2) Remove safety-wire, hardware and electrical cover plate.

(3) Tag and disconnect electrical leads.

(4) Remove tachometer generator.

B. Installation

(1) Lubricate quill shaft with light coat of oil and Dowty-Rotol approved gearbox oil.

(2) Apply light coat of Dowty-Rotol approved gearbox oil to generator shaft serrations.

(3) Install generator in gearbox. Torque nuts to 90 inch pounds.

NOTE: If gearbox modification plate does not indicate GB1334 and 1440 are incorporated gearbox should be modified at next overhaul. Quill shaft should be lubricated every 100 hours until modified.

(4) Remove tags and connect electrical leads.

(5) Install coverplate and safety-wire screws.

(6) Perform engine run and check tach generator system. See Engine Tachometer Indicator — Functional Test procedure this section.

5. Engine RPM Indicator — Removal / Installation

A. Removal

(1) Remove indicator from panel.

(2) Disconnect electrical connector.

(3) Remove indicator and shield.

B. Installation

NOTE: If indicator being installed is not modified to Smiths service Bulletin B-018 (mod.06). Refer to Lubrication — RPM Indicator procedure this section for lubrication information.

(1) Install shield.

(2) Connect electrical connector.

(3) Install indicator.

(4) Perform engine run and check operation. See Engine RPM Indicator — Functional Test procedure this section.

6. **Engine RPM Indicator — Functional Test**

(Using Fluke 1910A Digital Frequency Counter)

NOTE: Refer to Rolls-Royce Dart Maintenance Manual, M-Da7-G for alternate procedure.

Ensure batteries are fully charged on Fluke 1910A before using.

A. Locate test points for wild frequency alternators behind a spring loaded panel in aisle leading from flight deck to cabin.

CAUTION: FOR SAFETY REASONS IT IS VITAL THAT CORRECT POLARITIES ARE OBSERVED.

B. Ensure 2 prong plug on attenuator/filter P/N Y7201 is connected with proper polarity, i.e. ground side to black socket. (Ground is identified by a raised portion on pin housing bearing the legend "GND").

C. Connect red covered alligator clip to phase A test socket of engine alternator being tested. (A test lead pin may be used to facilitate this connection.) Connect black covered clip to a suitable ground.

D. On front panel of Fluke 1910A, connect Y7201 attenuator filter to BNC connector, turn ATTEN knob fully counterclockwise and set filter switch to DC-IKHZ.

E. Turn tripper level to PRE-SET.

F. Set attenuator switch to ON (Switch button depressed).

G. Set resolution switch to AUTO.

H. On rear panel of Fluke 1910A, ensure CLOCK switch is set to INT.

I. Depress POWER switch, panel should light up.

J. Start engine and bring appropriate wild frequency alternator on line.

CAUTION: DO NOT TURN ON WINDSHEILD HEAT OR PROPELLER/ENGINE DE-ICE.

K. Turn ATTEN knob on Y7201 attenuator/filter slowly clockwise until a steady reading appears on counter.

L. Increase engine RPM until frequency counter indicates 411.82. Engine RPM indicator should read 15,000 ± 100 RPM. Replace indicator if not within limits.

LEFT ENGINE RPM________ FREQUENCY________TECH________INSP________
RIGHT ENGINE RPM________ FREQUENCY________TECH________INSP________

NOTE: Table A or B will be a guide to the turbine RPM versus the frequency counter indication.

RPM indicator tolerance is ± 100 RPM throughout the entire engine operating RPM range.

The following RPM/Frequency tables are only accurate for a Gulfstream GI aircraft installation when checked at the wild frequency alternator test points.

TABLE A			
ENGINE RPM	**FREQUENCY CTR.**	**ENGINE RPM**	**FREQUENCY CTR.**
14,570	400.00	14,861	408.00
14,606	401.00	14,897	409.00
14,642	402.00	14,934	410.00
14,679	403.00	14,970	411.00
14,715	404.00	15,007	412.00
14,752	405.00	15,043	413.00
14,788	406.00	15,079	414.00
14,824	407.00		

TABLE B			
ENGINE RPM	**FREQUENCY CTR.**	**ENGINE RPM**	**FREQUENCY CTR.**
15,000	411.82	11,000	302.00
14,500	398.09	10,500	288.27
14,000	384.36	10,000	274.55
13,500	370.64	9,500	260.82
13,000	356.91	9,000	247.09
12,500	343.18	8,500	233.36
12,000	329.46	8,000	219.64
11,500	315.72		

M. Reduce engine RPM until frequency counter indicates 219.64. RPM indicator should read 8000 RPM.

NOTE: Replace indicator if not within limits. See this Section.

N. Select alternator off.

O. Reduce engine to idle and shut down engine.

P. Remove test equipment.

ENGINE OIL PRESSURE INDICATING SYSTEM — DESCRIPTION / OPERATION

1. **Description**

   A. The engine oil pressure indicating system provides aircraft crew with visual indication of engine oil pressure. The system consists of two pressure transmitters and two oil pressure indicators. (See Figure 2) The 26V ac power for the oil pressure circuits is obtained from individual 115/26V transformers. The transformers are fed from the instrument ac bus through 3/4-ampere circuit breakers. Each oil pressure transmitter consists of two variable inductance coils, which, when wired to two electromagnetic circuits of the indicator, form a bridge circuit to which 26V, 400 cycle ac is applied. The transmitter receives variations in oil pressure which change the inductance of one coil with respect to the other. This causes a current change in the bridge circuit which is reflected by a change in the indicator needle position. The oil pressure indicator is a 2 inch dial, single pointer instrument with a scale of 0 to 40 psi. (See Figure 1)

   B. Oil Pressures

      (1) Minimum acceptance for flight at maximum continuous rpm (15,000), is 12 psi at 120°C.

      (2) Minimum acceptable in flight at all rpms is 9 psi.

      (3) Minimum for runway check at 12,000 rpm is 13.5 psi at 55°C, and 12 psi at 120°C and prorated from 55°C to 120°C.

      CAUTION: THE PROPELLER MUST BE FEATHERED IF THE LOW OIL PRESSURE WARNING LIGHT GLOWS AT 10,400 OR HIGHER RPM, OR IF THERE IS ANY ABNORMAL CHANGE IN OIL PRESSURE. WHEN THE OIL PRESSURE GOES BELOW 6.0 ± 0.5 PSI IT CAUSES THE L OR R OIL PRESS LIGHT TO GO ON IN THE MASTER WARNING SYSTEM.

      NOTE: At oil temperatures between - 15°C and + 55°C, the minimum oil pressure is 13.5 psi at 12,000 rpm.

      The low oil pressure warning light may glow momentarily during violent maneuvers or during conditions of extreme turbulence.

   C. Marking

      (1) 9 psi (red line)

      (2) 9 psi to 40 psi (green arc)

   D. Oil Pressure indicators are operative in emergency.

2. **Major Components and Locations**

   A. Two oil pressure indicators, Smiths. These units are mounted in the right center instrument panel.

   B. Two oil pressure transmitters, Smiths. These units are mounted on the right side of each engine.

   C. 115/26 VAC INST transformers — under FLR No. 8.

3. **Engine Low Oil Pressure Warning**

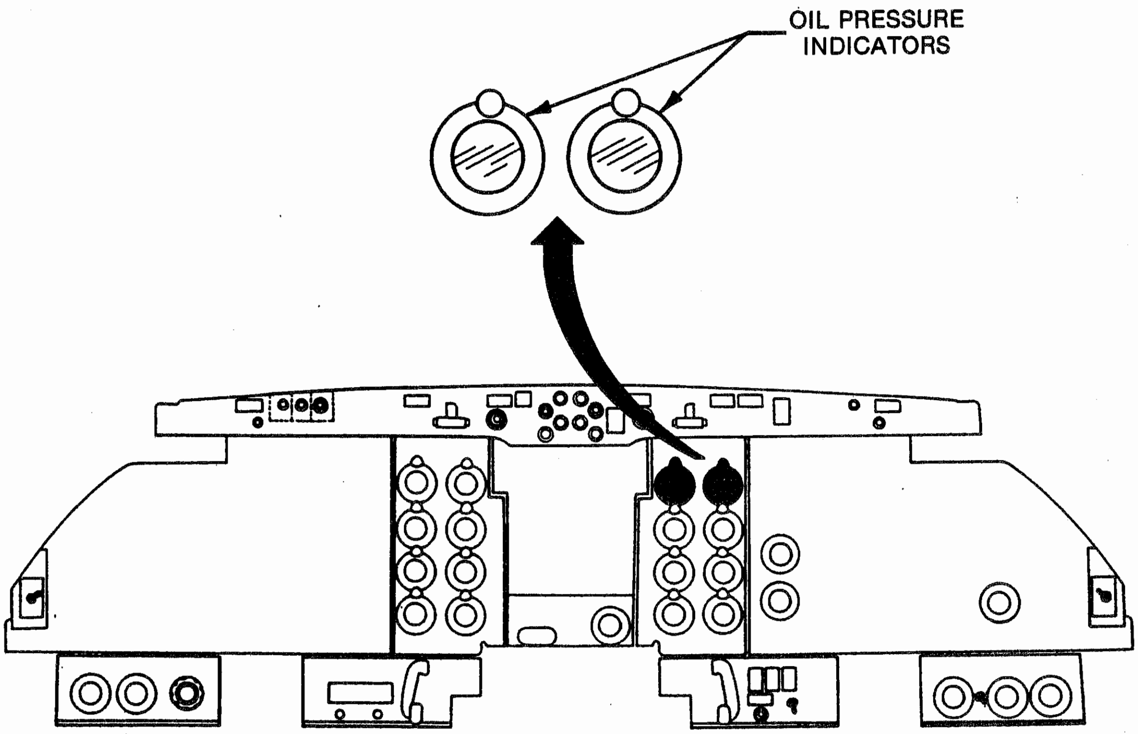
   The purpose of this system is to give the crew a warning in event that the engine oil pressure drops to 6 ± 0.5 psi or lower. This is accomplished by a low oil pressure switch on the engine completing an electrical circuit to the L or R OIL PRESS light of master warning system. Power for each circuit comes from the TORQUE PRESS circuit breaker for the appropriate engine and originates at the essential dc bus. (See Figure 3)

   NOTE: Any defect in this warning system can be immediately determined by observing that engine oil pressure indicator as a cross check to determine whether it is actually low or if the warning system has a malfunction.

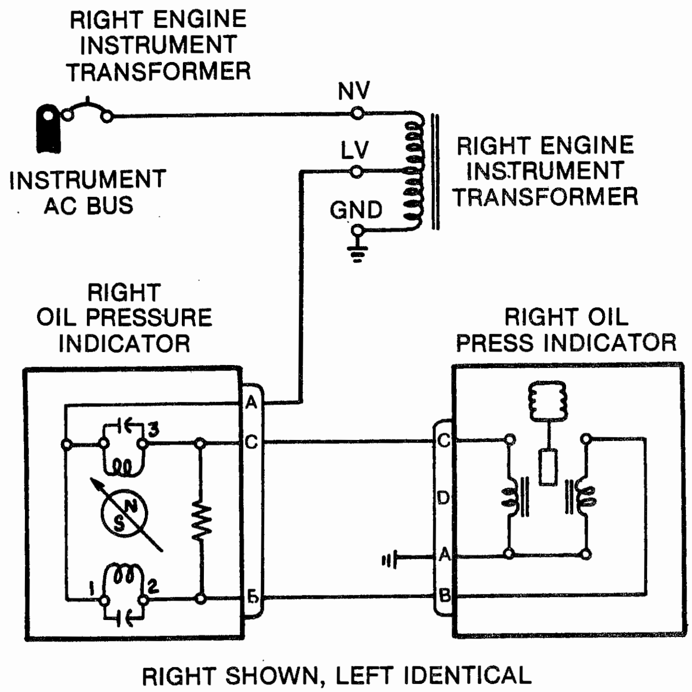
   The engine low oil pressure switches are mounted on the right side of each engine, adjacent to the oil pressure transmitter unit. The sensing unit is a pressure operated diaphragm which is physically connected to a set of electrical contacts. On decreasing oil pressure, the contacts will close and the warning light will come on when the oil pressure drops to 6 ± 0.5 psi through A and D pins of the pressure switch plug. It is furnished with the engine as part of the engine package.

4. Effects of ASC 107

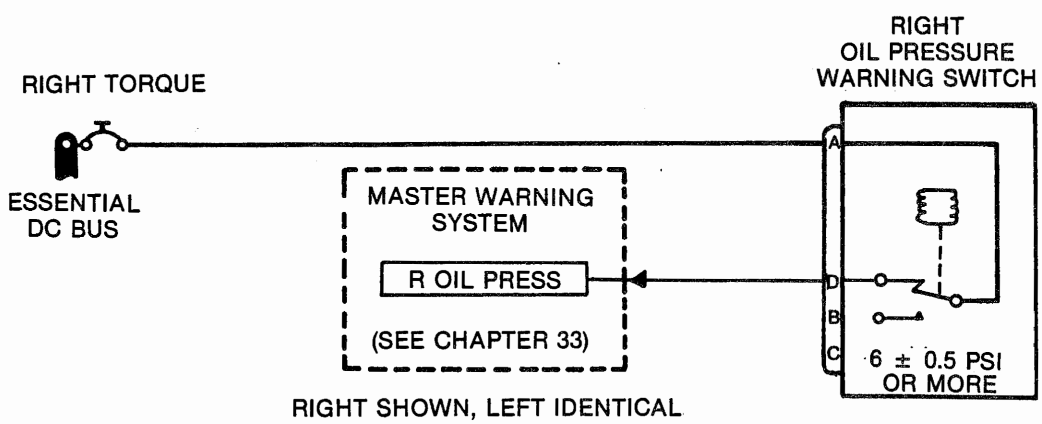
ASC 107 was an optional change for Aircraft 1 - 75, 114 and is standard equipment in Aircraft 76 - 200, 322 and 323. This change places a relay and protecting diodes between the pressure switch and the master warning lights for low oil pressure indication. (See Figure 4) Its purpose is to increase the reliability and service life of the low oil pressure switch by reducing the amount of current flowing through the pressure switch contacts. With ASC 107 installed, the pressure switch actuates a relay, which turns on the appropriate capsule of the master warning system. The relay draws in the order of 55 milliamperes. Since the pressure switch needs only to pass enough current to energize the relay, a substantial reduction of arcing at the pressure switch contact is the result. The relay contacts, controlled by the relay coil, take the heavy 455 milliampere load, which relieves the pressure switch contacts of this function. Two blocking diodes in the relay package prevent energy from the collapsing of the relay field from back tracking to the pressure switch contacts to cause additional arcing. A relay package is installed on its appropriate engine mount, fixed to it with suitable clamping devices. The operation of the system is not changed as far as the indication to the crew is concerned, however, it is believed that the service life of the Dart supplied pressure switch will be appreciably extended with the installation of this service change. See Figure 3 and Figure 4 Illustrate the unmodified and modified systems, respectively. Refer to Rolls-Royce Dart Maintenance Manual (M-Da7-G) Chapter 77 for location of engine installed components.



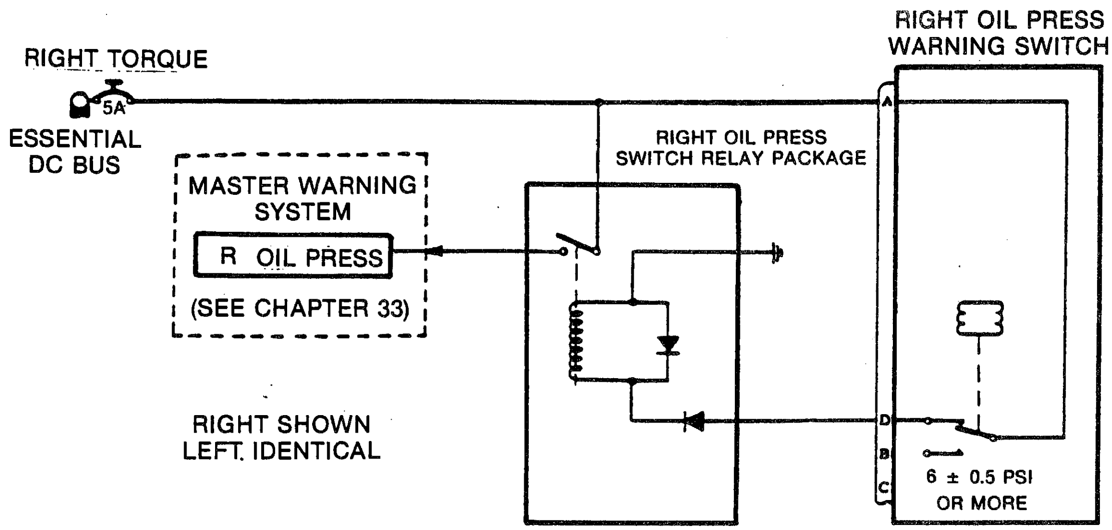
Oil Pressure Indicator Location
Figure 1.



Engine Oil Pressure Indicating — Schematic
Figure 2.



Engine Oil Pressure Warning light Circuit — Schematic
Figure 3.



Engine Oil Pressure Warning light Circuit with Relay Package
Figure 4.

ENGINE OIL PRESSURE INDICATING SYSTEM — MAINTENANCE PRACTICES

1. **Engine Oil Pressure Transmitter and Low Oil Pressure Switch — Removal / Installation**

   NOTE: Refer to Rolls-Royce Dart Maintenance Manual Chapter 77 for procedure.

2. **Engine Oil Pressure Warning Light Relay Package — Removal/ Installation**

   CAUTION: REMOVE ELECTRICAL POWER FROM AIRCRAFT.

   A. Removal

   (1) Open right side cowling aft of firewall on each nacelle.

   (2) Tag and cut electrical leads.

   (3) Remove relay package.

   B. Installation

   (1) Install relay:

   NOTE: Install screws from underside.

   (2) Splice electrical leads (stagger splices).

   (3) Lace tie leads.

   (4) Disconnect electrical connector from oil pressure switch.

   (5) Place jumper across connector pins A and D.

   (6) Energize main and essential dc buses. Oil pressure light (master warning panel) should come on.

   (7) Remove jumper wire. Light should go out.

   (8) Remove electrical power.

   (9) Connect electrical connector and safety-wire

   (10) Install access covers.

3. **Engine Instrument Transformers — Removal / Installation**

   CAUTION: REMOVE ELECTRICAL POWER FROM AIRCRAFT.

   A. Removal

   (1) Remove FLR No. 8 in forward cabin.

   (2) Tag and disconnect electrical leads from transformer.

   (3) Remove transformer.

   B. Installation

   (1) Install transformer.

   (2) Connect electrical leads and remove tags.

   (3) Energize essential dc and instrument ac buses.

   (4) Connect voltmeter between terminals LV and G of transformer and check for 26V ac.

   (5) Remove electrical power.

   (6) Install floorboard.

   (7) Run engines and verify operation of oil pressure indicator and fuel flowmeter.

   (8) Open engine cowl and check for leaks.

4. Engine Oil Pressure Indicator — Removal / Installation

CAUTION: REMOVE ELECTRICAL POWER FROM AIRCRAFT.

A. Removal

(1) Disconnect electrical connector.

(2) Remove indicator.

B. Installation

(1) Install indicator.

(2) Connect electrical connector.

(3) Run engine and check indicator operation.

(4) Open engine cowl and check for leaks.

ENGINE FUEL TRIM INDICATING SYSTEM — DESCRIPTION / OPERATION

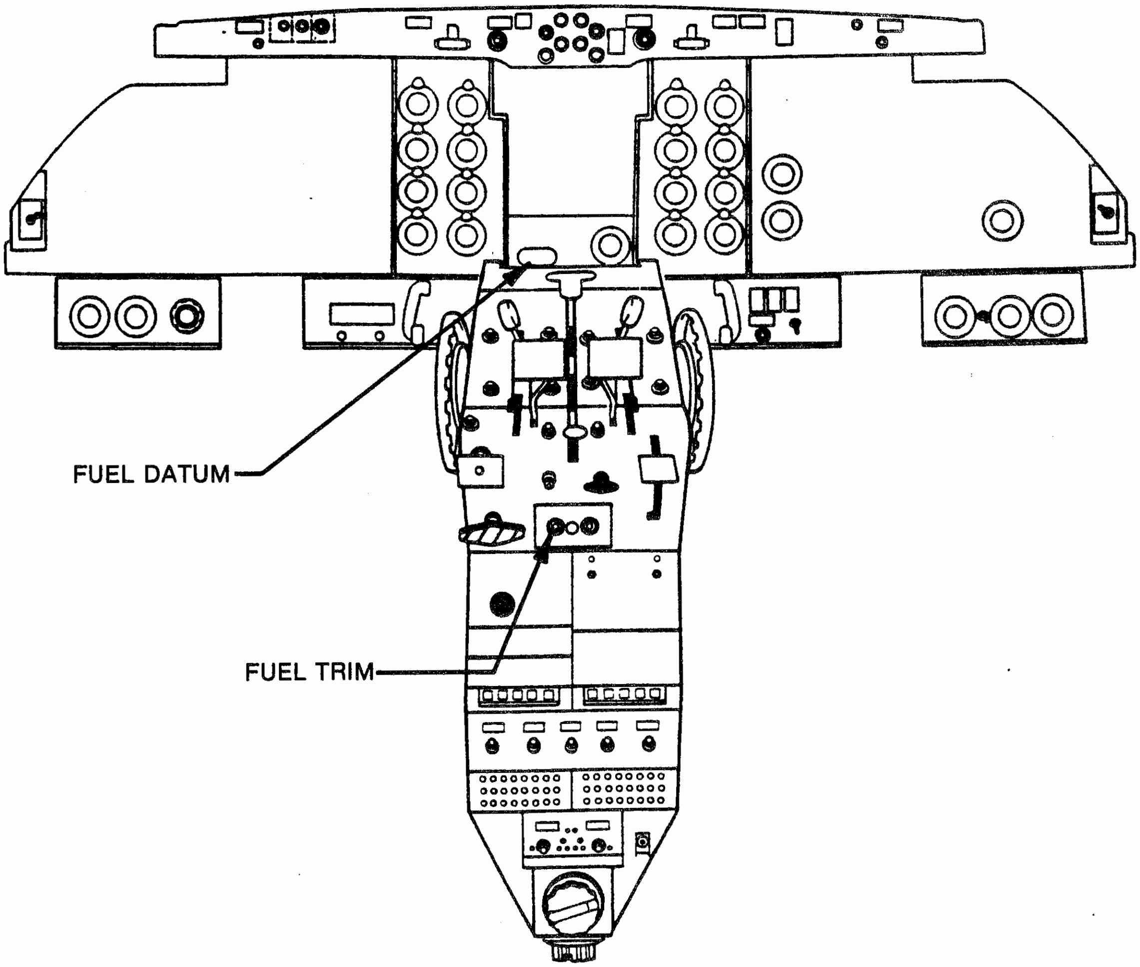
1. Description (See Figure 1).

The fuel trim indicating system consists of two fuel trim actuators, two fuel trim transmitters, dual indicator, necessary wiring and switches to connect the components together. Two switches, one for each actuator, mounted on the cockpit pedestal control console allow engine fuel flow to be either decreased or increased within the range of the trimmer. The actuators, one for each engine, in addition to trimming the fuel flow mechanically act upon the transmitter (See Chapter 76 for additional information on Engine Trim System). The dual indicator gives the crew a visible means of checking trim position.

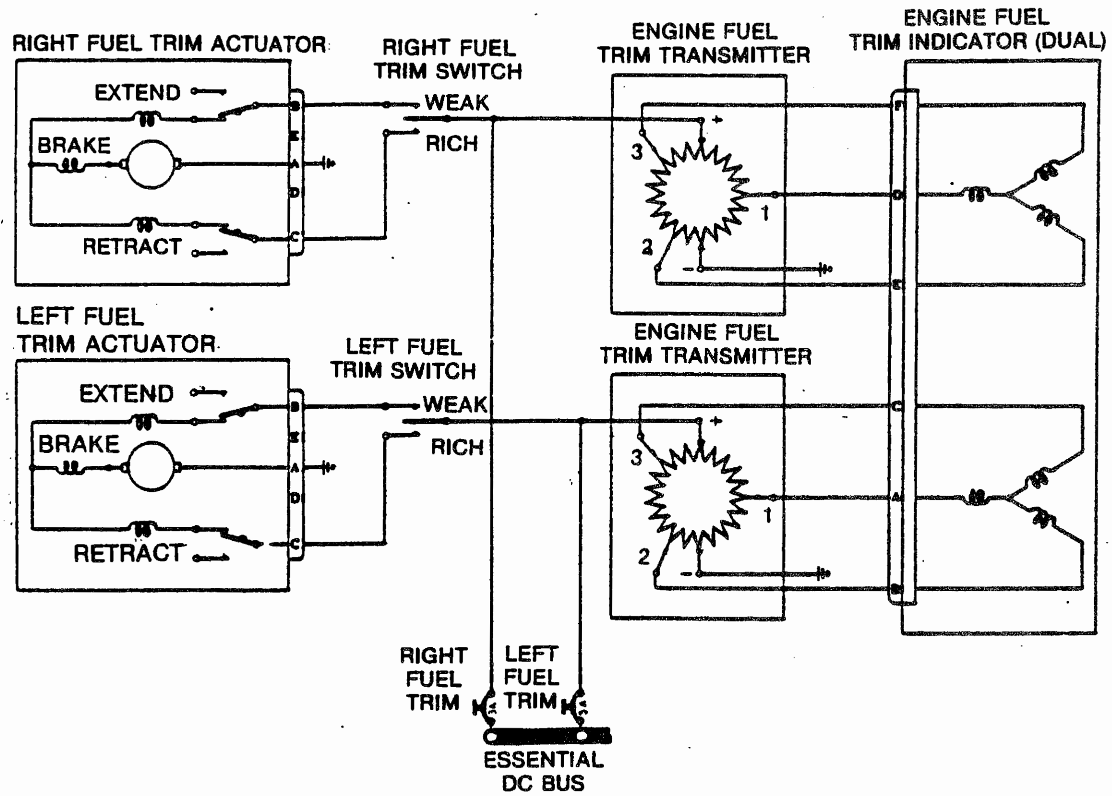
1. **Operation**

(See Figure 2).

The transmitter consists of an endless resistance wound on a ring (toroidal resistor) over which two contacts move. The toroidal resistor is tapped at three places, 120° apart and is supplied with 24V dc. The three taps are routed to the indicator, which is a two pole, permanent magnet rotor that rotates within a soft iron stator. When the actuator acts upon the transmitter, it causes the two contacts to move around the toroid, thus causing the current in the resistor legs to change. This change in current causes a change in the magnetic field of the indicator stator, which is registered as a change in the indicator needle position.



Location of Engine Fuel Trim Indicating System Controls and Indicators
Figure 1.



Engine Fuel Trim Circuit — Schematic
Figure 2.

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ENGINE FUEL TRIM INDICATING SYSTEM — MAINTENANCE PRACTICES

1. Fuel Trim System — Adjustment

(See Figure 201)

A. The fuel trim actuator system should be checked and adjusted for range after the throttle lever on the fuel control has been rigged and checked. To adjust the fuel trim system, proceed as follows:

(1) Disconnect control rod (A) at ball end (B). Trim to full increase and decrease in turn, checking that fuel datum indicator readings correspond at 100% and 0%, respectively.

(2) If pointer does not correspond with marking at one end, adjust transmitter link to make it correspond. Adjust D to left. This will bring needle up to 0 if below. If correct range between 0 and 100% is not obtained on indicator, adjust transmitter lever length at (E), shortening to increase range, and lengthening to decrease range (each serration roughly 2°).

NOTE: When setting up, adjust E first, this will give 180° travel. Then adjust D to center 0% to 100%.

(3) With fuel datum indicator now matched to trim actuator travel, proceed to determine correct length of trim control rod (A).

(4) Using cockpit fuel trim switch, select full increase 100% and position trim pickup lever on engine control box at full increase travel of its quadrant. Adjust length of trim control rod (A) to match forward position of ball end (B), but do not connect.

(5) Select full decrease 0%, using cockpit fuel trim switch, and position trim pickup lever on engine control box a full decrease travel of its quadrant. Check length of trim control rod (A) for length in relation to rearward position of ball end (B).

(6) if rod (A) is too long, revert to full increase position, lengthen ball end (B) and shorten rod (A) to suit.

(7) If rod (A) is too short, revert to full increase position, shorten ball end (B) and lengthen rod (A) to suit.

(8) After either adjustment, check at full decrease and repeat above steps until settings are correct. Only if adjustment at ball end (B) is insufficient, should it be necessary to adjust at (C).

2. Fuel Trim Transmitter — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(See Figure 201)

(1) Remove nacelle access cover.

(2) Apply electric power to aircraft.

(3) Actuate trim switch to full decrease position.

(4) Remove electric power from aircraft.

(5) Disconnect indicator linkage.

(6) Disconnect electrical connector from transmitter.

(7) Remove transmitter.

B. Installation

(See Figure 201)

(1) Install transmitter.

(2) Connect electrical connector to transmitter.

(3) Connect indicator linkage.

(4) Energize essential dc bus.

(5) Actuate FUEL TRIM switch on forward pedestal to full INCREASE and DECREASE position. Check that fuel datum indicator readings correspond at 0% and 100% respectively. If indication does not correspond make following adjustments.

   (a) If pointer does not center between 0% and 100%, adjust transmitter link length (4). Shortening transmitter link will bring pointer up to 0% if below. Lengthening will have opposite effect.

   (b) If pointer does not have full 180° travel, adjust transmitter lever length (5). Shortening will increase range, lengthening will decrease range. Each serration is approximately 2°.

(6) De-energize essential dc bus.

(7) Inspect for presence of foreign objects then install nacelle cover.

3. Fuel Datum Indicator — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Disconnect electrical connector from indicator.

(2) Remove indicator.

B. Installation

(1) Install indicator.

(2) Connect electrical connector to indicator.

(3) Perform Fuel Datum Indicator — Operational Test, see this Section.

4. Fuel Trim System — Operational Test

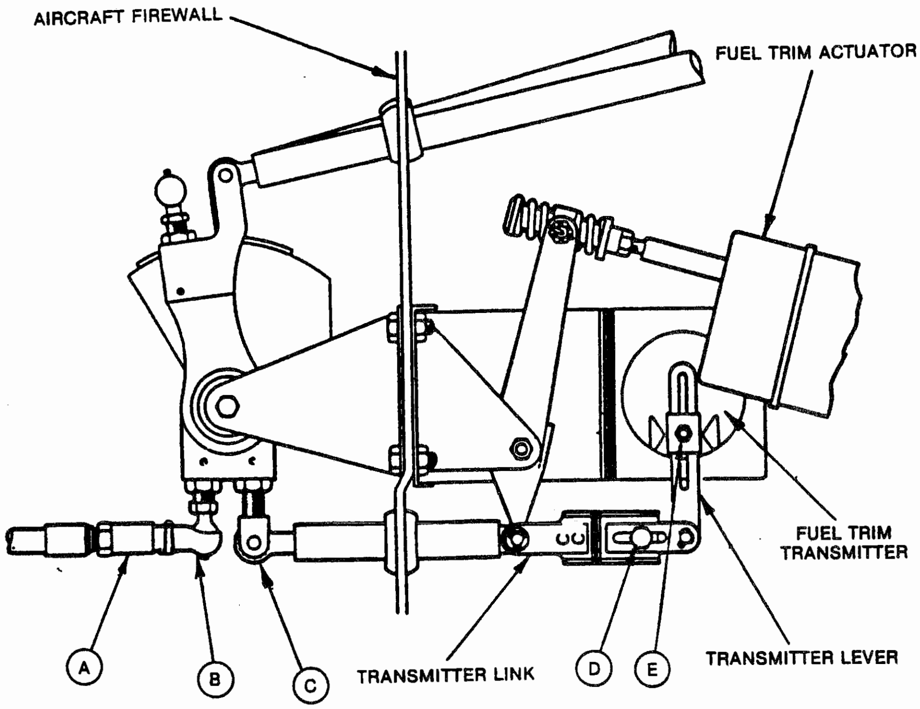
A. Procedure

(1) Apply dc power to essential bus.

(2) Place trim switch to full INCREASE, Indicator should read 100%.

(3) Place trim switch to full DECREASE, Indicator should read 0%.

(4) Remove electrical power.



Engine Fuel Trim System
Figure 201.

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ENGINE OIL TEMPERATURE INDICATING SYSTEM — DESCRIPTION / OPERATION

1. **Description**

The engine oil temperature indicating system consists of two oil temperature indicators located on the right center instrument panel and two temperature sensitive pickup elements (oil temperature bulbs) located on left side of each engine. The indicators read from —50° to +150°C. (See Figure 1) The system employs the Wheatstone bridge principle of measuring a resistance which is varied by temperature changes. This system is powered by 28V dc from the essential dc bus through the left and right 2 ampere OIL TEMP circuit breakers. A resistance type thermometer bulb is installed in the engine oil passage. The resistance of the element varies in proportion to the engine oil temperature. The bridge circuit, of which the element forms one leg, will assume an unbalanced condition corresponding to the particular temperature to which the bulb is exposed. The indicator windings form the other three legs of the bridge circuit, positioning the indicator needle to indicate engine oil temperature in degrees centigrade. The oil temperature indicators are operative in emergency dc operation. (See Figure 2 and Figure 3)

A. Oil Inlet Temperatures

(1) Maximum +120°C.

(2) Minimum for starting −30°C.

(3) Minimum for opening throttle lever above 11,000 rpm: −15° C.

NOTE: External heating will be required to raise oil temperatures to −30°C for cold weather starting. If the oil temperature is less than −15°C, the engine should be idled until this temperature is reached. It is permissible to run at flight idle (11,000 rpm) at which speed the engine will take approximately 3 minutes to warm up from the minimum starting temperature (−30°C) to the opening up temperature −15°C.

B. The instruments are marked as follows:

(1) −30°C - red line

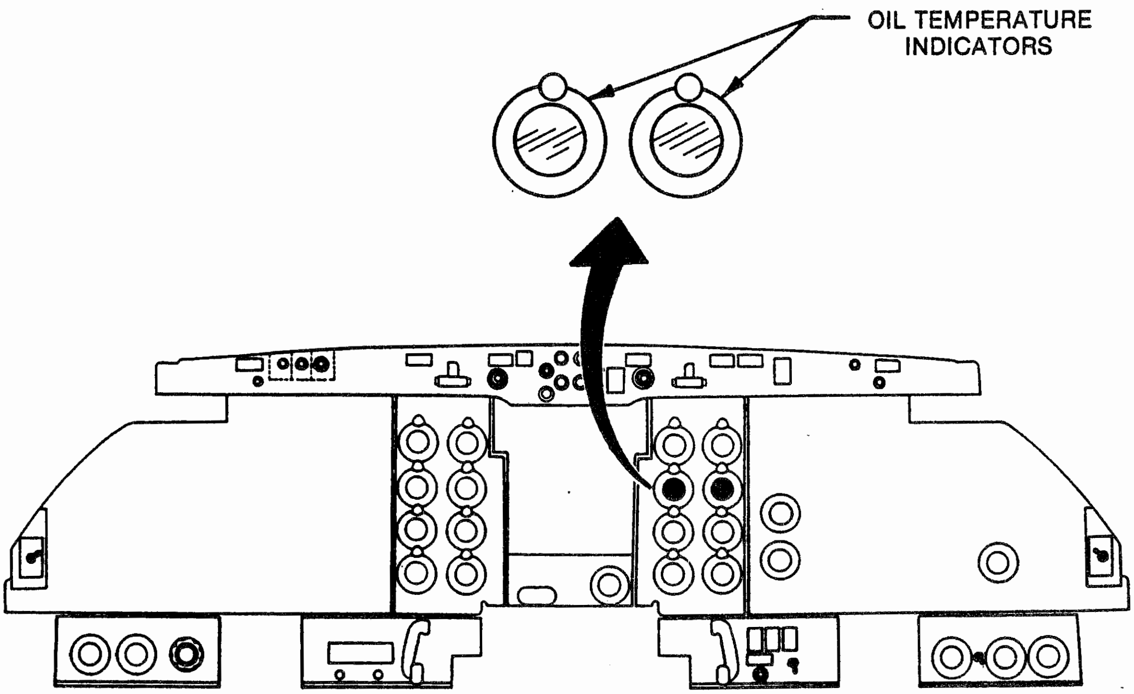
(2) +120°C - red line

(3) −15°C to −30°C - yellow arc

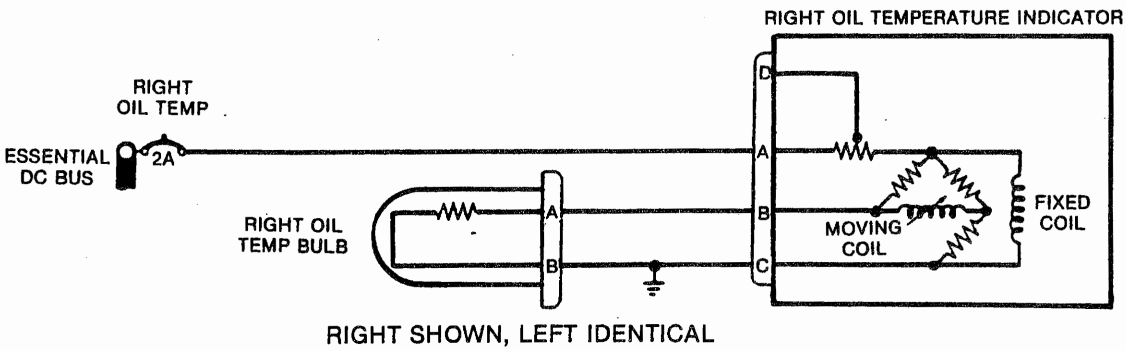
(4) −15°C to +120°C - green arc

The following list indicates temperature/resistance characteristics for platinum resistance thermometer bulbs.

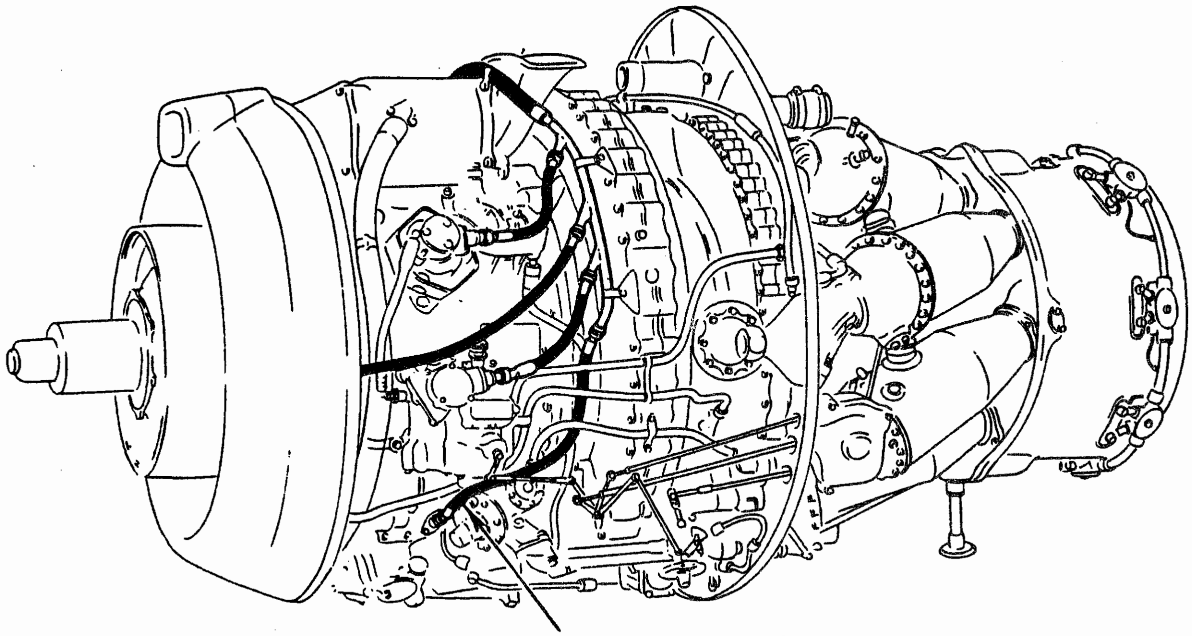
TEMPERATURE (°C)	RESISTANCE (ohms)	TEMPERATURE (°C)	RESISTANCE (ohms)
−70	93.58	50	155.54
−60	98.83	60	160.60
−50	104.07	70	165.65
−40	109.29	80	170.68
−30	114.49	90	175.70
−20	119.68	100	180.70
−10	124.85	110	185.69
0	130.00	120	190.66
10	135.14	130	195.61
20	140.26	140	200.56
30	145.37	150	205.48
40	150.46		



Oil Temperature Indicator — Location in Cockpit
Figure 1.



Oil Temperature Indicator Circuit — Schematic
Figure 2.



Oil Temperature Indicator Circuit — Schematic
Figure 3.

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ENGINE OIL TEMPERATURE INDICATING SYSTEM — MAINTENANCE PRACTICES

1. **Engine Oil Temperature Indicator — Removal / Installation**

   CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT.

   A. Removal

      (1) Disconnect electrical connector.

      (2) Remove indicator.

   B. Installation

      (1) Install indicator.

      (2) Connect electrical connector.

      (3) Run engine and check operation of indicator.

2. **Engine Oil Temperature Bulb — Removal / Installation**

   NOTE: Refer to Rolls-Royce Dart Maintenance Manual (M-Da7-G) for Engine Oil Temperature Bulb — Removal / Installation procedures.

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TURBINE GAS TEMPERATURE INDICATING SYSTEM — DESCRIPTION / OPERATION

1. **Description**

A. The turbine gas temperature indicating system consists of two individual systems, one for each engine. There is no interconnection between the two. The turbine gas temperature indicating system measures the temperature of the exhaust gases as they pass through the turbine stages. Each system consists of twelve thermocouples, a turbine gas temperature indicator, an adjustable resistor and thermocouple leads. The twelve thermocouples (6 long reach and 6 short reach) are arranged so as to form the leading edge of the intermediate stage nozzle guide vanes. The long reach and short reach are arranged so that a mean temperature can be sensed. The thermocouple leads are wired in parallel and connected through the resistor to the turbine gas temperature indicator.

B. As shown in Figure 1 the indicators are mounted in the left-center instrument panel below the torque meter instruments. On aircraft having ASC 115, TGT indicator and the tachometer indicator locations are interchanged. The TGT dial range is from 0° to 1000°C.

The indicators are marked as follows:

ENGINE MODEL				
Dart Mark 529-8E * &-8H	Dart Mark 529-8X *	Dart Mark 529-8X (Installed Per ASC 246)	Dart Mark 529-8Y (Installed Per ASC 246)	Dart Mark 529-8Z (Installed Per ASC 246)
INDICATOR MARKING				
930°C Red Radial Arrow Labeled Start	930°C Red Radial Arrow Labeled Start	930°C Red Radial Arrow Labeled Start	930°C Red Radial Arrow Labeled Start	930°C Red Radial Arrow Labeled Start
860°C Red Radial Line Labeled Wet	860°C Red Radial Line Labeled Wet	860°C Red Radial Line Labeled Wet	860°C Red Radial Line Labeled Wet	860°C Red Radial Line Labeled Wet
825°C Red Radial Dashed Line Labeled Dry	825°C Red Radial Dashed Line Labeled Dry	825°C Red Radial Dashed Line Labeled Dry	825°C Red Radial Dashed Line Labeled Dry	825°C Red Radial Dashed Line Labeled Dry
850°C to 860°C (Yellow ARC)	850°C to 860°C (Yellow ARC)	825°C to 870°C (Yellow ARC)	825°C to 910°C (Yellow ARC)	825°C to 920°C (Yellow ARC)
550°C to 850°C (Green ARC)	550°C to 850°C (Green ARC)	550°C to 825°C (Green ARC)	550°C to 825°C (Green ARC)	550°C to 825°C (Green ARC)
*** The Dart 529-8E is installed on Aircraft 1 - 146. The DART 529-8H is a modified 529-8E engine that is installed on Aircraft 147 - 164, 322 and 323. The 529-8X (RR MOD 1365) engine is installed on Aircraft 165 - 200 excluding 322 and 323, and Aircraft 1 - 164, 322 and 323 having ASC 176. The Dart 529-8X (RR MOD No. 1814), 529-8Y and 529-8Z engines provide a higher maximum continuous power and can be used only on Aircraft having ASC 246.**				

CAUTION: IT IS ACCEPTABLE TO INSTALL DART MK529 ENGINES OF DIFFERENT POWER RATINGS IN THE AIRCRAFT, HOWEVER, BOTH ARE TO BE OPERATED AT THE STANDARD OF LOWEST RATED ENGINE. ASC STATUS MUST BE CONSIDERED, I.E. A/C HAVING ASC 176, 176A AND/OR 246, PRIOR TO INSTALLATION OF A DIFFERENT STANDARD ENGINE MODIFICATION MAY BE REQUIRED TO BRING AIRFRAME AND HIGHER RATED ENGINE TO REQUIREMENTS OF LOWER RATED ENGINE. IT IS RECOMMENDED THAT ENGINE OPERATING LIMITATIONS CARD AND BOTH TGT INDICATORS BE REPLACED WITH CARD AND INDICATORS REQUIRED FOR LOWEST RATED ENGINE. OPERATIONAL RESTRICTIONS AND LIMITATIONS TO BE OBSERVED WILL BE AS PUBLISHED IN GI AFM OR APPROPRIATE GI ASC 246 SUPPLEMENT FOR LOWEST RATED ENGINE INSTALLED.

NOTE: The TGT indicators will be operative in emergency since they do not depend upon any dc power from the aircraft for operation.

Where auto feathering is due to fuel starvation, the engine may be considered serviceable after completion of a satisfactory ground run.

CAUTION: HIGH TURBINE GAS TEMPERATURES MAY BE EXPERIENCED WHEN STARTING IF THE STARTER TERMINAL VOLTAGE IS TOO LOW, OR IF THE PROPELLER IS NOT ON ITS GROUND FINE PITCH STOP.

Type ratings are established for certification purposes and for use in flight, in addition each engine has its individual maximum and minimum TGT limits at takeoff rpm and maximum continuous rpm; these are determined by test bed performance and are entered on the engine data plate. TGT limits at takeoff rpm and preset idling fuel flow together provide the basis for control interconnection and for performance ground checks. For more specific limitations concerning excessive TGT condition. Refer to Rolls-Royce Dart Maintenance Manual (M-Da7-G), Chapter 71.

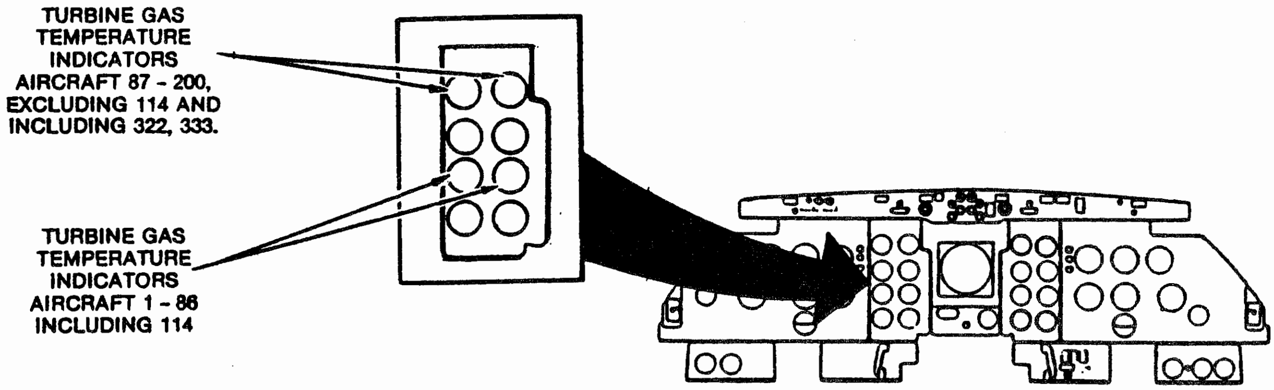
2. **Operation**

The principal of the turbine gas temperature indicating system is the measurement of electrical energy produced by heat energy. Gas temperature are not consistent over a cross section of the main flow through the turbine, as zones or layers of dissimilar temperatures can exist simultaneously. Therefore, it is necessary to sample gas temperature from a number of points well distributed over a cross section of the flow. The Dart engine TGT system incorporates twelve thermocouples which are positioned radially around the nozzle box and protrude into main gas stream at turbine intermediate stage. Metering of the temperatures from twelve points across a turbine stage enables the cockpit indicator to register an accurate mean turbine gas temperature. Each thermocouple consists of a sensitive wire probe encased by a sheath of heat-resisting steel shaped to form part of an airfoil. When positioned in the turbine, each thermocouple locates in a nozzle guide vane, which is recessed to accommodate it. The sheath forms the leading edge of the airfoil shaped guide vane. Two types of thermocouple, long reach and short reach, are used in the assembly. The wire probe of the long reach type senses, through sampling holes in the sheath, the temperature of gases passing over the inner portion of the nozzle guide vane leading edge. The short reach type senses gas flow temperature through sampling holes in the outer portion of the leading edge. The thermocouples are arranged around the nozzle box in pairs; each pair consists of one long and one short reach thermocouple which are connected by flexible armored cable to a junction box. Six junction boxes, interconnected by a aluminum conduit carrying the wiring, are arranged around the nozzle box heat shield and each box serves one pair of thermocouples. As the turbine gas passes over the thermocouples, a small voltage, proportional to the heat of the turbine gas, is generated in the thermocouple. This information is transmitted to the indicator, located on the left center instrument panel in the cockpit, by means of chromel/alumel wires. (See Figure 2)

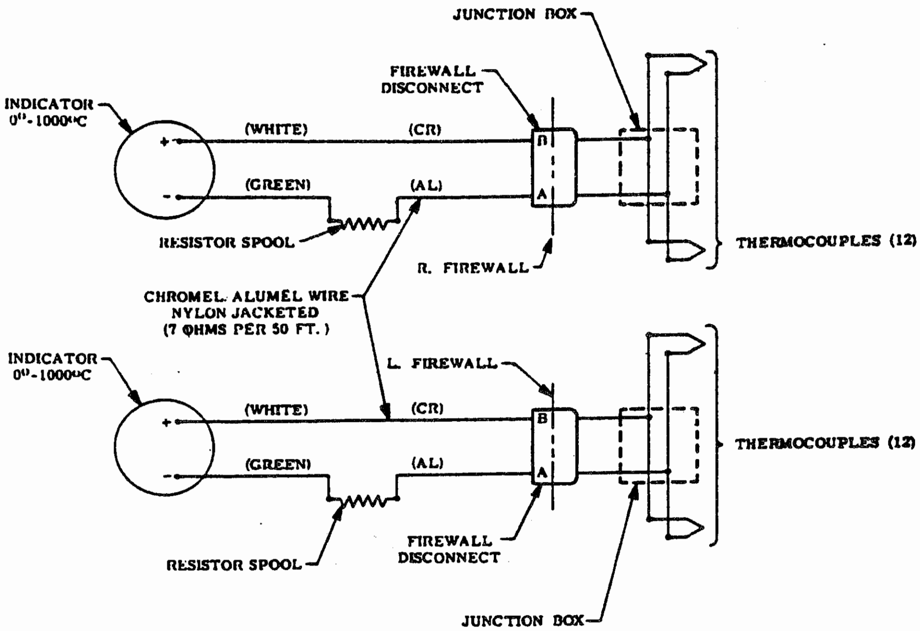
The indicator is an electrically operated, remote indicating instrument, basically a millivoltmeter, which measures the electromotive force produced when the thermocouples become heated. This value is indicated on the indicator face in degree centigrade. The indicators are the 8 ohm type, that is, the loop resistance of the chromel and alumel leads, plus the twelve thermocouples must be adjusted to 8 $\pm$ 0.05 ohms. This calibration of the loop is accomplished by the installation of a standard resistance spool located under FLR No. 7 in series with the alumel lead of each indication system. Since the tolerance is 8 $\pm$ 0.05 ohms, it is essential that an accurate measuring device, such as a resistance bridge, be used to set up the loop resistance.

NOTE: An ohmmeter cannot be used.

Polarity of the leads must be maintained. The alumel (green) lead must terminate at the negative terminal of the indicator, and the chromel (white) lead must terminate at the positive terminal of the indicator. Since the loop resistance greatly affects the accuracy of the system, extra care should be taken to ensure all electrical connections are clean and tight throughout the system, and that the 8 $\pm$ 0.05 ohms tolerance must not be expanded. When calibrating the loop resistance, the indicator must be disconnected, and the thermocouples must be ambient temperature, otherwise the calibration will not be accurate.



TGT Indicator Location in Cockpit
Figure 1.



Turbine Gas Temperature Circuit — Schematic
Figure 2.

TURBINE GAS TEMPERATURE INDICATING SYSTEM — MAINTENANCE PRACTICES

1. **General**

A. TGT Indicator System — Correcting

The maximum rpm must be within limits and the appropriate correction for prevailing atmospheric conditions applied to the data plate TGT limitations before this check is carried out. Because of the lag in TGT indicator readings, it is essential when carrying out the TGT check to run the engine for 1 1/2 minutes to permit the TGT to stabilize, and longer if necessary to raise the oil temperature to that required. Before ground running the engine and comparing the indicated TGT with the data plate limits, these limits require corrections to be subtracted to allow for the variation of prevailing atmospheric conditions from International Standard Atmosphere (ISA). Temperature correction chart is provided in Rolls-Royce Dart Maintenance Manual (M-Da-7) Chapter 71. To obtain accurate results when ground running, the prevailing air temperature should be checked accurately by suspending a mercury thermometer in the shade adjacent to the aircraft. A reading of the prevailing atmospheric pressure can be obtained from the aircraft altimeter by setting it to 1013 millibars or 29.92 in. hg. and reading off the pressure altitude in feet above or below sea level. To ensure that TGT readings recorded during checks are accurate, the prevailing temperature and pressure altitude should be checked immediately before a reading is taken, and adequate time for the engine to stabilize.

CAUTION: BEFORE INVESTIGATING ANY REPORTED TGT FAULTS AT MAXIMUM CONDITIONS, ENSURE THAT THE MAXIMUM RPM IS ACCURATELY SET. IT IS IMPORTANT ALSO TO VERIFY THAT THE TRIMMER SETTING PROCEDURE HAS BEEN CORRECTLY CARRIED OUT IN ACCORDANCE WITH PREVAILING OAT AND PRESSURE ALTITUDE READINGS. TORQUE PRESSURE IS DESCRIBED AS "SATISFACTORY" IN THE SCHEDULE WHEN IT FALLS WITHIN THE ENGINE POWER CHECK LIMITS, GIVEN IN THE TORQUE PRESSURE SECTION.

B. TGT System — Servicing

When a thermocouple, thermocouple harness or part of thermocouple harness has been disconnected or replaced, circuit insulation and resistance tests must be made before ground running engine to check pyrometry system. Run engine and check that indicated TGT is consistent with engine conditions throughout operating range.

If defect is suspected in pyrometry system, test turbine gas temperature indicator and associated aircraft equipment before checking serviceability of complete thermocouple harness assembly.

NOTE: Fitment of Mod. 1567 (Dart service bulletin (S.B.) Da77-8) trim resistor makes only minimal difference to the thermocouple harness resistance, therefore the specified values and tolerance remain unaltered.

If tests are being made to investigate low turbine gas temperature on engines fitted with pre-Mod. 1154 (S.B. Da77-4) thermocouples, remove each thermocouple from nozzle box and check that thermocouple probe is still visible through gas sampling holes. If probe is partly shrouded by detached insulating sleeve, or if either of the sampling holes are blocked, replace the thermocouple.

2. **TGT System — Loop Resistance Check**

NOTE: This is a check of the entire system (except indicator) for loop resistance. It must include permanently hooked up airframe and engine components.

TGT System Loop Resistance may be accomplished utilizing the resistance bridge (paragraph A) or the Barfield TT-1000A with mod C (paragraph B). An alternate procedure is also located in Rolls-Royce Dart Maintenance Manual (M-Da7-G) Chapter 77.

A. Resistance check utilizing Resistance Bridge.

(1) Remove indicator. See Engine TGT Indicator — Removal / Installation, this Section.

CAUTION: FAILURE TO DO THIS WILL RESULT IN DAMAGING THE INDICATOR SINCE THE RESISTANCE BRIDGE UTILIZES A VOLTAGE FOR MEASUREMENT PURPOSES.

(2) Connect accurate resistance bridge to system at indicator wire terminals ensuring a good connection.

(3) Measure resistance. It should be 8 $\pm$ 0.05 ohms.

(4) If resistance is not 8 $\pm$ 0.05 ohms, proceed as follows;

(a) Check security of all connections, paying particular attention to any contamination or moisture in firewall disconnect plug.

(b) If all connections are satisfactory, adjust resistance spool for that system (under FLR No.7) adding or subtracting resistance on active spool until acceptable limits are attached.

(c) Make good solder connection at spool lug.

(d) Coat soldered connection with heavy insulating varnish. See paragraph 5 for Corrosion Prevention procedures.

(5) Install indicator. See Engine TGT Indicator — Removal / Installation, this Section.

B. Resistance check utilizing Barfield TT-1000A TEST SET.

WARNING: WHEN UTILIZING BARFIELD TT-1000A, THE TEST SET MUST HAVE MOD "C"

NOTE: Ensure to comply with all precautions and self test prior to and during this test that are published in the Barfield TT-1000A test set operating instruction manual.

(1) Remove indicator. See Engine TGT Indicator — Removal / Installation, this Section.

(2) Connect Barfield TT-1000A to indicator wire terminals.

(3) Rotate "FUNCTION" to "RESISTANCE MEASURE".

(4) Rotate "RESISTANCE RANGE" to 20 ohms.

(5) Place "EXTENDED RANGE" to 20 ohm position.

(6) Place "ON/OFF" to "ON" and depress "BLACK pushbutton

(7) Resistance should be 8. $\pm$ 0.05 ohms. Interchange red and black test clip connections and display should repeat when "BLACK" pushbutton is depressed. If reading does not repeat, engine thermocouples may be hot. (Refer to hot engine testing).

(8) If resistance is not. $\pm$ 0.05 ohms, proceed as follows:

(a) Check security of all connections, paying particular attention to any contamination or moisture in firewall disconnect plug, or loss of pin tension.

(b) If all connections are satisfactory, adjust resistance spool for that system, (under FLR No. 7), adding or subtracting resistance on active spool, until acceptable limits are attained.

(c) Make good solder connection at spool lug.

(d) Coat soldered connection with heavy insulating varnish. See paragraph 5 for Corrosion Prevention procedures.

(9) Remove test set when test is completed.

(10) Install indicator. See TGT Indicator — Removal / Installation this Section.

(11) Inspect area for foreign objects.

3. Engine Thermocouple / TGT Harness — Resistance Check

WARNING: WHEN UTILIZING BARFIELD TT-1000A, TEST SET MUST HAVE MOD "C"

NOTE: Refer to Rolls-Royce Dart Maintenance Manual (M-Da7-G) Chapter 77 for alternate procedure.

Ensure to comply with all precautions and self test prior to and during this test that are published in the Barfield TT-1000A test set operating instruction manual.

A. Resistance Check - Firewall to Junction Box Short Lead

(1) Open lower engine cowling.

NOTE: Prior to starting, ensure all engine TGT components are installed and connected, junction boxes are intact with covers, short lead from bottom thermocouple junction box is connected to TGT junction box terminals.

(2) Disconnect short lead from fire wall connector.

(3) Remove bottom TGT junction box cover.

(4) Disconnect one end of short lead at terminal in TGT junction box.

(5) Short firewall short lead plug pins together with a short jumper and connect Barfield TT-1000A, as shown in Figure 201. Ensure clips make good electrical connections.

(6) Rotate "FUNCTION" control to "RESISTANCE MEASURE".

(7) Rotate "RESISTANCE RANGE" control to 20 ohm.

(8) Set "EXPANDED RANGE" control to M. ohms position.

(9) Place "ON/OFF" switch to "ON". Depress "BLACK" pushbutton and measure resistance. Value must be 0.272 ± 0.015 ohms.

(10) Turn off test set.

(11) Remove shorting jumper and test set.

(12) Perform paragraph B, Resistance Check - Engine Thermocouple and TGT Harness procedure.

B. Resistance Check - Engine Thermocouple/TGT Harness

NOTE: This procedure is normally accomplished in conjunction with TGT system functional test, also required when engine is replaced, or any TGT component is disturbed.

NOTE: On engines having MOD 1567 (S.B. Da77-8), trimmer resistor makes only minimal difference to thermocouple harness resistance, therefore the specified values and tolerances remain unaltered.

(1) If trim resistor is installed, remove trimmer resistor from junction box and clip test leads across resistor.

(2) Rotate "FUNCTION" control to "RESISTANCE MEASURE".

(3) Rotate "RESISTANCE RANGE" control to 20 ohm.

(4) Set "EXPANDED RANGE" control to 20 ohms position.

(5) Place "ON/OFF" switch to "ON" and depress "BLACK" pushbutton and measure resistance. Value must be 0.272 ± 0.015 ohms.

NOTE: 10 ° C, trimmer resistor shall measure 4.678 ± 0.094 ohms, at 20° C, trimmer resistor shall measure 2.339 ± 0.047 ohms.

(6) If trim resistance check is satisfactory, refit resistor and continue TGT Harness circuit resistance check.

(7) Connect test set as shown in Figure 202

(8) Rotate "FUNCTION" control to "RESISTANCE MEASURE".

(9) Rotate "RESISTANCE RANGE" control to 20 ohms.

(10) Place "EXTENDED RANGE" control to 20 ohm position.

(11) Place "ON/OFF" switch to "ON" and depress "BLACK pushbutton. Resistance shall be 0.065 ± 0.005 ohms. Interchange RED and BLACK test lead connections and display should repeat when "BLACK" pushbutton is depressed. If reading does not repeat, engine thermocouples may be hot.

NOTE: If results are not within tolerance, consult Rolls-Royce Dart Maintenance Manual (M-Da7-G) Chapter 77 for further troubleshooting procedures.

(12) Disconnect test equipment and reconnect short lead to terminal in TGT junction box. See Figure 203 for terminal hardware installation sequence. Remove jumper from short lead plug and reconnect at firewall and safety.

CAUTION: SNUG UP NUT, BUT DO NOT OVERTORQUE OR STUDS COULD BE BROKEN.

(13) Install cover on junction box.

(14) Inspect area for foreign objects, damage and leaks.

(15) Close engine cowling.

4. **Engine Harness — Insulation Resistance Check**

CAUTION: DISCONNECT FIREWALL PLUG BEFORE MAKING THIS CHECK.

NOTE: Engine harness insulation check may be accomplished utilizing a Megger (Paragraph A) or Barfield TT-1000A Test Set having MOD "C". (Paragraph B) An alternate procedure is also located in Rolls-Royce Dart Maintenance Manual (M-Da7-G) Chapter 77.

A. Engine Harness Insulation check using megger.

(1) Open lower engine cowl.

(2) Disconnect short lead firewall connector.

(3) Connect test set to either pin of firewall plug on short lead as shown in Figure 204. Clip other test lead to a good aircraft ground.

(4) Using 250 volt Megger, check resistance between either pin of short harness lead and ground. Minimum resistance is 10,000 ohms. See Figure 204

(5) Disconnect test set , reconnect short lead plug to firewall receptacle and safety.

(6) Inspect area for foreign objects, damage and leaks

(7) Close engine cowling.

B. Insulation check utilizing Barfield TT-1000A Test Set having Mod"C".

NOTE: Comply with all precautions and self test prior to and during this test that are published in the Barfield TT-1000A test set operating instruction manual.

Prior to starting, ensure all engine TGT components are installed and connected, junction boxes are intact with covers, short lead from bottom thermocouple junction box is connected to TGT junction box terminals.

(1) Open lower engine cowl.

(2) Disconnect short lead firewall connector.

(3) Connect test set to either pin of firewall plug on short lead as shown in Figure 204. Clip other test lead to a good aircraft ground.

(4) Rotate "RESISTANCE RANGE" control to 2M. ohms.

(5) Place "ON/OFF" switch to "ON", depress "BLACK" pushbutton. The minimum resistance is 10,000 ohms.

(6) Disconnect test set , reconnect short lead plug to firewall receptacle and safety.

(7) Inspect area for foreign objects, damage and leaks

(8) Close engine cowling.

5. **TGT System — Corrosion Prevention**

Purpose of this information is to inform maintenance personnel as to correct procedures to seal each TGT connection to prevent a resistance change, which would affect indication.

A. As system is connected, clean mating surfaces of each bolt connection until they are free of burrs and corrosion. A smooth, shiny connection is desirable. It is not necessary to remove plating to obtain it.

B. After a connection is made, cover with Glyptol varnish or other fairly thick insulating medium. Any thick, quick-drying paint is suitable. Spray lacquer is satisfactory if care is taken to prevent lacquer from entering joint.

C. After any subsequent disturbance of a connection, repeat Steps A. and B. above.

CAUTION: CONNECTIONS AT THE INDICATOR SHOULD NOT BE TREATED IN THIS MANNER, BUT SHOULD REMAIN BARE.

6. Engine TGT Indicator — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Remove TGT indicator from instrument panel, by loosing clamp screws around indicator.

(3) Disconnect leads from rear of indicator.

B. Installation

(1) Connect white chromel lead to (+) terminal on rear of indicator.

(2) Connect green alumel lead to (−) terminal on rear of indicator.

CAUTION: A GOOD ELECTRICAL CONNECTION IS CRITICAL IN THIS SYSTEM.

(3) Install TGT indicator in instrument panel and tighten clamp screws.

(4) Perform Engine TGT Indication System — Functional Test, see this Section.

7. Engine TGT Indication System — Functional Test

CAUTION: NEED FOR MAXIMUM ACCURACY OF THIS INDICATOR IN CRITICAL ENGINE TEMPERATURE RANGE CANNOT BE EMPHASIZED TOO STRONGLY.

NOTE: Engine TGT System — Functional Test can be accomplished with Leeds and Northrup equipment (paragraph A). The Barfield TT-1000A Test Set having Mod "C" (paragraph B) or Biddle Test Set 72035-1 (paragraph C). An alternate procedure is also provided in Rolls-Royce Dart Maintenance Manual (M-Da7-G) Chapter 77.

A. Functional Test Utilizing Test Set 159AV1001TEQ-5 and 8657-C

(1) Position a Fahrenheit thermometer in cockpit near the TGT indicators. Read and record indicated temperature.

(2) Opening lower engine cowl.

(3) Disconnect TGT short lead plug at firewall. (plug is at approximately 6 o'clock position).

(4) Connect test equipment as shown in Figure 205. Ensure that Leeds and Northrup box is level.

(5) Adjust needle in window A to O position using knob B. (See Figure 205)

(6) Apply temperature reading from Step (1) above. to Table 1 below to determine value to set static correction. If value is below 1 mV, set Peg C in top hole of scale selector on upper left corner of Leeds and Northrup box. If value is 1 - 5 mV, set peg C in lower hole.

°F	0	1	2	3	4	5	6	7	8	9	°F
30			0.00	0.02	0.04	0.07	0.09	0.11	0.13	0.15	30
40	0.18	0.20	0.22	0.24	0.26	0.29	0.31	0.33	0.35	0.37	40
50	0.40	0.42	0.44	0.46	0.48	0.51	0.53	0.55	0.57	0.60	50
60	0.62	0.64	0.66	0.68	0.71	0.73	0.75	0.77	0.80	0.82	60
70	0.84	0.86	0.88	0.91	0.93	0.95	0.97	1.00	1.02	1.04	70
80	1.06	1.09	1.11	1.13	1.15	1.18	1.20	1.22	1.24	1.27	80
90	1.29	1.31	1.33	1.36	1.38	1.40	1.43	1.45	1.47	1.49	90
100	1.52	1.54	1.56	1.58	1.61	1.63	1.65	1.68	1.70	1.72	100

TGT CORRECTION (STATIC CORRECTION)
TABLE 1

(7) Set reading obtained in Step (6) above with dial D. If value is below 1 mV use inner scale, if value is 1 - 5 mV, use outer scale.

(8) Select switch E to the H position since all readings to be checked are above 16 mV.

(9) While holding switch F in the S.C. position, use adjustment knob supplied in the test set (Leeds and Northrup) to null needle on meter A. The adjustment knob fits test box on left side as viewed. When no deviation from O is noted when Switch F is selected to either TC or SC position, remove adjustment knob. Leave Switch F in the TC position.

CAUTION: FOR HOWELL DIGITAL TGT INDICATORS, LEAVE POWER SWITCH L IN OFF POSITION.

(10) Turn power switch L to ON position for production installed indicators.

(11) Set reading for 760°C obtained from Table 2 in window G using Knob H. Use lower H scale for all readings since they are above 16 mV.

(12) Depress TC Button J (twisting button clockwise locks button in down position).

(13) Rotate knob on variable Pot K on test box until needle is nulled in window A.

NOTE: Panel may have to be tapped lightly to obtain correct readings. Adjustments is made only at the 760°C reading.

(14) Read cockpit indication, it should read 760° ± 5°. If not, production type indicators can be adjusted using adjustment screw at rear of indicator. All readings must be checked to be within tolerance after any adjustment has been made.

(15) Repeat Steps (11) through (14) above using values given in Table 2.

(16) Turn power switch OFF and remove test equipment.

NOTE: If functional test is satisfactory and additional TGT requirements are due, accomplish Step (17) below before proceeding with Steps (18) through (22).

(17) Accomplish Engine Harness/Thermocouple Resistance and Insulation checks. (see procedure in this section)

(18) Connect TGT short lead at firewall plug and lockwire.

(19) Inspect area for foreign objects, damage and leaks.

(20) Close lower engine cowl and secure.

(21) Repeat procedure for opposite engine if both systems are to be checked at this time.

EMF MILLI-VOLTS	TEMPERATURE °C	TOLERANCE °C
20.65	500	±20
24.91	600	±15
29.14	700	± 8
31.65	760	± 5
33.30	800	± 8
37.36	900	±15
41.31	1000	±20

TABLE 2

B. Functional Test — Utilizing Barfield TT-1000A Test Set

WARNING: WHEN UTILIZING BARFIELD TT-1000A, TEST SET MUST HAVE MOD "C"

CAUTION: ENSURE ALL RESISTANCE CHECKS ARE COMPLETED PRIOR TO THIS TASK. THIS PROCEDURE VALIDATES THE INDICATORS DURING ENGINE OPERATION.

ENSURE TO COMPLY WITH ALL PRECAUTIONS AND SELF TEST PRIOR TO AND DURING THIS TEST THAT ARE PUBLISHED IN THE BARFIELD TT-1000A TEST SET OPERATING INSTRUCTION MANUAL.

NOTE: Ensure test leads used in this test are chromel-alumel wire. Total resistance of test leads (one end shorted together) should be no more than 75 ohms. All terminals and connections must be clean and secure.

(1) Open lower cowl.

(2) Remove cover from thermocouple junction box.

(3) Connect chromel-alumel test leads to test receptacle or junction box terminals as follows, see Figure 206 GREEN test lead to BLUE engine harness wire.
WHITE test lead to RED engine harness wire.

(4) Route test lead outside engine cowl and secure with tape along engine nacelle, wing leading edge and over fuselage to kidney inspection plate on side of fuselage, under cockpit. Remove kidney plate and route test lead thru boot around base of control column and into cockpit.

(5) On test set rotate "FUNCTION" control to "TEMP MEASURE".

(6) Connect test lead clips to thermocouple test lead observing polarity. Alumel is negative (−) and connects to black clip, Chromel is positive (+) and connects to red clip. Ensure clips make a good electrical connection.

(7) Turn "ON/OFF" switch to "ON". Display will indicate thermocouple temperature directly in C°.

NOTE: If the word "OPEN" is displayed, there is an open circuit in the test lead wire.

(8) Close and secure engine cowl and perform engine run.

(9) Check TGT indicator against test set reading between 500 and 800° C. Indicator should be within tolerance provided in Table No. 3.

NOTE: If indicator is not within tolerance, perform indicator adjustment procedure in step (10). This will require indicator to be removed This will require indicator to be removed.

(10) TGT indicator adjustment utilizing Barfield TT-1000A.

(11) Shut down engine after engine run is completed.

(12) Turn test set off and remove test lead from aircraft fuselage.

(13) Secure boot around base of control column.

(14) Install kidney plate on side of fuselage.

(15) Disconnect test lead in engine junction box and remove from aircraft.

CAUTION: SNUG UP NUTS, BUT DO NOT OVERTORQUE OR STUDS COULD BE BROKEN.

(16) Install cover on junction box

(17) Inspect area for foreign objects, damage and leaks.

(18) Close engine cowl.

NOTE: This procedure only utilized when engine TGT indicator failed functional test procedure. See this section.

WARNING: WHEN UTILIZING BARFIELD TT-1000A, THE TEST SET MUST HAVE MOD "C"

NOTE: ensure to comply with all précautions and self test prior to and during this test that are published in the Barfield TT-1000A Test Set Operating Instruction Manual.

(19) Shut down engine.

(20) Remove electrical power.

(21) Remove TGT indicator, Refer to procedure this section

(22) Rotate "FUNCTION" control to "RESISTANCE MEASURE".

(23) Rotate "RESISTANCE RANGE" control to 20.00 ohms

(24) Place "ON/OFF" switch to "ON".

(25) Short test lead clips together and depress "RED" pushbutton while adjusting "SYSTEM RES" to 8.00 ohms.

(26) Rotate "FUNCTION" control to "INDICATION TEST".

(27) Connect test lead to indicator terminals, observing polarity.
Alumel is negative and connects to "BLACK" clip.
Chromel is positive and connects to "RED" clip.

NOTE: Ensure clips make a good electrical connection

(28) Set "TEMP ADJ" control for desired test temperature as displayed on digital indicator. Refer to Table 2 for tolerances at listed test points, indicator under test may have to be tapped lightly to obtain correct readings. If out of tolerance, adjust set screw on back of indicator. This adjustment should be made at 760° Test point, since it has smallest tolerance. Re-check indicator at all test points after making any adjustments.

(29) Turn off test set and remove test equipment, see Figure 203 for terminal hardware installation sequence.

(30) Install TGT indicator. Refer to procedure this section.

(31) Repeat functional test procedure.

C. TGT Indication System Functional Test — Using Biddle 72035-1 Test Set

NOTE: Ensure to comply with all precautions and self test prior to and during this test that are published in the Biddle 72035-1 Test Set Operating Instruction Manual.

(1) Initial Set Up

(a) Disconnect AC charger plug from Biddle test set.

(b) Move "TEMPERATURE RANGE" selector to °C.

(c) Move "MODE" selector to "CHECK"

(d) Select each range on "RANGE" selector and check indicator readings to manufacturer specifications as printed on test set lid.

(e) Return "MODE" selector to "OFF".

(2) Functional Test

(a) Gain access to firewall and disconnect TGT short lead plug from firewall.

(b) Connect Biddle test set to aircraft as follows (using supplied alumel/chromel (AL/CR) wire), see Figure 207.

(1) Connect RED (AL) lead to negative post on Biddle test set.

(2) Connect YELLOW (CR) lead to positive post on Biddle test set.

(3) Connect RED (AL) lead to pin "A" of firewall disconnect receptacle.

(4) Connect YELLOW (CR) lead to pin "B" of firewall disconnect. receptacle.

(c) Select the following on Biddle test set.

(1) "TEMPERATURE RANGE" selector to °C.

(2) "RANGE" selector to "K".

(3) "MODE" selector to "OUTPUT".

(d) Using output adjust control knobs, adjust output reading of Biddle test set to read 760°C.

NOTE: Indicator panel may have to be tapped lightly to obtain correct readings. Adjustments made only at 760°C reading.

(e) Read cockpit indication, it should read 760° ± 5°C. Note that production type indicators can be adjusted using screw at rear of indicator. All readings must be checked to be within tolerance after adjustments have been made.

(f) Repeat steps (d) and (e) using temperature values and tolerances in Table 3.

NOTE: EMF VALUES ARE NOT VALID WITH THIS PROCEDURE.

(g) Return "MODE" Selector to "OFF" and remove test equipment.

NOTE: If functional test is satisfactory and additional TGT requirements are due, accomplish Step (h) below before proceeding with Steps (i) thru(l) below.

(h) Accomplish Engine Harness/Thermocouple Resistance and Insulation Check. See procedure this section.

(i) Connect TGT short lead plug at firewall and safety.

(j) Inspect area for foreign objects, damage and leaks.

(k) Close lower engine cowl and secure.

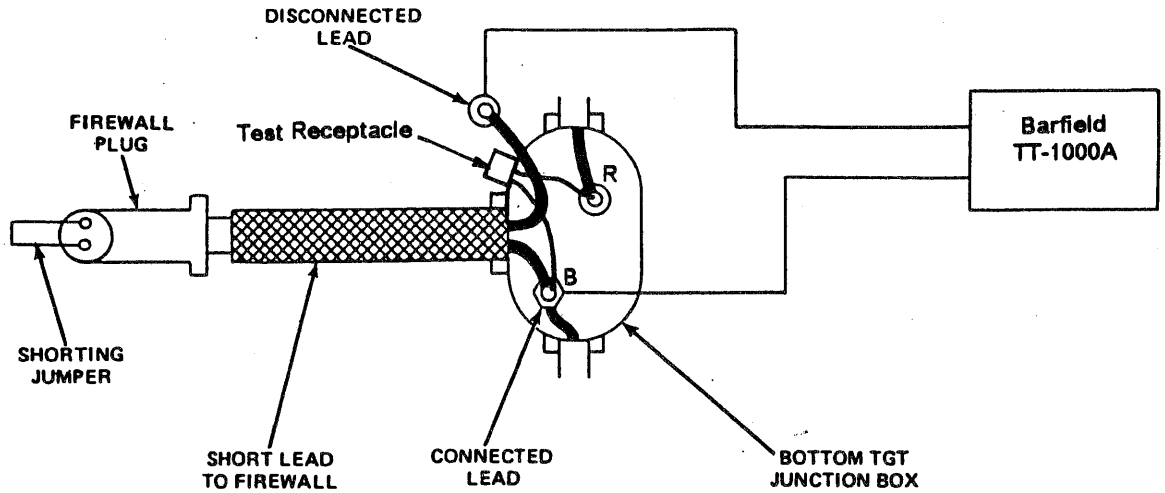
(l) Repeat procedure for opposite engine if both systems are to be check at this time.

TEMPERATURE °C	TOLERANCE °C
500	±20
600	±15
700	± 8
760	± 5
800	± 8
900	±15
1000	±20

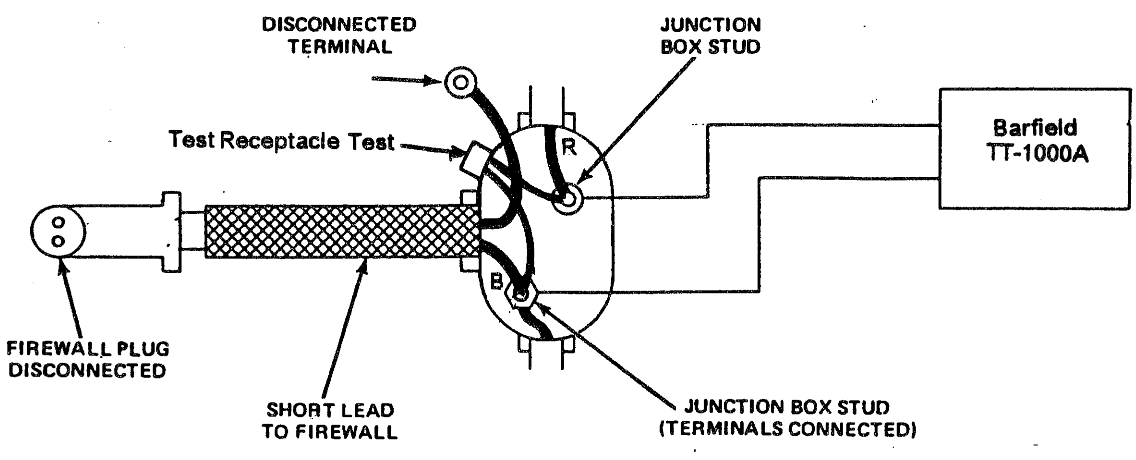
TABLE 3

8. **Thermocouples — Removal / Installation**

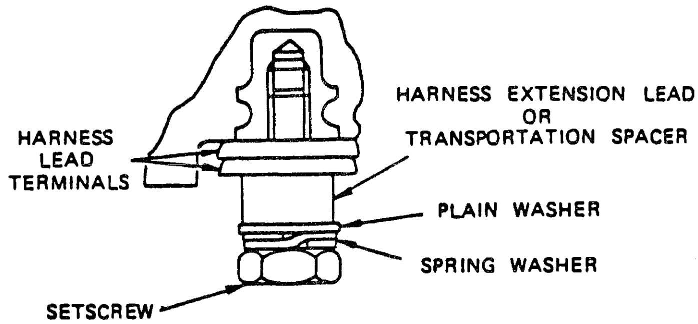
   A. For removal or installation of thermocouples, or their wiring harness on the engine, see Rolls Royce Maintenance Manual (M-Da7-G), Chapter 77.



Engine Junction box/Short Lead Resistance Check — Hook-up
Figure 201.



Engine Thermocouple/TGT Harness Resistance Check — Hook-up
Figure 202.



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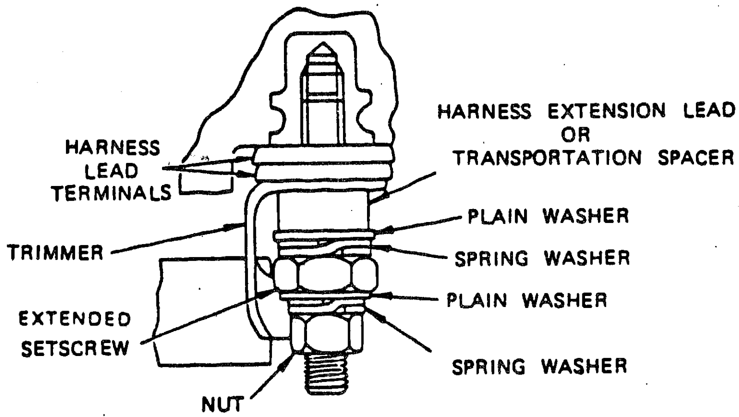
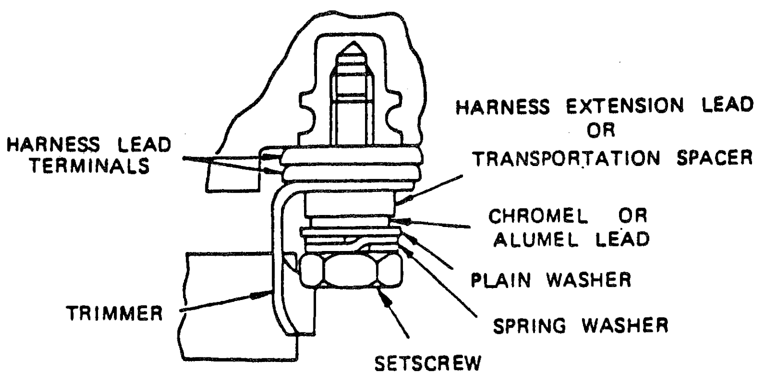
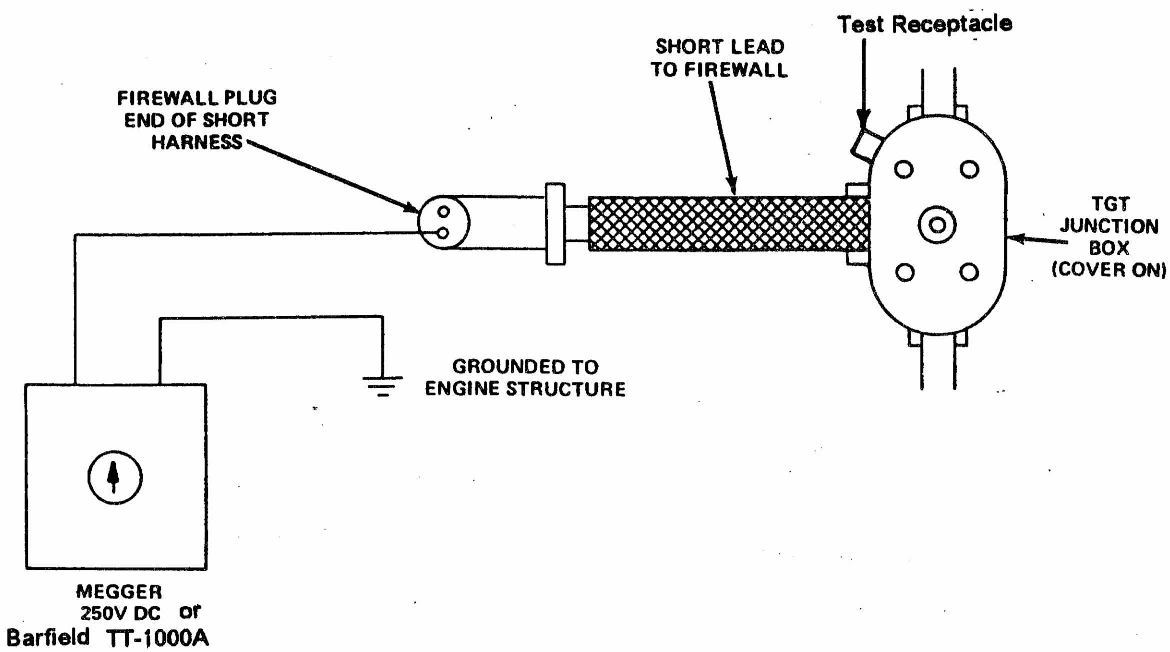


Mod.1567, pre-Mod.1792

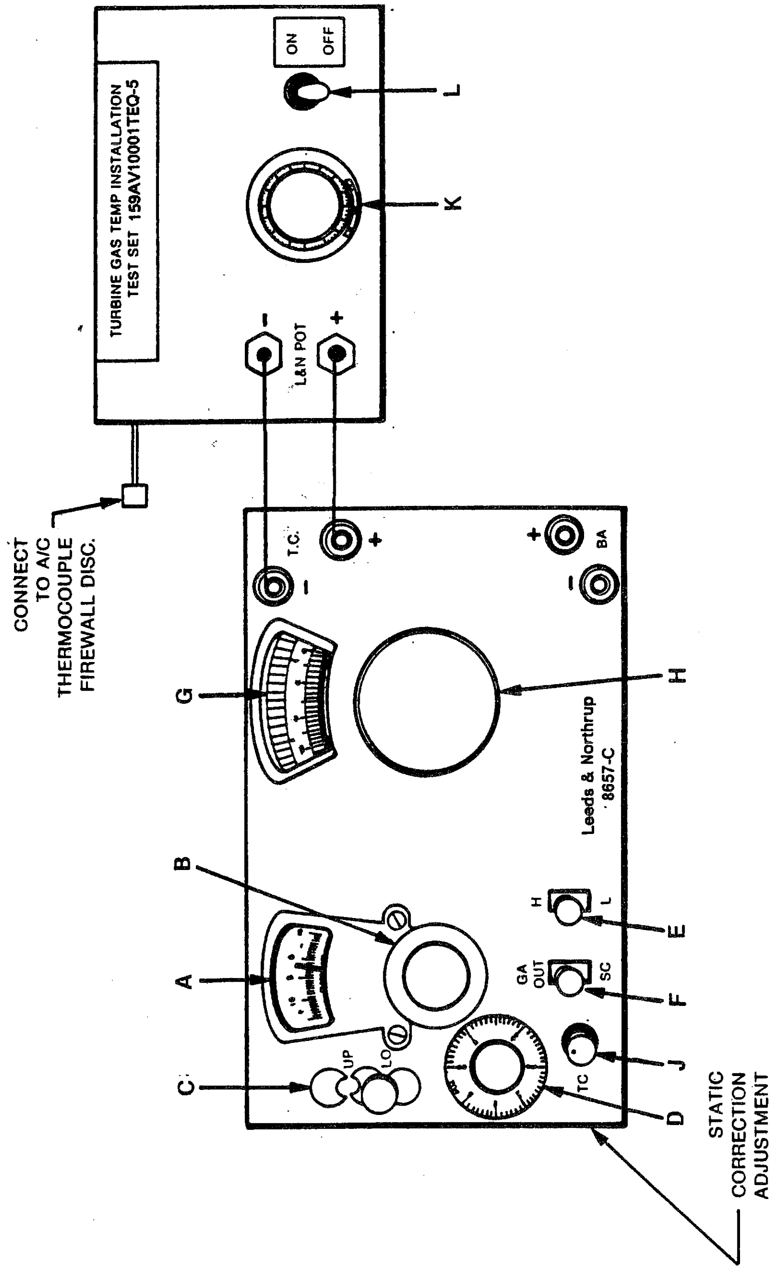


Mod.1792

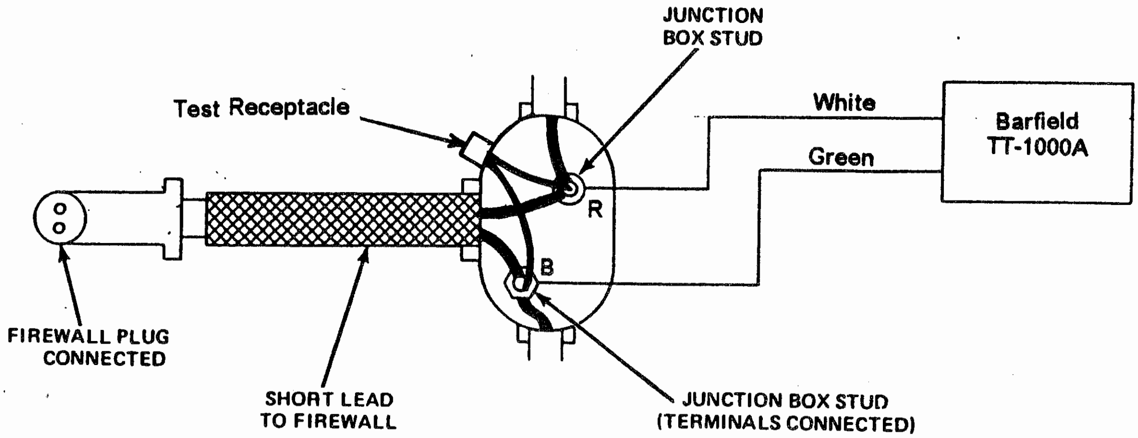
Junction Box Terminal Hardware stack-up
Figure 203.



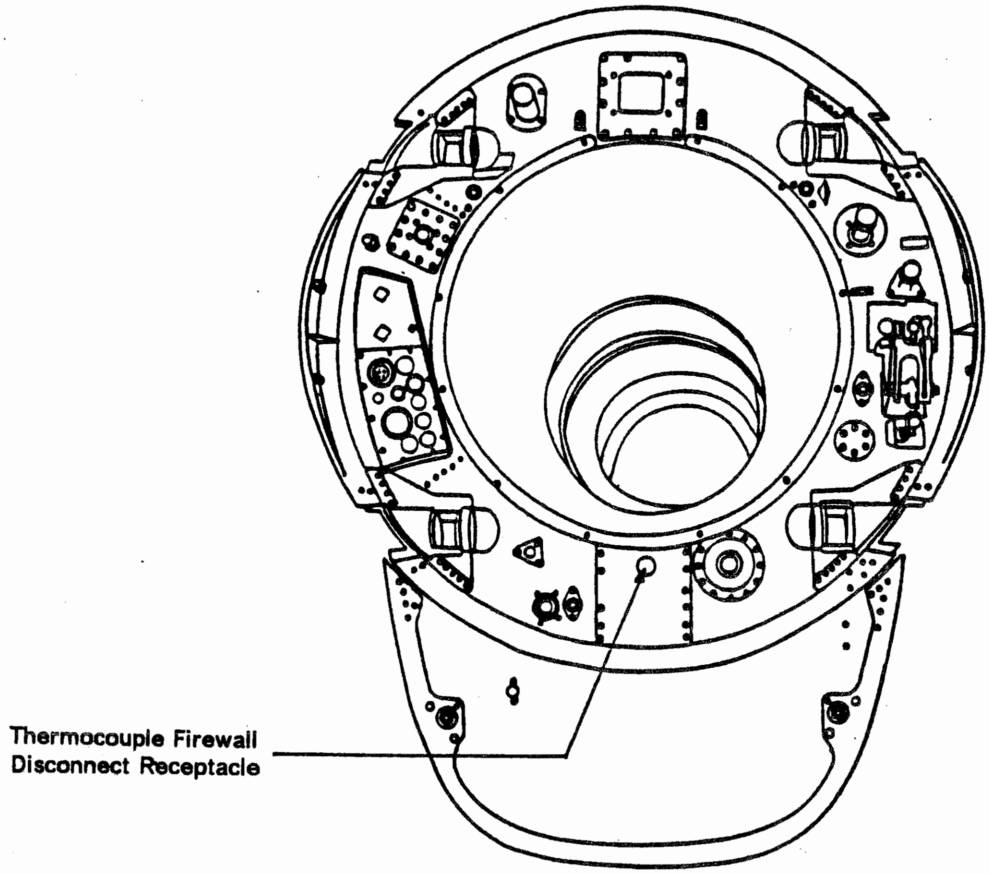
Engine Harness Insulation Check — Hook-up
Figure 204.



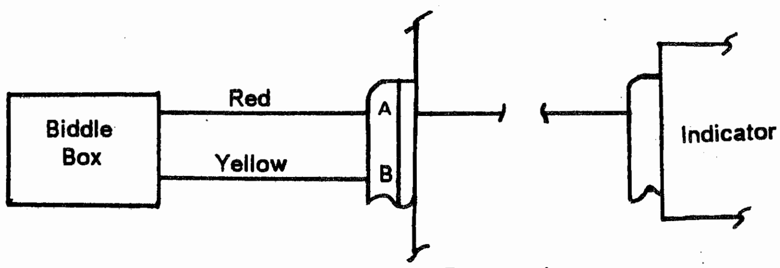
TGT Indicator Functional Test — Hook-up
Figure 205.



TGT Indicator Functional Test — Hook-up
Figure 206.



Engine Nacelle Firewall



Firewall Disconnect Receptacle

TGT Indicator Functional Test — Hook-up
Figure 207.

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TORQUE PRESSURE INDICATING SYSTEM — DESCRIPTION / OPERATION

1. Description

The torque pressure indicators are mounted in the left center instrument panel in the cockpit and have two inch dials. These indicators have single pointers and read 0 to 600 psi. (See Figure 1).

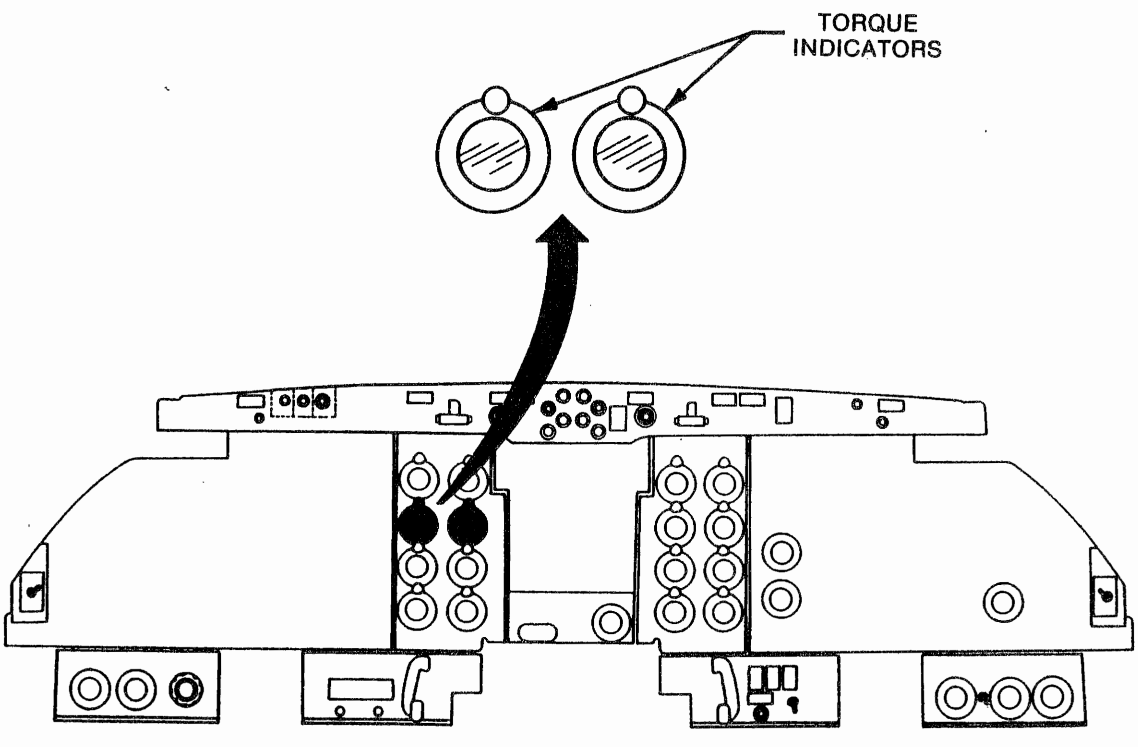
A torque pressure transmitter is mounted on the left side of each engine, enclosed in a special antivibration mount. (See Figure 2). Torque pressure is sensed mechanically and amplified to move brushes contacting the windings of the transmitter. These windings are excited by 28V dc from the essential dc bus and connected by three wires to three coils in the respective indicator. Thus, motions of the brushes in the transmitter vary the field strengths of the indicator coils causing the permanent magnet indicator rotor to move. This movement positions the pointer. (See Figure 3). Torque meter pressure is used to provide a convenient guide to abnormality in engine power. This is achieved by running the engine dry, i.e., without water methanol, at takeoff conditions with the oil inlet temperature at 70°C and comparing the observed torque meter pressure with the corrected power check pressure entered in the engine log book.

NOTE: Since the observed torque meter pressure can be affected to some extent by engine oil inlet temperature, it is important that the oil temperature is within $\pm$ 5°C of the nominal 70°C when taking torque meter readings.

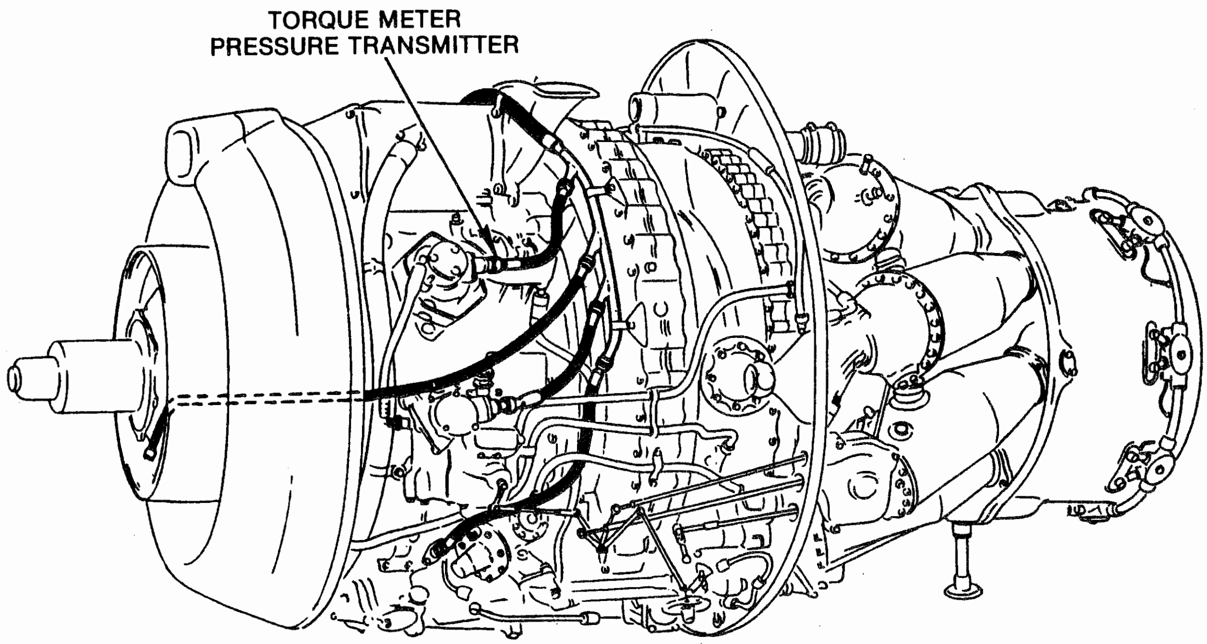
The indicators are marked as follows:

40 psi - red line

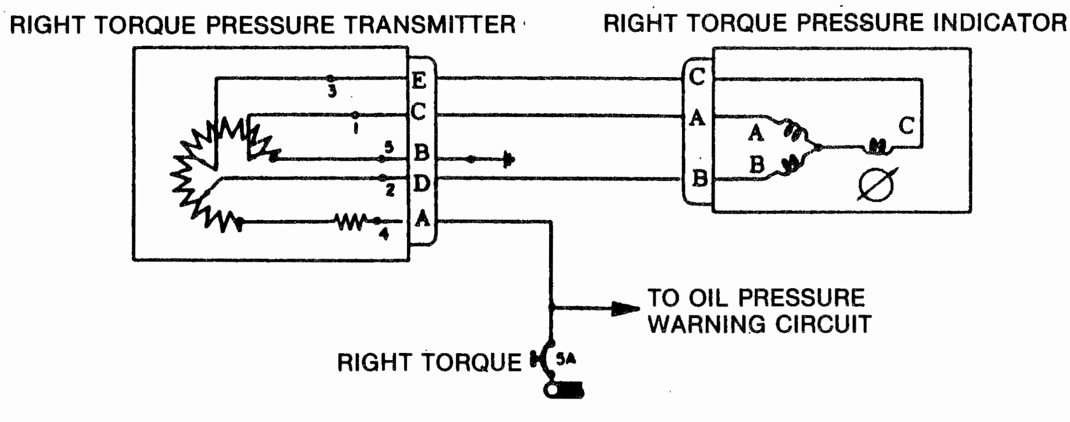
The torque pressure indicators are operative during emergency dc operation.



Torque Indicator Location in Cockpit
Figure 1.



Torque Meter Pressure Transmitter Electrical Connection on Engine
Figure 2.



Essential DC Bus — Right Shown, Left Identical
Figure 3.

TORQUE PRESSURE INDICATING SYSTEM — MAINTENANCE PRACTICES

1. **General**

Torque meter pressure is useful for providing supporting evidence of an engine defect indicated in flight by incorrect TGT or excessive staggering of fuel datum positions. It should not be regarded as a main symptom of a defect unless the discrepancy registered is severe or unless there are irregular readings not due to a faulty transmitter. In reading a torque meter pressure indicator, distinction must be made between a regular fluctuation of up to 30 psi on either side of a mean value, which is acceptable and a indicator reading which flickers constantly downwards, which is frequently a sign of a defective transmitter system. The latter may be confirmed by moving the throttle to a new position where the flicker should be temporarily eliminated. The recommended method of checking the torque pressure transmitter system is the fitting of an accurately calibrated Bourdon tube type of indicator. Where an engine has been feathered for loss of torque pressure, the oil filters must be examined. If satisfactory, the torque pressure transmitter system must be checked for serviceability, the engine ground run and the filters re-examined before the engine is accepted for further service.

2. **Engine — Power Check**

(See Table 201)

A. Correct power check pressure as entered in logbook for ambient conditions as given in chart which follows.

B. Ensure that all necessary electrical load is switched off and blower is dumped.

C. Carry out serviceability ground check at full throttle (15,000 rpm).

D. If TGT is satisfactory, allow sufficient time for torque meter See Rolls-Royce Dart Maintenance Manual Chapter 77 for procedure. reading to stabilize (with oil temperature 65 to 75°C) and record dry torque pressure. Time taken to obtain a stabilized reading may vary with ambient conditions: it should never be less than 1 1/2 minutes and at very low temperature may extend to 4 minutes.

E. Compare observed torque meter pressure with corrected check pressure. If observed torque pressure falls within + 20, to − 50 psi of corrected check pressure, engine may be considered serviceable in respect to power.

F. If observed torque meter pressure falls outside limits, carry out following checks before rejecting engine as suspect.

(1) Perform Engine RPM Indicator — Functional Test, to see that it is correct to within 50 rpm, see this Chapter.

(2) Fit a Bourdon tube type master torque pressure indicator. (See Rolls-Royce Dart Maintenance Manual Chapter 71)

(3) Check thermocouples, pyrometry and TGT indicator, TGT reading must be correct to within $\pm$ 5°C at 760°C.

(4) Perform Engine — Operational Test and repeat power check. (See Chapter 71 or Rolls-Royce Dart Maintenance Manual Chapter 71)

(5) If results are still unsatisfactory, attempt to bring torque pressure within limits by resetting engine controls to give a TGT within −5°C of corrected maximum power temperature. (See Rolls-Royce Dart Maintenance Manual Chapter 77 or this Section)

3. **Engine Torque Pressure Indicator — Removal / Installation**

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT.

A. Removal

(1) Disconnect electrical connector.

(2) Remove indicator.

B. Installation

(1) Install indicator.

(2) Connect electrical connector.

(3) Run engine and check operation of indicator.

4. **Engine Torque Pressure Transmitter — Removal / Installation**

NOTE: Refer to Rolls-Royce Dart Maintenance Manual Chapter 77 for procedure.

°C	PRESSURE ALTITUDE — ft.														
	-1000	-500	-250	SLO	500	1000	2000	3000	4000	5000	6000	7000	8000	9000	10,000
50	-97	-103	-106	-109											
45	- 82	- 89	- 92	- 94	-101	-107	-120								
40	- 66	- 74	- 77	- 79	- 86	- 93	-106	-118	-132	-142					
35	- 49	- 57	- 60	- 83	- 71	- 78	- 91	-105	-119	-129	-142	-154			
30	- 32	- 40	- 43	- 46	- 54	- 62	- 76	- 90	-105	-117	-130	-143	-156	-163	-174
25	- 13	- 22	- 26	- 29	- 37	- 45	- 60	- 75	- 91	-103	-117	-131	-144	-152	-163
20	- 5	- 3	- 7	- 10	- 19	- 28	- 44	- 60	- 76	- 90	-104	-118	-132	-140	-152
15	- 7	0	5	9	- 1	- 10	- 27	- 44	- 60	- 5	- 90	-105	-119	-129	-141
10	- 9	- 3	1	6	3	0	- 9	- 28	- 44	- 61	- 75	- 92	-106	-118	-130
5	-12	- 6	-2	2	- 1	- 3	-8	- 10	- 28	- 46	- 61	- 78	- 93	-106	-120
0	-16	- 10	- 6	- 2	- 5	- 7	- 12	- 19	- 24	- 31	- 45	- 63	- 79	- 94	-109
- 5	- 21	- 15	- 11	- 7	- 10	- 12	- 16	- 23	- 28	- 34	- 40	- 48	- 65	- 83	- 96
- 10	- 26	- 21	- 16	- 12	- 15	- 17	- 21	- 28	- 33	- 39	- 44	- 50	- 55	- 71	- 87
- 15	- 33	- 27	- 22	- 18	- 20	- 22	- 26	- 33	- 37	- 43	- 48	- 54	- 59	- 67	- 76
- 20	- 39	- 34	- 29	- 24	- 26	- 28	- 31	- 38	- 42	- 48	- 53	- 58	- 64	- 71	- 77
- 25	- 46	- 42	- 36	- 31	- 32	- 34	- 37	- 44	- 48	- 53	- 58	- 63	- 68	- 75	- 81
- 30	- 53	- 50	- 44	- 38	- 40	- 41	- 44	- 50	- 54	- 58	- 62	- 68	- 73	- 79	- 85
- 35	- 61	- 58	- 51	- 45	- 47	- 48	- 51	- 56	- 60	- 64	- 67	- 73	- 78	- 84	- 89
- 40	- 69	- 68	- 61	- 53	- 54	- 55	- 58	- 63	- 66	- 70	- 73	- 78	- 83	- 89	- 94

Corrections to power check pressure (psi)
Table 201

LIST OF EFFECTIVE PAGES

CHAPTER 78

EXHAUST

SECTION	PAGE	DATE
*78	i	July 10/92
*78	ii	July 10/92
*78	1	July 10/92
*78	2	July 10/92
*78-0	1	July 10/92
*78-0	2	July 10/92
*78-0	201	July 10/92
*78-0	202	July 10/92
*78-0	203	July 10/92
*78-0	204	July 10/92
*78-0	205	July 10/92
*78-0	206	July 10/92

* Pages revised by Revision 47
** Pages added by Revision 47
*** Pages deleted by Revision 47

TABLE OF CONTENTS

CHAPTER 78

EXHAUST

ENGINE EXHAUST SYSTEM — DESCRIPTION / OPERATION

1. **General**

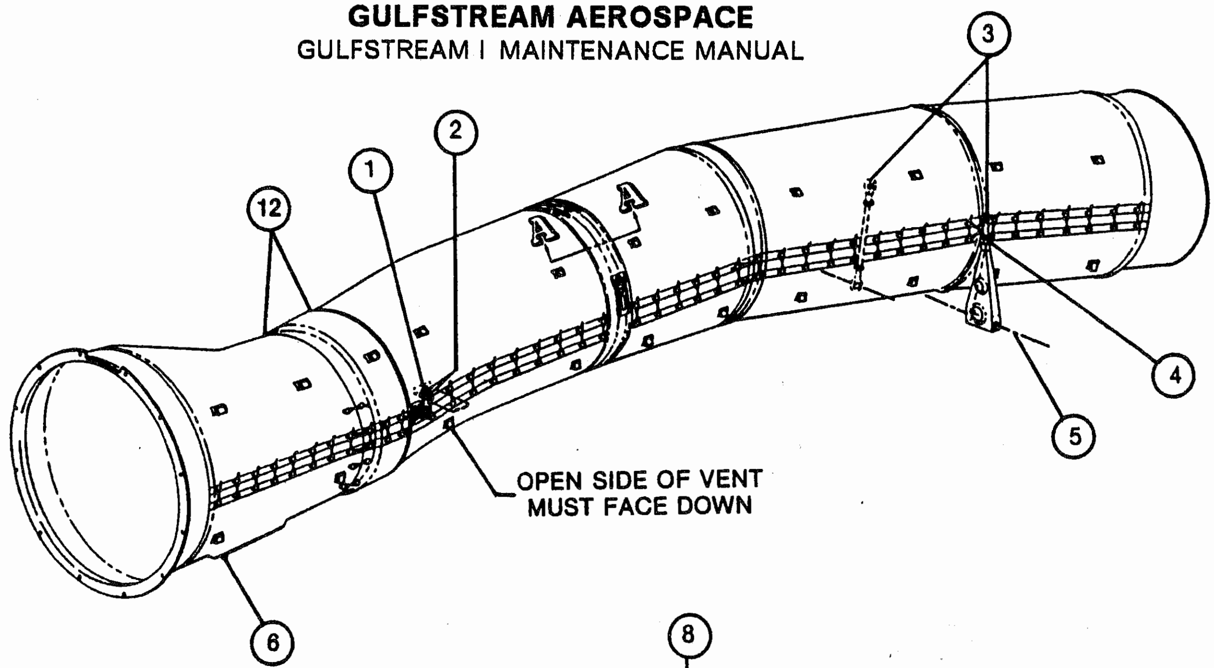
(See Figure 1)

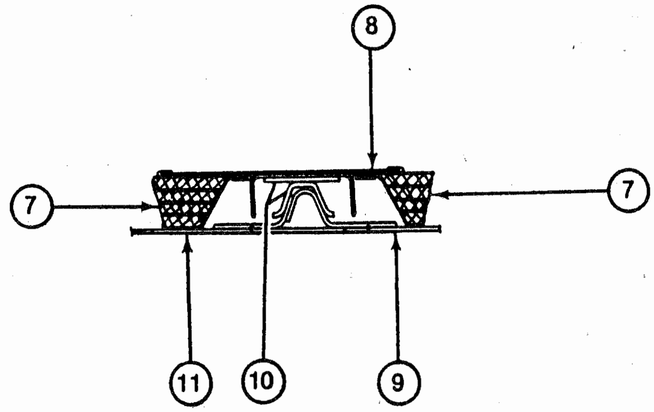
The exhaust system directs hot exhaust gases from the turbine section of the engine aft and overboard through a tailpipe assembly. This tailpipe assembly is mounted in the nacelle rearward extension and is attached to the firewall, engine mount tubes behind the firewall and wing upper surface. The forward section of the tailpipe is funnel shaped and surrounds, but does not contact, the engine exhaust section. An annular gap is thus formed between the engine exhaust section and tailpipe. This gap serves as an air ejector for air surrounding the engine hot section. As high velocity engine exhaust gas enters the tailpipe, it creates a low pressure effect which causes air from around the engine hot section to flow through the annular gap into the tailpipe. Air entering the cooling scoops in the engine cowl passes around the engine and exhausts through the tailpipe.

2. **Description**

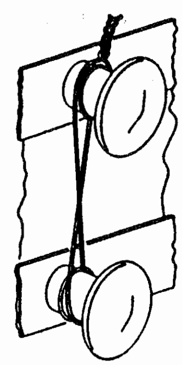
The tailpipe assembly is made up of two sections; forward funnel shaped section and rear section. Each section is constructed of corrosion resistant steel and all seams are closed by a continuous weld. A corrosion resistant high temperature clamp is used to secure the two sections in a gas tight joint. The funnel shaped front portion of the forward section has a mounting flange welded around its forward edge. This mounting flange mates with the engine side of the firewall and is secured to it with screws. An integral bellows section in the forward section of the tailpipe allows for expansion between firewall and fixed mounting at Fuselage Station 290. Two bearing fittings, one on each side of the pipe, are attached to the nacelle structure at Fuselage Station 290 and can be adjusted to move tailpipe in a vertical plane. The rear section of the tailpipe is secured to the airframe at Fuselage Station 365 by two support arms, one on each side of pipe. The support arms are attached to the wing upper surface in such a manner as to allow free movement fore and aft to compensate for expansion. Two bearing fittings which can be adjusted vertically are provided on support arms to adjust tailpipe vertically. The two rear support arms, plus the clamp between fore and aft sections of the tailpipe form the attaching points for the rear section.

The tailpipe assembly is wrapped in an insulating blanket to shield the surrounding area from the high heat generated by the exhaust gases. The blanket is laminated of thin stainless steel sheet outside and fiberglass inside. It is secured in place around the tailpipe by stainless steel safety wire which is laced around buttons. A stainless steel strap is positioned over the tailpipe clamp and is secured by T-bolts and high temperature self-locking nuts. This strap forms a heat shield for the uninsulated area.





DETAIL A



DETAIL B
TYPICAL FASTENING

1. Forward Bearing Fitting Nacelle Station 290 (Left Side Shown)
2. Adjustment Nuts
3. Rear Bearing Fitting
4. Adjustment Lock Nut
5. Nacelle Station 365
6. Tailpipe Drain Fitting
7. Blanket
8. Tailpipe Clamp Strap
9. Aft Tailpipe
10. Tailpipe Clamp
11. Forward Tailpipe
12. Bellows Inspection Blanket

Engine Exhaust System
Figure 1.

ENGINE EXHAUST SYSTEM — MAINTENANCE PRACTICES

1. Forward Tailpipe — Removal / Installation

NOTE: Tailpipe assembly must be removed in two separate sections, forward and aft, and cannot be removed as a single assembly.

A. Removal

NOTE: Engine, accessory gearbox access cover, and tailpipe clamp access doors must first be removed before forward section of tailpipe.

(1) Remove engine. (See Engine — Removal / Installation, Chapter 71)

(2) Remove accessory gearbox access cover.

(3) Remove tailpipe clamp access doors.

(4) Remove steel strap around tailpipe clamp. NOTE orientation of clamp.

(5) Remove clamp securing forward to aft section of tailpipe.

(6) Disconnect fluid drain line at tailpipe fitting drain trap.

(7) Remove nuts securing bearing fittings to nacelle support brackets.

(8) Remove hardware securing tailpipe flange to nacelle firewall.

(9) Remove forward section by pulling forward and guiding past opening in firewall, be careful to avoid damaging insulating blanket.

B. Installation

(1) Remove aft tailpipe cover with upper half of tadpole seal and 1/4 inch mounting bolts to allow for proper installation of tailpipe clamp.

NOTE: When installing tailpipe, flange cutout must be centered at top to clear drive shaft mounting pad.

(2) Guide forward section of tailpipe through opening in firewall and under accessory drive drip pan.

(3) Insert threaded end of bearing fitting into support bracket; install nut hand tight.

(4) Align holes in flange with firewall and secure with hardware previously removed.

NOTE: Install long screws on each side of cutout in flange. If replacement tailpipe is being installed, it will be necessary to drill holes in mounting flange for screws. There are two predrilled pilot holes at top on either side of cutout used to position tailpipe section to firewall. Remaining 10 holes are picked up from firewall.

(5) Align front and rear sections; install 1/4 inch mounting bolts left and right; install clamp and tighten.

(6) Position clamp halves where clamp trunnions are on right and left sides of centerline of tailpipe.

(7) Check for clearance of 3/4 inch between bottom of tailpipe assembly and wing structure heat shield; adjust nuts on bearing fittings at support bracket to move tailpipe up or down to obtain clearance.

(8) Tighten nuts after adjustments have been made. Torque nuts to 40 inch pounds above torque required to run nut down T-bolt.

(9) Install steel strap over tailpipe clamp and secure.

(10) Connect fluid drain line to fitting on tailpipe drain trap. Install tailpipe clamp access doors.

(11) Install accessory gearbox access covers.

(12) Install aft tailpipe cover with upper half of tadpole seal, fitting snug around tailpipe.

(13) Install engine. (See Engine — Removal / Installation, Chapter 71)

2. Aft Tailpipe — Removal / Installation

A. Removal

(1) Remove tailpipe cover with upper half of tadpole seal.

(2) Remove tailpipe clamp access doors.

(3) Remove steel strap from tailpipe clamp.

(4) Remove 1/4 inch bolts securing bearing fitting of support arms to tailpipe.

(5) Remove clamp securing forward and aft sections. NOTE orientation of clamp.

(6) Remove aft tailpipe assembly: exercise care to avoid damage to insulating blanket.

B. Installation

(1) Guide aft tailpipe into position and align mounting studs with bearing fittings.

(2) Align forward and aft sections of tailpipe: install 1/4 inch mounting bolts left and right: install clamp and tighten.

NOTE: Position clamp halves where clamp trunnions are on right and left sides of center line of tailpipe.

(3) Install steel strap over clamp.

(4) Check for clearance of 3/4 inch between bottom of tailpipe assembly and wing structure heat shield, adjust nuts on bearing fittings at support bracket to move tailpipe up or down to obtain required clearance.

(5) Tighten nuts after adjustments have been made. Torque nuts to 40 inch-pounds above torque required to run nut down T-bolt.

(6) Install tailpipe cover with upper half of tadpole seal, fitting snug around tailpipe.

(7) Install tailpipe clamp access doors.

3. Tailpipe Bellows — Inspection

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

NOTE: On Aircraft 1-5, engine must be removed if not modified with split blankets.

A. Working from inside main gear wheel well, cut lockwire holding bellows blanket halves.

B. Remove bottom half of blanket, then slide top half around to bottom for removal.

C. Inspect tailpipe bellows for cracks and evidence of failure. If cracks are noted, see Engine Tailpipe Cracks — Limitations and Repair this Section.

CAUTION: WHEN INSTALLING BLANKET HALVES TOGETHER, ENSURE GROOVE IN REAR TAILPIPE BLANKET IS MATED WITH LAND ON INNER AFT SURFACE OF INSPECTION BLANKET. DO NOT INSTALL INSPECTION BLANKET HALVES IN REVERSE AS LARGE GAP MAY RESULT IF GROOVE AND LAND ARE NOT PROPERLY ALIGNED.

D. Install inspection blanket by sliding top half around bellows into proper position.

E. Position bottom half of inspection blanket.

F. Using lockwire, secure two halves of inspection blanket together, then secure to forward and rear tailpipe blankets.

G. Inspect blanket for position and fit.

NOTE: A 1/8 inch clearance between bellows and inner surface of blanket must be maintained to allow bellows to expand and contract freely.

4. Tailpipe and Tailpipe Blanket — Inspection

NOTE: If aft tailpipe only is to be removed, engine need not be pulled.

A. Remove engine. (See Chapter 71 for procedure)

B. Remove tailpipe assemblies. (See procedures this section)

C. Remove blankets from tailpipes. If damage is noted. See Engine Tailpipe Blanket — Limitations and Repair, this section.

D. Inspect tailpipes for cracks, dents and evidence of weld failures, pay particular attention to bellows section. If cracks are noted, see Engine Tailpipe Cracks — Limitations and Repair, this Section.

E. Inspect tailpipe blankets for tears and evidence of failure. If damage is noted, see Engine Tailpipe Blanket — Limitations and Repair, this Section.

F. Install tailpipe blankets as follows:

(1) Position blanket on tailpipe with left and right joints following horizontal centerline of tailpipe.

(2) Attach clamping tool to upper and lower bottoms of blanket and pull together evenly.

(3) Lace with steel wire lace remaining buttons and remove clamping tool.

CAUTION: A 1/8 INCH CLEARANCE BETWEEN BELLOWS AND INNER SURFACE OF BLANKET MUST BE MAINTAINED TO ALLOW BELLOWS TO EXPAND AND CONTRACT FREELY.

G. Install tailpipe assemblies.

H. Install engine, See Engine — Removal / Installation, Chapter 71.

5. Engine Tailpipe Blanket — Limitations / Repair

A. Types of damage not needing repair.

(1) 12 pinholes or less per side (minimum distance 12 inches between pinholes).

(2) Slits that are less than 1/8 inch in length.

(3) Minor damage such as wrinkled edges.

(4) Small snags in screen.

B. Allowable repairs with spot welding.

(1) Holes from 1/16 inch diameter to 6 inches diameter.

(2) Slits from 1/8 inch to 18 inches in length.

(3) Up to 25% of a blankets total surface area. (This includes both sides.)

C. Allowable repairs (other than spot weld)

(1) Replacement of hooks and grommets.

(2) Reworking wrinkled edges with hand crimper.

(3) Replacement of batt insulation when it is forced out due to exterior foil damage.

D. Repair by Spot Weld

(1) Use duck-bill snips to trim foil or screen to required size for pitch. Use 0.005 Stainless Steel foil type 302 or 321 annealed. Allow a minimum of 1/4 inch distance all around. Patch material must be clean and edges should be smooth. Spot weld should not exceed 1/4 inch spacing.

(2) Place patch foil over damaged area, being careful to align patch properly so that edge distances are adequate. Where necessary, form patch to fit grooves.

(3) Proceed to weld around edges of patch, placing welds uniformly, not exceeding 1/4 inch apart.

6. Engine Tailpipe Cracks — Limitations / Repairs

NOTE: Cracks may be primarily evident in there region of bellows forward and aft flanges. If inspection reveals the existence of cracks, check length of crack using dye penetrant for accuracy if necessary.

A. Tailpipe Limitations

(1) If crack length does not exceed 4 inches and immediate area surrounding crack does not exhibit deformation*, continued operation is authorized, provided that a 25 fight hour visual inspection is performed. Inspection should monitor crack growth and any development of local deformation.

(2) Cracks in excess of 4 inches and to a maximum of 6 inches without local deformation* are authorized for continued operation provided that a 10 flight hour visual inspection is performed. Inspections intent is the same as 25 hour inspection.

(3) Immediate repair or replacement of tailpipe is mandatory when the following conditions occur:

(a) When crack length exceeds 6 inches.

(b) When any crack exhibits deformation.

(c) When more than two cracks are present on bellows.

*Deformation: Is defined as the displacement of material adjacent to the crack, which creates a scoop in the tailpipe. A scooped condition, either convex or concave, will direct exhaust gases into the nacelle. Prolonged exposure to exhaust gases can cause structural and component damage within the nacelle.

B. Tailpipe Bellows Crack Limitations

NOTE: The following conditions are acceptable on the bellows:

(1) If radial crack length does not exceed 4 inches and the immediate area surrounding the crack **does not** exhibit deformation*, continued operation is authorized, provided that a 25 flight hour visual inspection is performed. The inspection should monitor crack growth and any development of local deformation. (See Figure 201)

(2) Radial Cracks in excess of 4 inches and to a maximum of 6 inches without local deformation* are authorized for continued operation provided that a 10 flight hour visual inspection is performed. The inspections intent is the same as the 25 hour inspection.

(3) If axial crack does not extend past first rise on bellows and the immediate area surrounding the crack **does not** exhibit deformation*, continued operation is authorized, provided that a 25 flight hour visual inspection is performed. The inspection should monitor crack growth and any development of local deformation. (See Figure 201)

(4) Immediate repair or replacement of the tailpipe is mandatory when the following conditions occur:

(a) When the radial crack length exceeds 6 inches.

(b) When any crack exhibits deformation.

(c) When more than two cracks are present on the bellows.

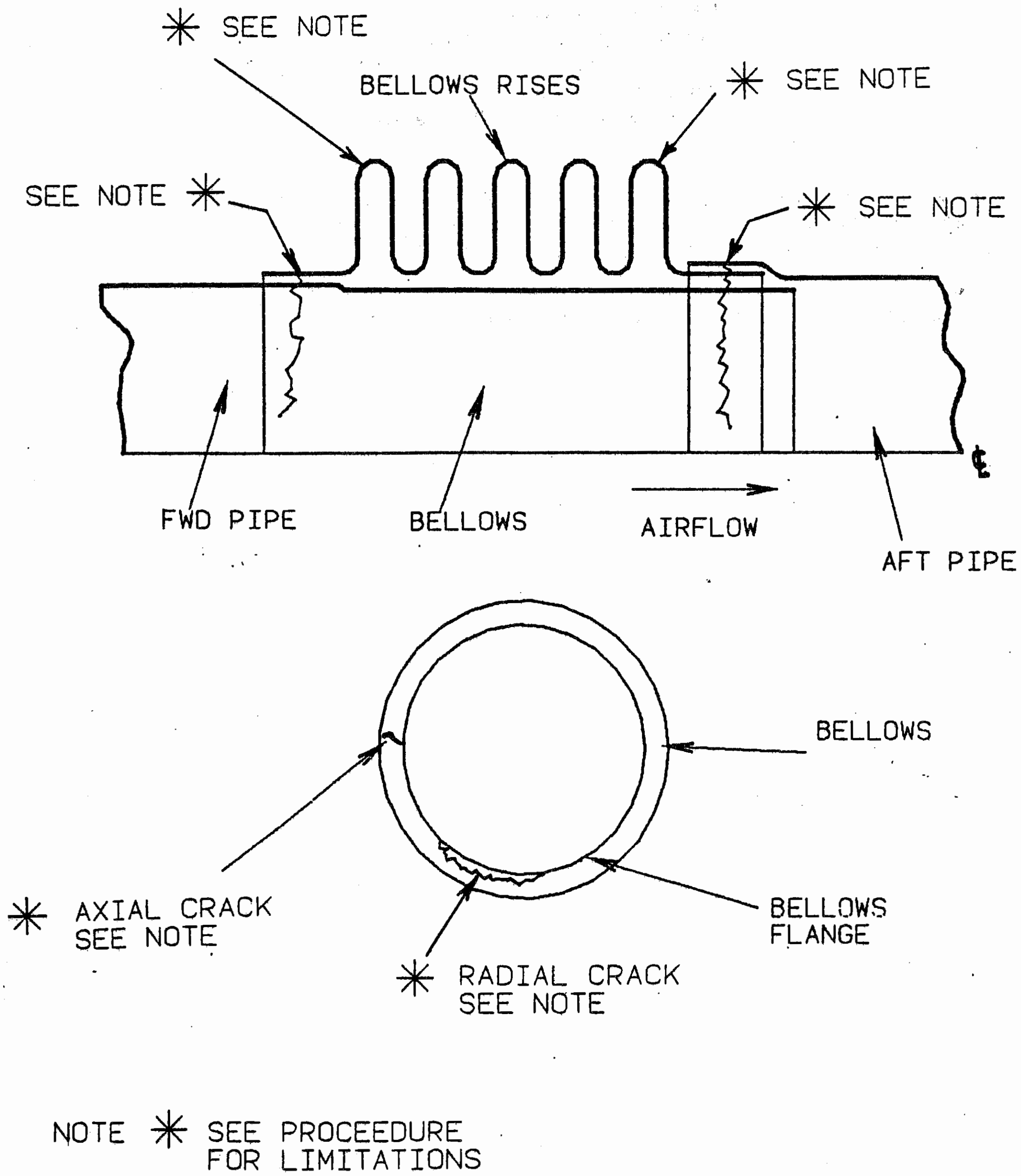
(d) When any axial crack extends past first rise on bellows (either end).

*Deformation: Is defined as the displacement of material adjacent to the crack, which creates a scoop in the tailpipe. A scooped condition, either convex or concave, will direct exhaust gases into the nacelle. Prolonged exposure to exhaust gases can cause structural and component damage within the nacelle.

C. Repair

Gulfstream Aerospace does not recommend field welding of cracks. However, if the operator elects to weld and assume the risk, the tailpipe material is 19-9DL (with an alternate of Hastelloy x per AMS 5536). Both are weldable using weld rod ERNiCrMo-2 per AWS 5.14.

The welded repair should have a recurring 75 flight hour visual inspection until it is assured that the welded area is satisfactory, after which the standard 150 flight hour inspection should be resumed.



Bellows
Figure 201.

LIST OF EFFECTIVE PAGES

CHAPTER 80

STARTING

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CHAPTER 80

STARTING

STARTING SYSTEM — GENERAL

1. General

A. This section will be separated into 2 parts:

STARTING SYSTEM - Pertains to all aircraft

IGNITION SYSTEM - Pertains to all aircraft

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STARTING — DESCRIPTION / OPERATION

1. Description

A. The engine starting system has four functions:

- To provide cranking of the engine for ground starts.
- To supply ignition energy for both ground and air starts.
- To inform the crew when ignition is in use.
- To inform crew when the starter is engaged (Aircraft having ASC 209).

B. There are three starting controls for ground use, which are located centrally in the lower overhead panel.

- The ENG SELECTOR switch has positions L, OFF, and R, which allow selection of the engine to be activated.
- The START SELECTOR switch has positions START, OFF and CRANK.
- The third control is the ENGINE START switch, which is a momentary push button. (See Figure 1)

C. Each engine is equipped with an electrically driven starter motor, and a starter relay. The electric starter has the task of overcoming inertia friction and windage effects to crank the engine and propeller to an rpm sufficient for combustion within the engine to take place.

Thereafter, the starter must assist the combustion gases in accelerating the moving parts to a point where the engine will run self-sustaining.

- The main dc bus provides 28V dc power to the control circuit through the GND START circuit breaker.
- The starter motor is electrically connected to the dc bus tie cable when the starter relay is energized.

D. The starter relay actually contains two relays:

- A main contactor relay.
- A current sensitive relay.

E. Refer to Chapter 80 of the Rolls-Royce Maintenance Manual (M-Da7) for maintenance details of igniter plugs and leads.

2. Ground Starting Sequence

A. The starter relay main contactor relay is energized by setting the START SELECTOR switch to START, the ENG. SELECTOR switch to the desired engine, and pressing the ENGINE START push button. The current sensitive relay is then energized by the current flow to the starter motor and acts as a holding circuit on the main contactor relay, providing current flow exceeds 150A. At this time the starter engaged light will come on (Aircraft having ASC 209). It is not necessary to hold the ENGINE START push button switch depressed. At the same time that the main contactor relay is energized, the appropriate ignition relay is energized to provide power to the proper ignition units. When the current flowing to the starter motor drops to 150A (starter motor rpm sufficient for the engine to maintain self-sustained operation), the current sensitive relay contact opens, de-energizing the main contactor relay.

At this time the STARTER ENGAGED light will go off (Aircraft having ASC 209)

NOTE: STARTER ENGAGED AND IGNITION LIGHTS SHOULD BOTH COME ON AND GO OUT SIMULTANEOUSLY. (AIRCRAFT HAVING ASC 209).

B. In general the following occurs on an engine start:

These numbers vary in response to a multitude of factors.) At 0 to 1600 rpm the starter turns the engine over, from 1600 - 3500 rpm combustion occurs, the starter aids the engine and then cuts out. Ignition also drops out. Within the range 3500 to IDLE rpm (6500 - 7500 rpm), the engine accelerates, self-sustaining.

3. **Description - Cranking**

For cranking the engine without starting it, the START SELECTOR switch is set to CRANK and the ENG SELECTOR switch is set to L or R. The starting system functions in the same manner as described above, except that the ignition relay is not energized. The engine will crank, but without ignition or fuel there can be no combustion. This cycle will continue until one of the two switches is moved to the OFF position. It is advisable not to terminate the cycle for at least 3 seconds after its initiation, and it is mandatory not to extend beyond 30 seconds.

4. **Air Start Sequence**

Air starts require only ignition. The starter is not involved. The engine is allowed to windmill via control of propeller blade angle. The appropriate AIR START ignition switch is moved to ON. Power is supplied directly to the engine ignition units which activate the igniter plugs. Admission of fuel to the engine produces combustion, and the air start is complete. For detailed description See Section 80-1-1.

5. **Emergency DC Operation — Effect on System**

Ground starting or cranking is not available in EMERGENCY.

Airstart ignition is available in EMERGENCY.

6. **Crash Start Prevention**

A. To prevent accidental engagement of the starter with the engine gear train while the engine is rotating but not running, an anti-crash relay, has been incorporated in the aircraft starter circuit. This relay is so called because it acts to prevent gear crashing damage to the engine gear train, which would result if starter engagement with the engine gear train were attempted while they were in motion. The function of this relay is to limit application of power to the starter motor to 75 milliseconds, unless sufficient current builds up to show that the starter is engaged. Therefore, if the starter button is depressed in an attempt to start on a rotating, but not running engine, the anti-crash relay will break the button circuit within 75 milliseconds. Since the starter is not under load, because the engine is already rotating, the starter undercurrent control relay will not pull in, since less than 150A is passing through it. This circuit will not allow sufficient time for the starter to attain enough speed to cause crash damage to the engine because power is being supplied for only 75 milliseconds. During normal starts or cranking with the engine not rotating, the control relay will hold in, thus maintaining starter engagement until completion of a normal start or cranking period.

B. Crash damage to an engine cannot occur when the engine is running since the mechanical engagement system of the starter to the engine is so constructed that the centrifugal force of the running engine will keep the starter dogs from engaging. Therefore, the critical time for crash damage is when the engine is turning, but not under its own power. The actual rpm at which the starter dogs disengage due to centrifugal force is determined by the engine manufacturer. As a rule of thumb, no engine start should ever be attempted on an engine which is turning. If starter engagement is attempted whether accidentally or otherwise, the anti-crash relay in the engine starting control system is designed to protect against damage.

7. **Major Electrical Components**

- There are two Lucas Rotax starter motors, one mounted on the upper right front portion of each engine. Each are rated at 3800 RPM and 550 amps of current. They are of the four pole, compound type with four pole shunt coils connected in series and the series field coils connected in series-paralleled
- Two Hartman electric starter relays, one mounted on each nacelle relay panel. These units are undercurrent relays rated at 600A with 100% overload capabilities. Current passing through the unit maintains the main contacts closed until the current declines to 150A, at which time the main contacts open.
- Two Leach ignition relays. Each is a 10A, 4 PDT hermetically sealed relay located in the left Fuselage Station 133 relay panel.
- One Rotax anti-crash relay located under FLR No. 6, just forward of the pitot heat sensing relays between Fuselage Station 206 and 219, right side. This relay prevents crash starts, and subsequent engine damage.

  WARNING: THE ELECTRICAL ENERGY DESCRIBED HERE IS LETHAL.

- Four Lucas Rotax ignition units, two located in each nacelle. These are capacitor discharge units utilizing 24V dc input power to produce output energy surges of 12 joules. Each surge lasts 50 microseconds and is 1500A at 2000V. Discharge rate is once per second.
- Four Rolls-Royce igniter plugs, located in the combustion No.3 and No. 2 cans of each engine. These units are supplied with the engine.
- Four forward and four aft high tension ignition leads. These are shielded high tension cable assemblies designed to mate with engine igniter plugs and ignition boxes. Firewall fittings (Smiths) allow forward and aft units to connect.

8. Ground Starting — Details of Operation

(See Figure 1)

A. To start, the operator selects the engine desired, and positions the START SELECTOR switch to START. If the engine is fitted with a propeller brake, this control must be released. On Aircraft 149 - 200, 322 and 323 the prop brake is not installed. The starting button is then pressed. Energy is supplied from the button, through the anti-crash relay contacts 3 and 4 to terminal C of the proper starting relay. The main contacts close allowing main dc bus energy to flow through the starter. The flow of current through the starting relay current sensitive coil causes the control contacts to close completing a circuit which will maintain the main contacts closed as long as control energy is present at terminal S, and until the main current flow decreases to its preset dropout value. These actions occur very rapidly and allow the operator to release pressure on the start button after only a moments pressure.

B. Simultaneous with the energizing of starting relay terminal C, energy actuates the proper ignition relay. This device powers the engine ignition units, causing the igniter plugs to begin sparking. Whenever ignition is in use it will be indicated by the master warning light system. The ignition continues in operation, now maintained by the current sensitive contacts in the starter relay.

C. Also simultaneous with the energizing of starting relay terminal M, power is applied to the starter and also to terminal 1 of the anti-crash relay, breaking the starter button circuit. This occurs within 75 milliseconds from the time that the button is depressed (0.075 of a second), regardless of how long the starter button is held depressed. During this time the STARTER ENGAGED light will come on (Aircraft having ASC 209).

D. Coincident with the energizing of starter relay terminal C, a feed is routed to an engine start interlock relay, which functions in the dc power system to connect an added heavy power cable into the dc system to aid in carrying loads from one battery to the other across the aircraft. Placing this extra heavy cable connecting the two batteries, when in a starting sequence, will lessen the cross-aircraft resistance between batteries, thus allowing more equal sharing of the load. This cable is actually the essential dc bus feeder cable, and is used in this case as a starting aid. As soon as the starting cycle is completed, the interlock relay drops out and this cable becomes dormant until the dc system is placed in EMERGENCY.

E. At an appropriate time, the operator allows fuel to reach the engine, and combustion takes place. Acceleration continues until the starter current falls to 150A at which point the current sensitive contact opens, cutting off ignition power, relaxing the engine start interlock relay, and allowing the starter relay 1main contacts to open, turning off the STARTER ENGAGED light and de-energizing pin M. When pin M becomes de-energized, the anti-crash relay is relaxed in preparation for its next use. Electrically, the start is now complete, although the engine still has to accelerate itself to idle rpm to obtain a steady-state condition.

F. During the foregoing period, the operator has at his disposal means for halting the cycle at any time. Turning fuel off will cause combustion to cease, and placing the engine start selector switch to OFF will remove the control energy from the start relay, causing all electrical activity to cease.

NOTE: Limiting factors for a ground start are the engines turbine gas temperature (TGT), and the total duration of starter effort. Both of these are controlled by crew training. If TGT approaches 9300 or if a start has not reached the self-sufficient stage within 30 seconds, the start should be aborted.

9. **Engine Starting Procedure — Battery Conservation**

A. MONITOR BUS TEST SWITCH shall be in the OFF position.

B. Check the battery state. With all individual circuit switches in OFF position, move the battery switch to NORMAL, and turn on inverter E and one fuel pump. Observe bus voltmeter. A reading below 22.0V indicates the batteries are discharged to a degree where difficulty will be experienced if a start is attempted.

C. Charge batteries and/or connect external power as required.

CAUTION: AFTER SIX CONSECUTIVE START ATTEMPTS OR MOTORING CYCLES, WAIT 15 MINUTES TO ALLOW STARTER TO COOL ONLY FOUR CONSECUTIVE START OR MOTORING CYCLES ARE PERMITTED ON INTERNAL BATTERIES.

D. Proceed with engine starting as follows: (Three methods are listed in order of preference.)

(1) Method I

- Start both engines in quick succession from batteries and/or external power.
- With both engines at 8000 rpm or above, place generator switches ON.

(2) Method II

- Start one engine.
- Run engine at 8000 rpm or above, place respective generator switch ON.
- Wait until ammeter shows movement and stabilizes.
- Place generator switch OFF. Start second engine.
- With both engines at 8000 rpm or above, place both generator switches ON.

CAUTION: DO NOT COMBINE METHOD III WITH METHOD I. IF USING METHOD III, DO NOT USE GENERATOR TO BOOST MORE THAN TWO STARTS OR CRANKS IN ANY 15 MINUTE PERIOD.

A BOOST START HEATS THE GENERATOR RAPIDLY AND MUST BE FOLLOWED BY A PERIOD OF MODERATE DUTY. RUNNING GENERATOR'S ENGINE AT 11,000 RPM WILL INDUCE COOLING AIRFLOW THROUGH GENERATOR. IF KEPT BELOW 11,000 RPM, THE GENERATOR SHOULD BE TURNED OFF TO PERMIT COOLING.

(3) Method III

NOTE: This is the same as Method II except for the following:

- During the start of the second engine, after cranking has been in progress for 5 seconds, place first engine generator switch ON to achieve a boost effect.

10. **Starting External Power Requirements**

A. The aircraft is capable of being started without external power however, ground power is often desirable. In such cases the following applies:

(1) Ground Checks

External power devices using three pin oval plug may be used. The size of unit depends upon the amount of equipment to be energized. A capacity of 1000A is recommended as a minimum size because of aircraft battery inrush currents. The ground source of voltage can be within limits of 24 to 29.5V under load. If ground source is to remain connected for an extended time, its voltage should not exceed 29V.

NOTE: Shutdown APU generator before engine start. An APU generator/main engine starter interlock circuit may be used to ensure generator is shutdown before engine start. (Refer to Chapter 24 for details.)

(2) Engine Starts

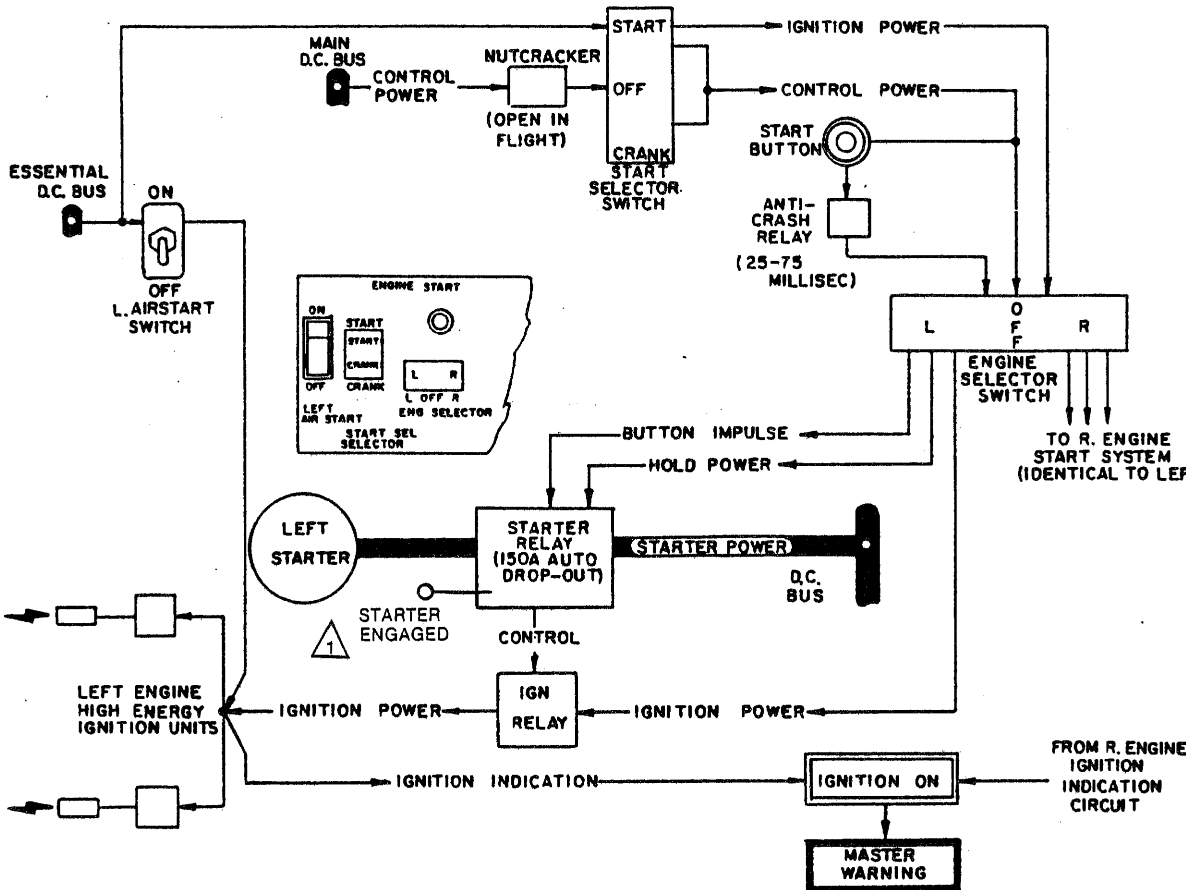
Ground power serves the dual purpose of cranking engine and charging batteries. The voltage of ground power unit determines extent of its participation.

A well regulated, 600A capacity ground unit should be adjusted to 27.5V no load. During a start cycle, this will deliver full output for about 20 seconds.

With softer control, the no load voltage setting may have to be higher. Voltages higher than 30V should only be used when necessary, and should be promptly disconnected after start to avoid gassing the batteries.

An ideal ground start unit would deliver up to 1500A at 22V, graduating to no load at 29V. It should be noted that this is an ideal condition only.

For reference only, starter inrush is approximately 1300A. After about 5 seconds, aircraft batteries relinquish starting load to outside power and begin taking a charge.



Aircraft having ASC 209

Engine Starting System — Block Diagram
Figure 1.

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STARTING — MAINTENANCE PRACTICES

CAUTION: BEFORE PERFORMING ANY OF THE FOLLOWING PROCEDURES, ENSURE ELECTRICAL POWER IS OFF. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

1. Engine Starter and Clutch — Maintenance Practices

Refer to Rolls-Royce Dart Maintenance Manual M-Da7-G.

2. Engine Starter Relay — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Remove nacelle relay panel access plate to gain access to starter relay.

NOTE: Start relay is lower forward large relay identified as: RELAY - 600 AMP - SPST - AUTOMATIC DROPOUT.

(2) Disconnect leads and bus bars from relay terminals and identify.

(3) Remove mounting hardware and relay.

CAUTION: ENSURE ELECTRICAL POWER REMAINS OFF WHEN RELAY IS DISCONNECTED AS ONE BUS BAR IS CONNECTED TO MAIN DC BUS.

B. Installation

(1) Install relay using removed hardware.

(2) Connect bus bars and leads to correct terminals.

(3) Perform engine cranking cycle to check relay operation.

(4) If satisfactory, inspect for presence of foreign objects then replace nacelle access panel.

3. Anti-Crash Relay — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF BEFORE REMOVING OR INSTALLING COMPONENT. DISENGAGE APPLICABLE CIRCUIT BREAKER OR FUSE.

A. Removal

(1) Gain access to anti-crash relay located under floor (forward cabin, right hand side) by removing FLR No. 6.

NOTE: Relay is located on a mounting bracket with two pitot heat sensing relays. It can be identified by a black dome shaped cover.

(2) Tag and disconnect electrical leads from terminal board on relay.

(3) Remove relay.

B. Installation

(1) Install relay on mount using hardware previously removed.

(2) Connect electrical leads to proper terminals as previously identified.

(3) Perform Anti-crash Relay Functional Test, before installing FLR No. 6, see this Section.

(4) Inspect for presence of foreign objects then install FLR No. 6.

4. Anti-Crash Relay — Functional Test

A. Open engine cowling to gain access to either engine starter.

B. Disconnect positive lead from starter terminal at starter, connect positive lead of voltmeter to disconnected starter lead and negative lead of voltmeter to ground (set voltmeter to accept 28V dc).

CAUTION: CAREFULLY INSULATE STARTER LEAD WITH VOLTMETER LEAD ATTACHED FROM STRUCTURE.

NOTE: A 24V test light may be substituted for voltmeter.

C. Energize main and essential dc buses.

D. Place START SELECTOR switch to CRANK and ENGINE SELECTOR switch to applicable engine.

WARNING: UNDER NO CONDITION SHOULD START BUTTON EVER BE DEPRESSED UNLESS PROPELLER IS FULLY STOPPED.

E. With one man in cockpit and another observing meter (or test light), depress and hold in START button, should observe a very quick meter needle pulse (or a flicker of test light).

NOTE: Meter or test light should not show a continuous voltage indication but only a very short duration pulse even though button is still being held in.

F. Release START button.

G. De-energize main and essential dc buses.

H. Remove meter (or test light), connect starter cable to starter, torque securing nut to 110 inch pounds.

I. Inspect for presence of foreign objects then close engine cowling,

J. Inspect for presence of foreign objects then close install FLR No. 6.

CAUTION: IF CRANK CYCLE IS USED, ABORT CYCLE BY MOVING EITHER START-CRANK OR LEFT/RIGHT SWITCH TO OFF. OBSERVE STARTER DUTY CYCLE.

K. Perform normal start or crank of engine and check for normal start er operation (1800 rpm).

NOTE: Preceding quick check determines following:

(1) Anti-crash relay passes a momentary impulse to starter and if starter does not pick up a load in predetermined short time, power to starter is broken (meter pulse or light flicker).

(2) Anti-crash relay will give an impulse long enough to engage starter (starter holds in after START button is release).

IGNITION SYSTEM — DESCRIPTION / OPERATION

1. **Description — Ground Starts**

(See Figure 1 and Figure 2

The ignition system consists of an ignition relay, two high energy ignition units, four high tension leads, two firewall feed-through connectors and two igniter plugs for each engine. In addition, there is an airstart switch for each engine, and one indicating light, part of the master warning light system, for both engine ignition systems.

2. **Operation**

During ground start of the left engine, the left ignition relay is energized when the left starter relay and starter motor are energized (See Section 80-0). When the ignition relay is energized, it provides power from the essential dc bus through the IGN PWR circuit breaker (Aircraft 1 - 148, 322and 323 not having ASC 170) or through the L No. 1 and L No. 2 IGN circuit breakers (Aircraft 149 - 200 and 1 - 148, 322 and 323 having ASC 170) to the ignition units. The right ignition systems is identical to the left.

For airstarts, when either the left or right airstart ignition switch is set to ON, the dc power is routed directly to the corresponding ignition units.

On Aircraft 1 - 148, 322 and 323 not having ASC 170 the IGNITION ON indicating light comes on when either the left or right ignition units are energized. The essential dc bus provides dc power through the IGN. IND. circuit breaker for operation of this light. During ground starts, the ignition relay that is energized completes the circuit to the light. During air starts, the air switch that is set to ON completes the circuit to the light.

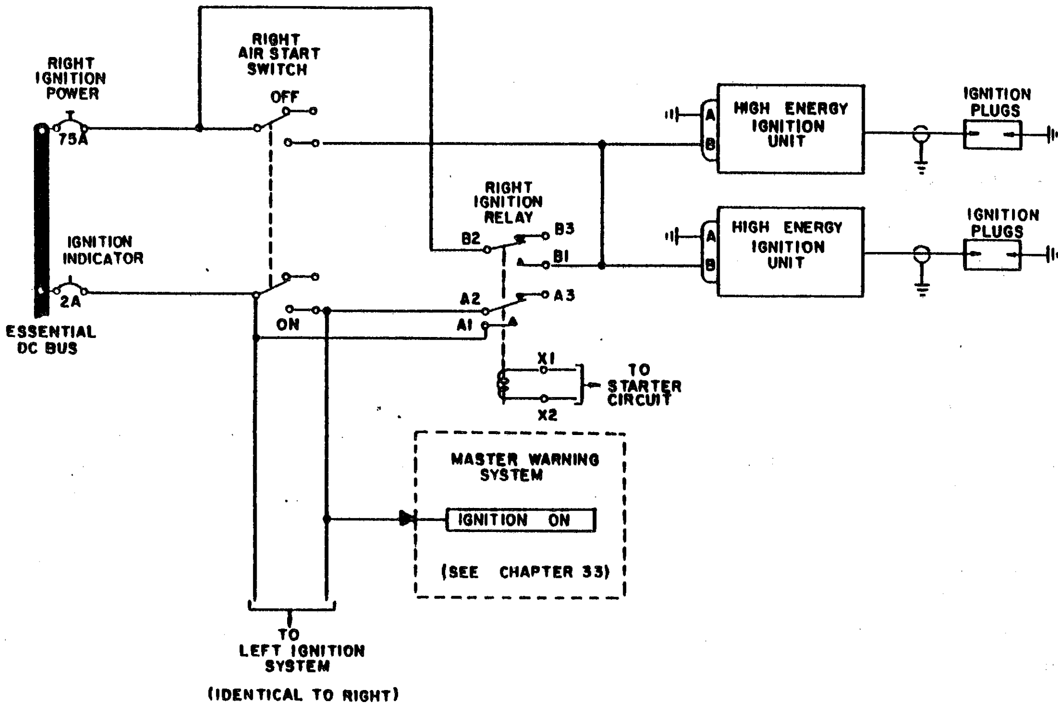
On Aircraft 1 - 148, 322 and 323 having ASC 170 and Aircraft 149 - 200 the IGNITION ON indicating light comes on when either the left or right, number 1 or number 2 ignition units are energized. When a high energy unit is energized a circuit is completed through a 1/2 AMP circuit breaker and a diode to the light. During ground starts, the ignition relay that is energized completes the circuit to the light. During air start, the air start switch that is set to ON completes the circuit to the light.

3. **Description — Air Starts**

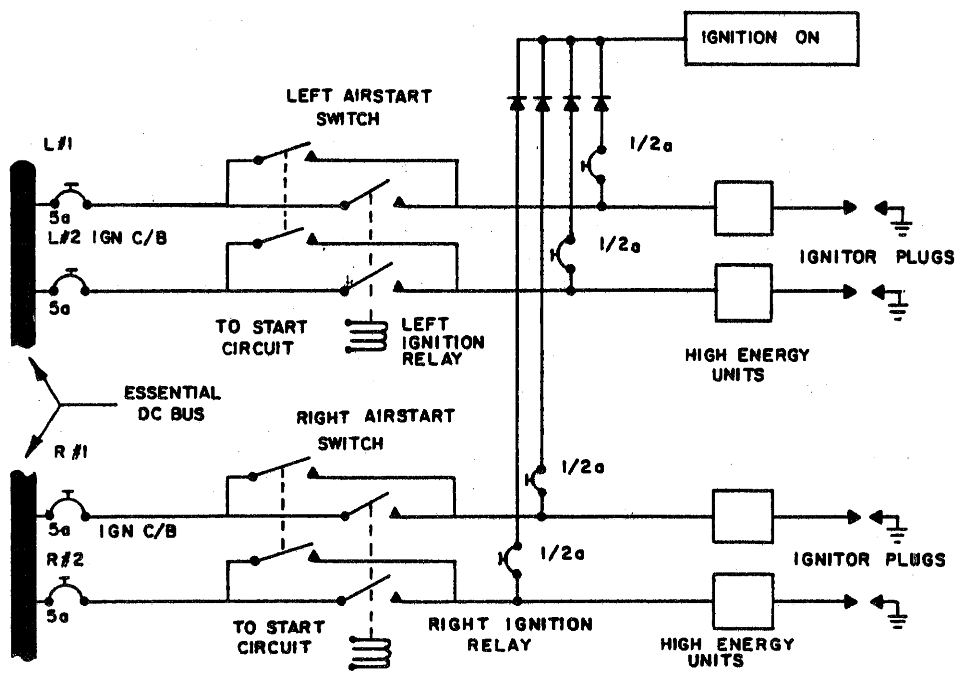
Air starts merely require ignition from the LEFT or RIGHT AIR START ignitions switch. The engine is allowed to windmill via control of propeller blade angle. The appropriate AIR START ignition switch is moved to ON. Power is supplied directly to the engine ignition units which activate the igniter plugs. As this occurs the IGNITION ON light is powered. This is a reminder to the crew to turn ignition off after using. Admission of fuel to the engine produces combustion and the engine accelerates until the air start is complete. The air start ignition switches are provided with mouse trap guards. Two distinct motions are therefore required to energize a set of igniters. This arrangement is a precaution against inadvertent applications of ignition power.

The energy produced from each ignition unit is in the order of 1500A at 2000V. It utilizes dc voltage through a vibrator and transformer, through an ionization gap and a capacitor discharge, producing this very high energy.

WARNING: REMEMBER THAT HIGH TENSION IGNITION ENERGY IS LETHAL. NO WORK SHOULD EVER BE PERFORMED ON OR NEAR THE IGNITION SYSTEM HIGH TENSION COMPONENTS WITHOUT FIRST TAKING STEPS TO ENSURE THAT POWER REMAINS OFF. FURTHERMORE, THE IGNITION UNIT CONTAINS A LARGE CAPACITOR, WHICH IN CERTAIN ISOLATED CASES OF BLEEDER RESISTOR FAILURE WITHIN THE UNIT, CAN REMAIN CHARGED OVER LONG PERIODS OF TIME. DISCHARGING OF THIS CAPACITOR THROUGH THE BODY CAN ALSO BE LETHAL. NEVER OPEN OR TAMPER WITH THIS UNIT IN ANY WAY UNLESS SPECIFICALLY QUALIFIED TO DO SO.



Ignition — Schematic
Aircraft 1 - 148, 322 and 323 Having ASC 170 and 149 - 200
Figure 1.



Ignition — Schematic
Aircraft 1 - 148, 322 and 323 Having ASC 170 and 149 - 200
Figure 2.

IGNITION SYSTEM — MAINTENANCE PRACTICES

1. **Engine Ignition Harness — Inspection / Check**

WARNING: ELECTRICAL ENERGY STORED IN CONDENSER OF A HIGH ENERGY IGNITION UNIT IS POTENTIALLY LETHAL. BEFORE TOUCHING UNIT, HIGH ENERGY LEAD OR IGNITER PLUG, ENSURE ELECTRICAL POWER WILL REMAIN OFF. DISCONNECT LOW ENERGY LEAD POWER SUPPLY FROM IGNITION UNIT AND WAIT FOR AT LEAST 1 MINUTE TO ALLOW STORED ENERGY TO DISSIPATE.

A. Leads and Feed-Through Fitting Inspection

(1) Gain access to engine ignition harness and igniter boxes by opening engine cowling and nacelle access covers as required.

(2) Disconnect following high energy leads:

- Igniter plugs numbers 3 and 7.
- Firewall feed-through fittings.
- Igniter boxes.

WARNING: USE SOLVENT IN A WELL VENTILATED AREA. AVOID BREATHING FUMES. KEEP AWAY FROM FLAMES.

(3) Inspect contact button at end of each lead for pitting and cleanliness, remove light pitting with a fine file or wire brush, clean buttons and insulators thoroughly with trichlorethane to remove any residue, reject any lead that is deeply pitted.

(4) Ensure there is NO gap between insulator and contact button.

(5) Inspect weld at each lead end fitting for cracks (area where braid is joined to fitting), reject any lead with cracks in this weld.

(6) Inspect firewall feed-through fittings and igniter plug recesses for pitting and cleanliness, clean contacts and wash thoroughly with trichlorethane to remove any residue.

B. Center Conductor Continuity Test

(1) Check continuity of each lead center conductor from one end button to other end button, using an ohmmeter or a dc power supply of 12 volts or less in series with a test light, reject any lead which does not show continuity.

C. Hi-Pot Test (Insulation Breakdown Check)

WARNING: DO NOT HOLD LEAD DURING THIS TEST.

CAUTION: OTHER END BUTTON OF LEAD UNDER TEST SHALL BE KEPT CLEAR OF STRUCTURE DURING TEST.

(1) Check that each lead can withstand a peak voltage of 10,000 volts (7070 volts rms) for 3 to 5 seconds, using a suitable high potential power supply connected between one end button and outer conduit, reject any lead which is not able to withstand this Hi-Pot test.

D. Full Harness Continuity and Hi-Pot Check.

(1) Check end to end harness center conductor continuity from center conductor contact of open end of nacelle connector to open ignitor end center conductor contact, using an ohmmeter or a dc power supply of 12 volts or less in series with a test light.

WARNING: DO NOT HOLD LEAD DURING FOLLOWING TEST.

CAUTION: KEEP END BUTTONS CLEAR OF STRUCTURE DURING TEST.

NOTE: This checks entire lead center conductor continuity from one end to other including firewall feed-through connector.

(2) Check that each complete harness and associated feed-through can withstand a peak voltage of 10,000 volts (7070 volts rms) for 3 to 5 seconds, using a suitable high potential power supply connected between one end button and outer conduit, and with other end of assembly disconnected from igniter box (or igniter).

NOTE: If system does not pass this test, firewall feed-through connector is faulty.

(3) Disconnect Hi-Pot Tester.

(4) Test bonding circuit resistance of high tension lead outer return conductor sheath and rigid end fittings using a Wheatstone bridge.

(a) Connect both rigid end fittings of high tension lead outer sheath to terminals of a Wheatstone bridge with a pair of copper test leads of known resistance not in excess of 0.020 ohms.

(b) Check total circuit resistance value of braided sheath and rigid end fittings does not exceed 0.050 ohms.

NOTE: The total resistance value of the high tension lead is the Wheatstone bridge reading minus the resistance value of the test leads.

(5) Connect engine lead to ignitor, torque nut to 216-240 inch pounds, safety wire nut.

(6) Connect nacelle lead to ignition box, torque nut to 216-240 inch pounds, safety wire nut.

(7) Return system to normal configuration.

(8) Close engine cowling and any access covers previously opened.

(9) Perform Engine Ignition System — Aural Check, see this Section.

2. Ignition Units — Removal / Installation

WARNING: HIGH ENERGY IGNITION IS LETHAL. NO WORK SHOULD EVER BE PERFORMED ON OR NEAR IGNITION SYSTEM HIGH TENSION COMPONENTS WITHOUT FIRST TAKING STEPS TO ENSURE POWER REMAINS OFF. IGNITION UNIT CONTAINS A LARGE CAPACITOR, WHICH IN CERTAIN ISOLATED CASES OF BLEEDER RESISTOR FAILURE WITHIN UNIT, CAN REMAIN CHARGED OVER LONG PERIODS OF TIME. DISCHARGING OF THIS CAPACITOR, THROUGH BODY CAN ALSO BE LETHAL. NEVER OPEN OR TAMPER WITH THIS UNIT IN ANY WAY UNLESS SPECIFICALLY QUALIFIED.

A. Removal

(1) Open main wheel well clamshell doors and secure in open position with ground safety struts.

(2) De-energize main and essential dc buses.

(3) Gain access to unit involved as follows:

- Left engine outboard unit - remove forward nacelle access panel.
- Right engine inboard unit - remove forward nacelle access panel.
- Left engine inboard unit - main wheel well.
- Right engine outboard unit - main wheel well.

(4) Disconnect low tension connector from unit involved, install protective cap on plug.

CAUTION: DO NOT DAMAGE INSULATOR AT END OF LEAD.

(5) Disconnect high energy connector from unit involved, install protective cap on lead.

(6) Remove unit.

B. Installation

(1) Install unit on mounting bracket using hardware previously removed.

(2) Remove protective cap and connect high energy lead connector to unit, torque nut to 216-240 inch pounds, safety wire nut.

(3) Remove protective cap and connect low energy connector to unit, safety wire nut.

(4) Remove main wheel well clamshell doors ground safety struts and close doors, install access panels.

(5) Perform Engine Ignition System — Aural Check, see this Section.

3. Engine / Nacelle Ignition Cable — Removal / Installation

WARNING: ELECTRICAL ENERGY STORED IN CONDENSER OF A HIGH ENERGY IGNITION UNIT IS POTENTIALLY LETHAL. ENSURE POWER WILL REMAIN OFF BEFORE TOUCHING UNIT, HIGH ENERGY LEAD OR IGNITER PLUG. DISCONNECT LOW ENERGY LEAD POWER SUPPLY FROM IGNITION UNIT AND WAIT FOR AT LEAST 1 MINUTE TO ALLOW STORED ENERGY TO DISSIPATE.

A. Removal

(1) Gain access to section of lead to be replaced.

NOTE: Section of each ignition system cable which connects igniter to firewall feedthrough connector is called engine cable. Section which connects ignition unit to firewall feedthrough connector is called nacelle cable.

(2) Break safety-wire and remove both ends from igniter and feedthrough, or feedthrough and igniter box (depending on section of cable to be replaced).

(3) Remove clamps holding cable to structure.

(4) Remove cable.

B. Installation

(1) Install cable and route in same manner as cable that was removed.

(2) Install clamps.

(3) Install each end fitting to appropriate receptacles, torque to 216-240 inch pounds, safety-wire fitting.

(4) Close access previously opened.

(5) Perform engine ignition system aural check, located in this section.

4. Engine Ignition System — Aural Check

WARNING: DO NOT PERFORM THIS CHECK ON ANY PART OF IGNITER SYSTEM BY REMOVING IGNITER PLUG TO VIEW SPARK, OR BY DISCONNECTING HIGH ENERGY LEAD TO MAKE AIR GAP BETWEEN END OF HIGH ENERGY LEAD AND PLUG TO PART OF ENGINE OR STRUCTURE, AS THIS COULD RESULT IN FATAL PERSONNEL INJURY. BEFORE CHECKING IGNITERS ENSURE RESIDUAL FUEL IN COMBUSTION CHAMBERS IS BLOWN OUT BY MOTORING ENGINE.

IF A SYSTEM DEFECT IS FOUND DURING IGNITION SYSTEM CHECK, BEFORE TOUCHING UNIT HIGH ENERGY LEAD OR IGNITER PLUG, DISCONNECT LOW ENERGY SUPPLY FROM IGNITION UNIT AND WAIT AT LEAST 1 MINUTE TO PERMIT STORED ENERGY TO DISSIPATE.

A. Energize essential dc bus.

B. Place START SELECTOR and ENGINE SELECTOR switches (lower overhead panel in cockpit) OFF.

C. Ensure HP cock levers are in FUEL OFF.

D. Place AIR START switch (engine being checked) on, Master warning panel IGNITION ON lights should come on.

E. Listen for sharp cracks of igniters firing and check that both igniters are operating.

NOTE: Both igniters operating sound as non-synchronous, intermixed, irregular cracks occurring more than once per second and one igniter sounds as a single, synchronous crack about once per second.

F. Place AIR START switch off, IGNITION ON lights should go off and igniter firing should cease.

NOTE: Aircraft having ASC 170 (optional on Aircraft 1 - 147, 322 and 323) and Aircraft 148 - 200 continue with Steps G. thru M. below.

G. Pull NO. 2 IGNITION circuit breaker on side being checked (Pilot circuit breaker panel).

H. Place AIR START switch (for engine to be checked) on, IGNITION ON warning lights should come on.

I. Listen for one igniter firing.

J. Place AIR START switch off, IGNITION ON lights should go out and igniter should stop firing.

K. Depress NO. 2 IGNITION circuit breaker (for side being checked) and pull NO. 1 IGNITION circuit breaker.

L. Place AIR START switch (for engine to be checked) on, IGNITION ON lights should come on, and listen for single igniter firing.

M. Place AIR START switch off, depress NO. 1 IGNITION circuit breaker.

N. Repeat Steps A. through F. above (or M. as applicable) for second engine if both engines are to be checked at this time.

O. De-energize essential dc bus.

5. Ignition Firewall Feedthrough Connectors — Removal / Installation

WARNING: ELECTRICAL ENERGY STORED IN CONDENSER OF A HIGH ENERGY IGNITION UNIT IS POTENTIALLY LETHAL. ENSURE POWER WILL REMAIN OFF BEFORE TOUCHING UNIT, HIGH ENERGY LEAD OR IGNITER PLUG. DISCONNECT LOW ENERGY LEAD POWER SUPPLY FROM IGNITION UNIT AND WAIT FOR AT LEAST 1 MINUTE TO ALLOW STORED ENERGY TO DISSIPATE.

A. Removal

(1) Gain access to firewall feedthrough by opening cowling and nacelle access panels involved.

(2) Break safety wire and remove lead end fittings from feedthrough involved.

(3) Remove hardware and feedthrough connector.

B. Installation

(1) Install feedthrough connector through firewall and secure with removed hardware.

(2) Connect ignition cable fitting, torque to 216-240 inch pounds and safety.

(3) Inspect area for presents of foreign objects, security of all attachments.

(4) Close access panels and cowling previously opened.

(5) Perform Engine Ignition System - Aural Check, see this Section.

6. Engine Igniter Plugs — Maintenance Practices

Refer to Rolls-Royce Dart Maintenance Manual M-Da7-G.

LIST OF EFFECTIVE PAGES

CHAPTER 82

WATER INJECTION

* Pages revised by Revision 49
** Pages added by Revision 49
*** Pages deleted by Revision 49

TABLE OF CONTENTS

CHAPTER 82

WATER INJECTION

WATER/METHANOL INJECTION SYSTEM — DESCRIPTION / OPERATION

1. General

The water/methanol injection system contains and supplies all the water/methanol liquid mixture required by the two turboprop engines during wet takeoff or landing of the aircraft. Wet engine operation (water/methanol injection) restores power lost when engine fuel flow is trimmed by the pilot because of ambient air temperatures above ISA or dry power is decreased because of ambient temperatures below ISA. The allowable water/methanol injection range is ISA −15°C thru ISA +35°C (-8E and -8H engines) or ISA − 10°C thru ISA +35°C (-8X, -8Y and -8Z engines) from sea level to 10,000 feet. The water/methanol system uses a liquid mixture that is to Rolls-Royce Specification Type 1 water/methanol mixture AEP-1-W/M. Refer to Rolls-Royce Dart Maintenance Manual M-Da7-G, for additional information/requirements on water/methanol mixture.

Since the Gulfstream I installation may consist of Bladder Cells or Metal Tanks and will be identified throughout this chapter as follows:

- Aircraft 1 - 200 not having ASC 125. (Bladder cells)
- Aircraft 1 - 200 having ASC 125. (Metal tanks)

In utilizing this chapter, first ascertain which system is installed in the aircraft in question, then to the section applicable to the aircraft condition/malfunction. Refer to Table of Contents for page numbering and applicable sections. The following provides a break down of this chapter:

- Water/Methanol Injection System.
- Water/Methanol Cells/Tanks.
- Water/Methanol Injection Control System.
- Water/Methanol Injection System Boost Pumps.
- Water/Methanol Injection System Shutoff Valves.
- Water/Methanol Injection System Filters.
- Water/Methanol Quantity Indicating System.

NOTE: On aircraft 1 - 200, 322 and 323 with ASC 240 (G1C) excluding 116 and 027, utilize information pertaining to Aircraft not having ASC 125.

Aircraft 027 has Water/Methanol System installed by ASC 165.

Aircraft 116 has Water/Methanol System installed by ASC 125.

WARNING: WATER/METHANOL IS AN EXTREMELY VIOLENT POISON. IF IT IS INHALED OR SWALLOWED, OR ENTERS THE BODY BY ANY OTHER MEANS, IT WILL CAUSE BLINDNESS AND GENERAL PHYSICAL DECAY, WITH DEATH RESULTING EITHER SOON OR ULTIMATELY. IT CANNOT BE APPLIED EXTERNALLY WITHOUT CAUSING SERIOUS INJURY. AVOID INHALATION OF ITS VAPORS OR PROLONGED CONTACT WITH SKIN, AND PROMPTLY WASH OFF ANY SPILLED ON THE SKIN.

ANTIDOTE: IF ACCIDENTALLY TAKEN INTERNALLY:

A. GIVE FIRST AID AS FOLLOWS: IMMEDIATELY GIVE EMETIC OF MUSTARD IN MILK TO INDUCE VOMITING. GIVE MILK, WHITE OF EGG, OR FLOUR IN WATER AND PURGATIVE OF EPSOM SALT.

B. CALL PHYSICIAN AT ONCE.

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WATER/METHANOL INJECTION SYSTEM — DESCRIPTION / OPERATION

1. **General**

The water/methanol injection system contains and supplies all the water/methanol liquid mixture required by the two turboprop engines during wet takeoff or landing of the aircraft. Though two separate configurations of the water/methanol injection system are presently in use in aircraft and their installation varies accordingly. Basic description operation and maintenance are the same. Differences between the two configurations will be reflected accordingly throughout this section.

In utilizing this section, first ascertain whether the aircraft in question has ASC 125 installed or not, then follow the applicable parts of this section.

2. **Description**

The system (See Figure 1 thru Figure 7) consists of water/methanol (W/M) cells/tanks, a control system, quantity indicating system, filters, flow check valves, drain valves and restrictors (Aircraft 1 - 164, 322 and 323 not having ASC 176A). A feed line supplies water/methanol to both engines through the action of boost pumps in each cell/tank. Normally, the left cell/tank feeds the left engine, the right cell/tank feeds the right engine. In event of boost pump failure in one cell/tank, both engines are supplied from the other cell/tank by a crossfeed system which operates automatically.

Feed lines from the pumps are routed and connected to a T-fitting on each lower left nacelle structure. A low-pressure warning switch, is tapped into the main feed line. This switch senses a pressure drop in the system if it should occur and sends a signal to the water/methanol low-pressure warning lights (L or R W/M PRESS) in the master warning system. This switch is set at approximately 19.5 to 20.5 psi.

A check valve is installed in the main feed line downstream of the low-pressure warning switch. This check valve is installed to prevent the flow of water/methanol from one cell/tank to the other, also to prevent water/methanol from flowing into the cell/tank containing a failed pump. The pilot has an indication of which pump has failed by the W/M PRESS warning light on the master warning system.

A shutoff valve and filter is located in each nacelle. From the filter, the line continues to a firewall fitting which contains a restrictor plate for 529-8E and -8H engine applications only (Aircraft 1 - 164, 322 and 323 not having ASC 176A). The restrictor limits water/methanol flow in the event that the water/methanol control unit should jam fully open. This restrictor is removed in 529-8X engine applications (Aircraft - 164, 322 and 323 having ASC 176A and 165 - 200) and the 529-8X (uprated), -8Y and -8Z (Aircraft 1 - 200, 322 and 323 having ASC 246) as these engines requires higher water/methanol flows. From the firewall, the line continues to the water/methanol control unit on the engine. Aircraft 1 - 164, 322 and 323 not having ASC 178, a manually operated drain valve is installed in water/methanol line located in the aft main wheel well. Aircraft 1 - 163, 322 and 323 having ASC 178 and 164 - 200 the drain valve is located in the flap well at T-fitting connecting the crossfeed line (Aircraft not having ASC 125). or feed line (Aircraft having ASC 125).

When the system is electrically activated, water/methanol is pumped from either or both cells/tanks to the first stage compressor of either or both operating turboprop engines. Water/methanol should be armed below 14,000 rpm when the engines are operating at ambient air temperatures above ISA +15°C to restore engine takeoff power. Water/methanol injection is automatically controlled by a control (metering) unit in each engine supply line.

A. Aircraft not having ASC 125.

- On Aircraft 1 - 106 and 114 not having ASC 165, 2 cells are installed in each wing outboard of fuel tank. The total (See Figure 1) usable quantity of water/methanol mixture (62-64% water, 36-38% methanol by weight) carried in the cells is 84 gallons (660 pounds)
- On Aircraft 1 - 106 and 114 having ASC 165, and 107 - 200, 322 and 323, 1 cell is installed in each wing outboard of fuel tank. (See Figure 2). The total usable quantity of water/methanol mixture (62-64% water, 36-38% methanol by weight) carried in the cell is 46 gallons (331 pounds)

Weight stated in above paragraphs are based on the fact that the mixture weighs 7.85 pounds/gallon at a temperature of 60°F (15.6°C). Specific gravity of the mixture shall not be less than 0.9412 and not more than 0.9445 at this temperature. At other temperatures, specific gravity must be within chart limits provided in the Rolls-Royce Dart Maintenance Manual, M-Da7-G.

Components of the water/methanol system and their locations are as follows:

Unit	No. Per A/C	Location
W/M Cell	4	Two in each wing (A/C 1-106 & 114 not having ASC 165)
W/M Cell	2	One in each wing (A/C 1-106 & 114 having ASC 165, 107-200, 322 & 323)
Boost Pump	2	One in each cell (Inboard cell A/C 1-106 & 114 not having ASC 165)
Cell Quantity Transmitter (Float type)	2	One in each cell (Inboard cell A/C 1-106 & 114 not having ASC 165)
Vent Valve	2	One in each cell (Outboard cell A/C 1-106 & 114 not having ASC 165)
Filler Cap	2	One in each cell (Outboard cell A/C 1-106 & 114 not having ASC 165)
Low Pressure Switch	2	One located in each mid wing gap band (dry bay)
Flow Check Valve	2	One in each feed line, downstream of pressure sensing line, under mid-wing leading edge.
Crossflow Check Valve	2	Two parallel but opposing, in crossfeed lines in left flap well.
Shutoff Valve	2	One at left forward side of each main wheel well
Filter	2	One attached downstream of each shutoff valve
Drain valve	2	One in the aft end of each nacelle (A/C 1-163, 322 & 323 not having ASC 178)
Drain valve	2	One in the aft of each nacelle on wing rear beam (flap well) (A/C 1-163, 322 & 323 having ASC 178 & 164-200).
Restrictor	2	One in each firewall fitting (A/C 1-164, 322 & 323 not having ASC 176A or ASC 246).

B. Aircraft having ASC 125.

Aircraft 1 - 200, 322 and 323 one tank is mounted each side of mid fuselage in wing fillet (See Figure 3).

The total usable quantity of water/methanol (62-64% water, 36-38% methanol by weight) carried in the tanks is 47.4 gallons or 372 pounds. The weight stated above is based on the fact that mixture weighs 7.85 pounds/gallon at a temperature of 60°F (15.6°C). Specific gravity of mixture shall not be less than 0.9412 and not more than 0.9445 at this temperature. At other temperatures specific gravity must be within limits given on the curves in the Roll-Royce Dart Maintenance Manual M-Da7-G.

Components of the water/methanol system and their locations are as follows:

Unit	No. Per A/C	Location
W/M Tanks	2	One in each wing fillet area
Boost Pump	2	One in each tank
Tank Quantity Transmitter (Float type)	2	One in each tank
Vent Valve	2	One in each tank
Filler Cap	2	One in each fillet area
Low Pressure Switch	2	One located under each tank
Flow Check Valve	2	One in each feed line, downstream of pressure sensing line under each tank.
Crossflow Valve	2	Two parallel but opposing, in crossfeed line under right tank
Shutoff Valve	2	One at left forward side of each main wheel well
Filter	2	One attached downstream of each shutoff valve
Drain valve	2	One in the aft end of each nacelle (A/C 1-163, 322 & 323 not having ASC 178)
Drain valve	2	One in the aft of each nacelle on wing rear beam (flap well) (A/C 1-163, 322 & 323 having ASC 178 & 164-200).
Restrictor	2	One in each firewall fitting (Aircraft 1-164, 322 & 323 not having ASC 176A)

3. Operation

WARNING: AT LOW ATTITUDE AT TEMPERATURES BELOW ISA, THIS ENGINE MAY PRODUCE MORE POWER FOR TAKEOFF THAN THE AIRCRAFT HAS BEEN CERTIFIED FOR. UNDER THESE CONDITIONS, THE PLACARDED TORQUE METER LIMITATIONS (POSTED IN COCKPIT) SHOULD NEVER BE EXCEEDED. SEE ENGINE CALIBRATION CARD (POSTED IN COCKPIT) FOR INDIVIDUAL ENGINE LIMITATIONS.

CAUTION: IF SYSTEM IS ARMED WHEN THROTTLE LEVERS ARE ABOVE 14,700 RPM (AND ENGINE IS CALLING FOR RESTORATION), THERE WILL BE A SURGE OF W/M INTO THE ENGINE WITH SUBSEQUENT OVERTORQUING, THEREFORE IT IS RECOMMENDED THAT ARMING SHOULD BE DONE AT 14,000 RPM OR BELOW TO PREVENT THE POSSIBILITY OF ENGINE DAMAGE OR FAILURE.

NOTE: Minimum temperature approved for operation with water/methanol is ISA −15°C (-8E and -8H engines) or ISA −10°C (-8X, -8Y and -8Z engines).

When the WATER/METHANOL switch on the right side of the cockpit lower overhead panel is depressed to ARMED, the system is electrically energized to supply water/methanol from cells/tanks to the operating engine(s) under control of the water/methanol control (metering) unit in each engine supply line. The control unit, sensitive to power lever position, propeller torque and ambient air pressure, automatically activates the

supply system when engine operating conditions are consistent with unit settings. When the system is energized and activated.

A. An electric motor operated, positive displacement boost pump inside each cell/tank pumps water/methanol (at 21 to 24 psi and at a rate of approximately 720 gph maximum) through its respective engine supply line (See Figure 3 and Figure 4). As described above, each supply line contains the following components.

   (1) Water/methanol low-pressure warning switch (set at 19.5 to 20.5 psi and connected to the master warning system).

   (2) Check valve (prevents reverse flow to cell/tank).

   (3) Drain valve (manually operated).

   (4) Shutoff gate valve (electric motor operated).

   (5) Filter (attached to shutoff gate valve).

   (6) Restrictor (orifice at fire wall limits water/methanol supply to engine to a maximum flow of 9.2 US gpm) (Aircraft 1-164, 322 and 323 not having ASC 176A or ASC 246).

   (7) Control unit (unit mounted on the forward left side of each engine).

B. The electrically operated shutoff gate valve in each engine supply line is opened.

C. W/M PRESS warning lights on the master warning system blink and then remain out.

D. An inoperative boost pump is indicated by a lighted L W/M PRESS or R W/M PRESS warning light on the master warning system.

E. The left or right shutoff gate valve closes automatically if the cockpit respective HP FUEL COCK lever is placed in the FEATHER position or the cockpit eyebrow panel respective FIRE PULL T-handle is pulled out.

4. System Indicating

Water/methanol quantity in each cell/tank is measured and transmitted by a float-actuated quantity transmitter (potentiometer-transmitter) mounted inside each cell. Liquid level in the cell, mechanically measured by the float, is electrically transmitted to and indicated by FULL, 3/4, 1/2, 1/4, EMPTY on a dual indicator dial on the cockpit center instrument panel, when the battery switch is placed in NORM position. See Section 82-6-0 for additional information regarding water/methanol quantity indicating system.

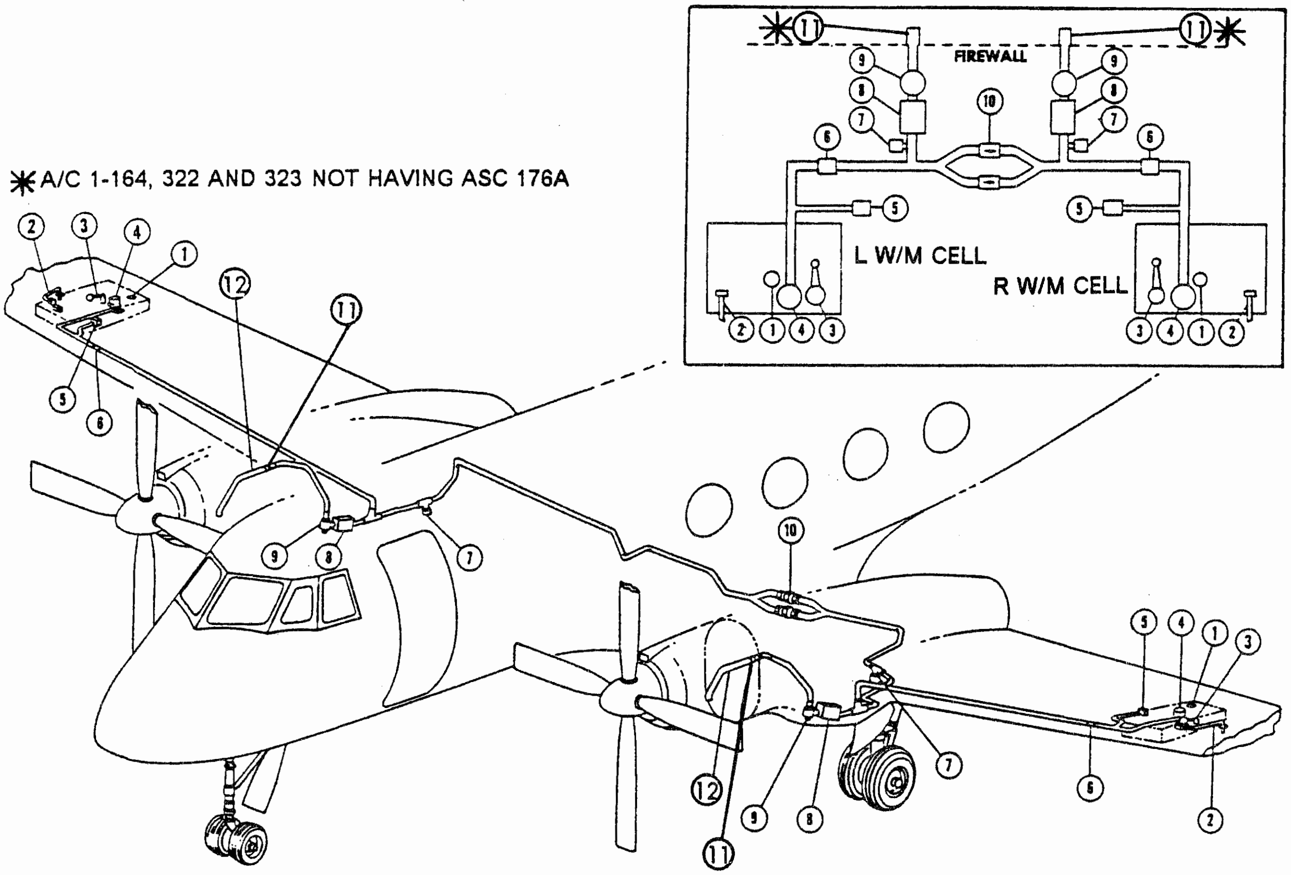
5. System Draining and Servicing

A manually operated drain valve located in each aft nacelle permits system drainage. Either valve drains both cells/tanks if system is electrically ARMED. Dumping in flight cannot be accomplished. The cells/tanks are gravity filled through a filler cap in each cell/tank. When filling, check that only water/methanol cells are being filled. Filler caps are painted RED, GRAY, RED. Check specific gravity of water/methanol mixture being added, (Refer to Rolls-Royce Dart Maintenance Manual, M-Da7-G).

WARNING: METHANOL VAPOR IS POISONOUS. DO NOT INHALE. ALL SPILLED FLUID ON SKIN OR AIRCRAFT SHOULD BE FLUSHED IMMEDIATELY WITH FRESH WATER.

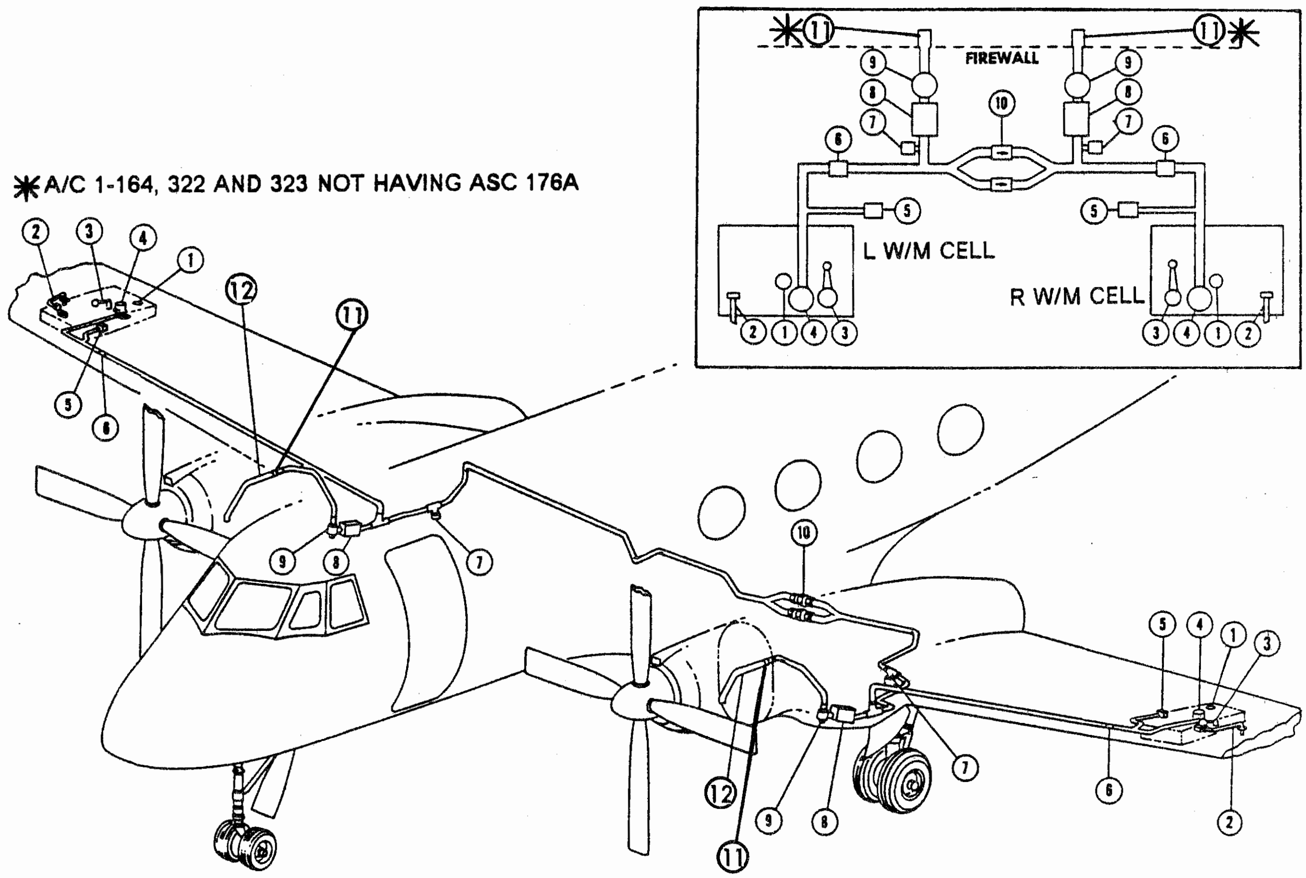
CAUTION: IT IS RECOMMENDED THAT THE WATER/METHANOL FLUID COMPLETELY COVER THE BOOST PUMPS AT ALL TIMES (AT LEAST 1/2 FULL), SINCE METALS WHICH HAVE BEEN IMMERSED IN WATER/METHANOL BECOME HIGHLY SUSCEPTIBLE TO CORROSION WHEN EXPOSED TO AIR.

NOTE: The fluid in the cells/tanks can be fully utilized if necessary, however, the 1/2 full level should be replenished within 12 hours.



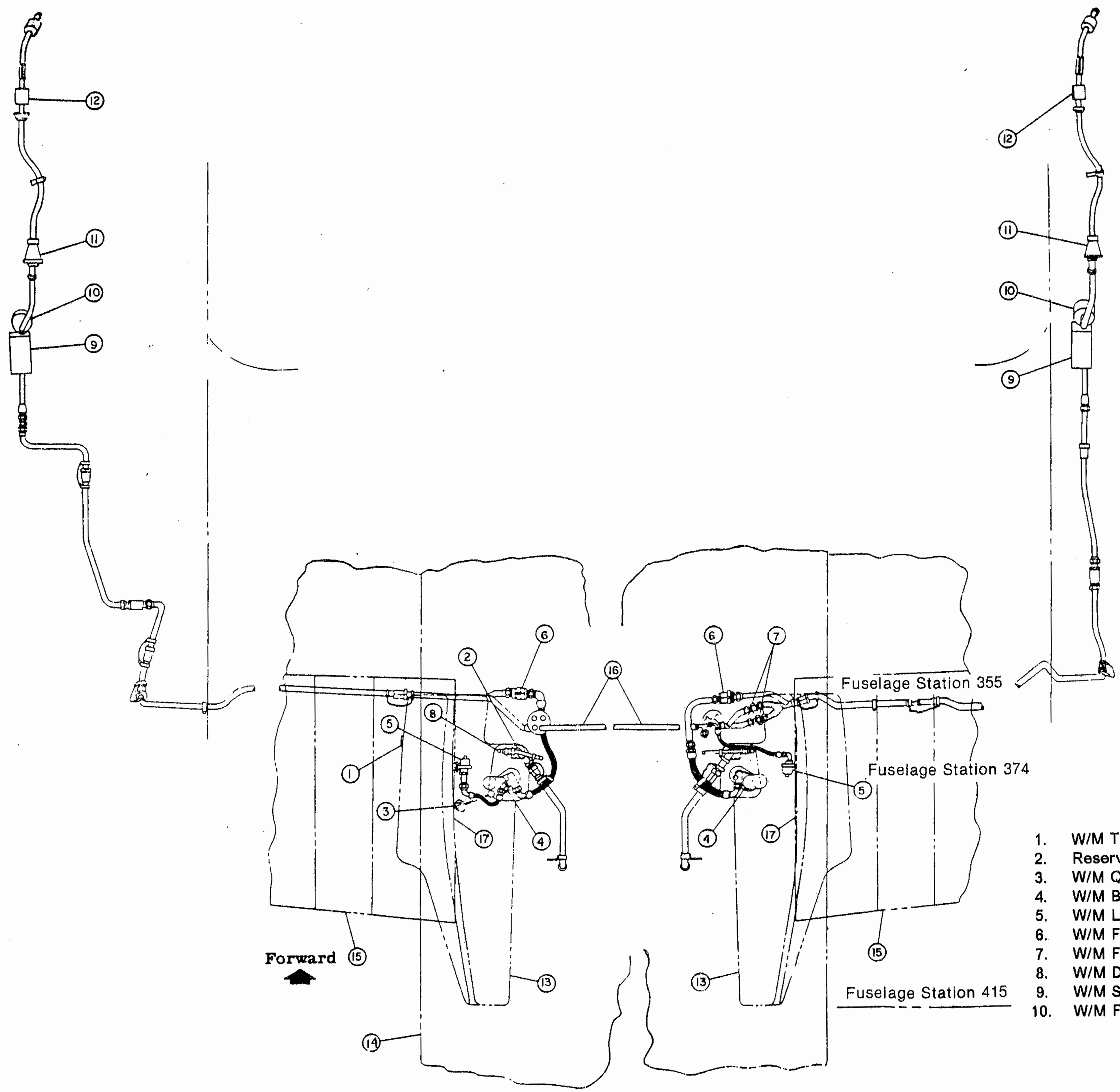
1. Filler Cap
2. Water/Methanol Cell Vent
3. Water/Methanol Quantity Transmitter
4. Boost Pump
5. Low Pressure Switch
6. Flow Check Valve
7. Drain Valve
8. Shut-off Valve
9. Filter
10. Cross-flow Check Valve
11. Restrictor *
12. Engine Line/Control Unit

Water/Methanol Injection System
(Aircraft 1 - 106 and 114 not having ASC 165)
Figure 1.



1. Filler Cap
2. Water/Methanol Cell Vent
3. Water/Methanol Quantity Transmitter
4. Boost Pump
5. Low Pressure Switch
6. Flow Check Valve
7. Drain Valve
8. Shut-off Valve
9. Filter
10. Cross-flow Check Valve
11. Restrictor *
12. Engine Line/Control Unit

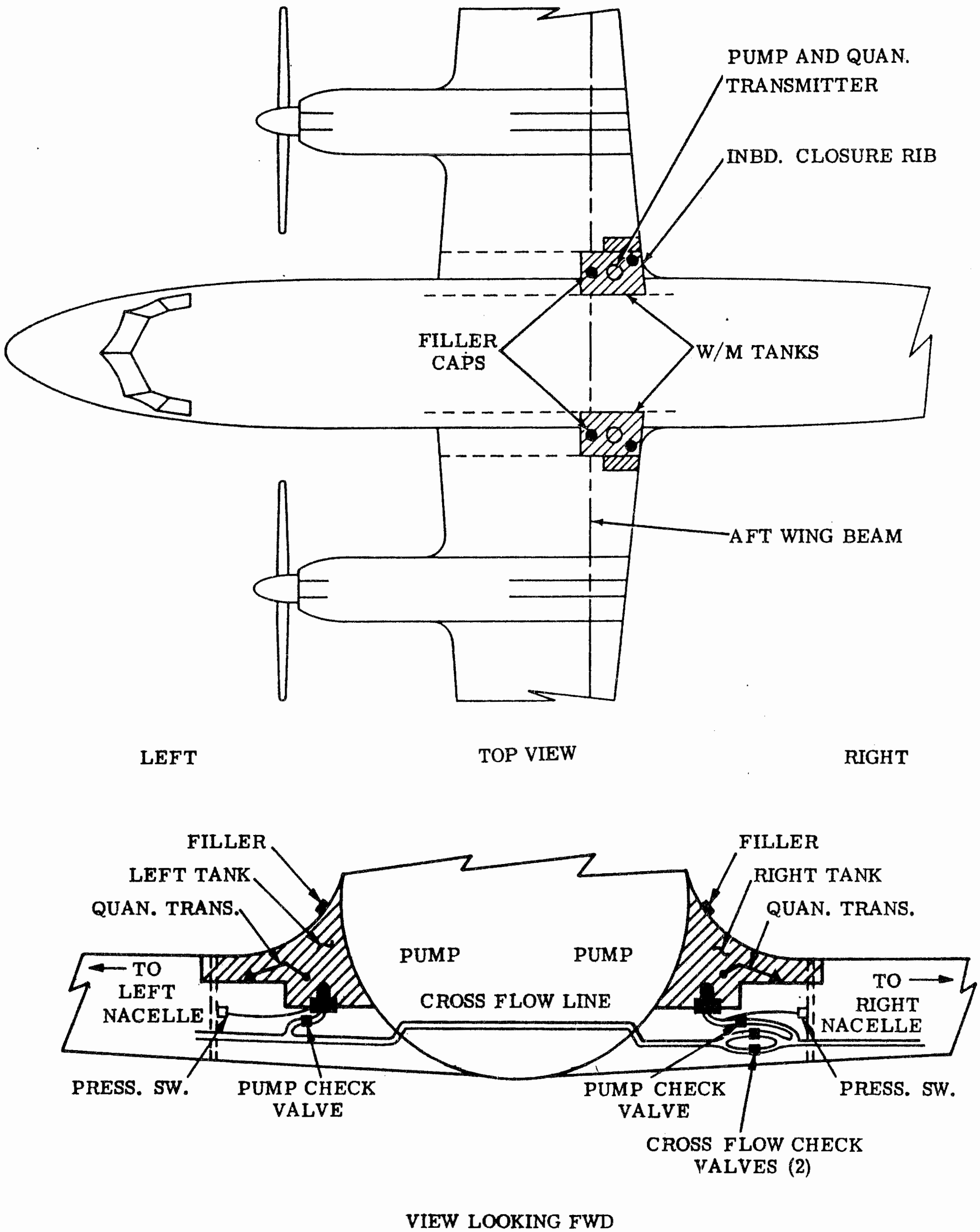
Water/Methanol Injection System
(Aircraft 1 - 106 having ASC 165 and 107 - 200, 322 and 323)
Figure 2.



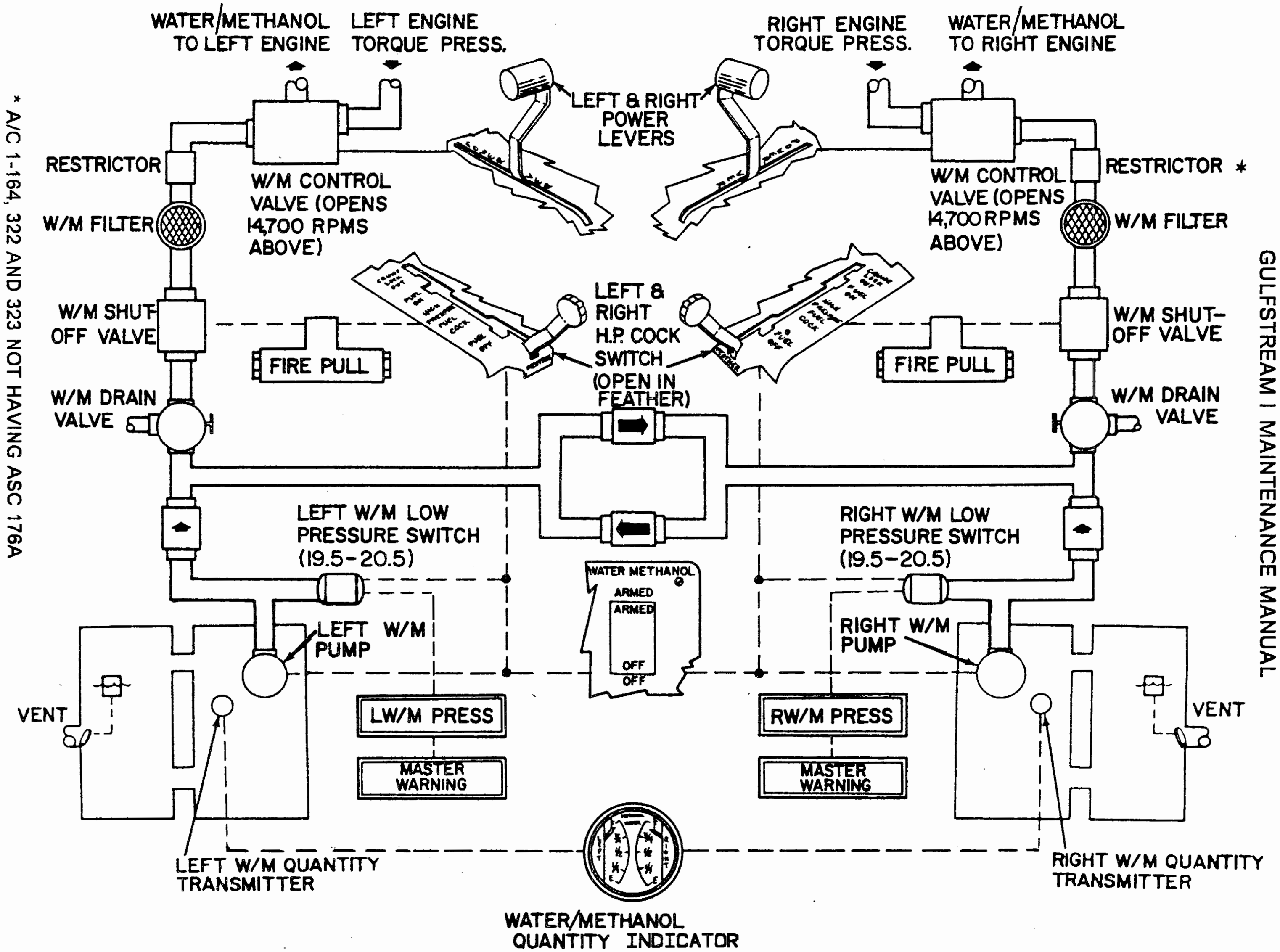
1. W/M Tank Reservoir Filler Cap
2. Reservoir Tank Vent
3. W/M Quantity Tank Transmitter
4. W/M Boost Pump
5. W/M Low Pressure Switch
6. W/M Flow Check Valve (Engine Supply Line)
7. W/M Flow Check Valve (Crossfeed Line)
8. W/M Drain Valve (Engine Supply Line)
9. W/M Shutoff Valve (Engine Supply Line)
10. W/M Filter (Attached to Shutoff Valve)
11. W/M Restrictor *
12. W/M Flow Control Unit (FCU)
13. W/M Tank Reservoir (Integral)
14. Fuselage
15. Wing Trailing Edge
16. W/M Crossfeed Line
17. Wing Inboard Closure Rib

* With 529 – 8E and 8H Engines Only

Water/Methanol Injection System
(Aircraft 1 - 200, 322 and 323 having ASC 125)
Figure 3.

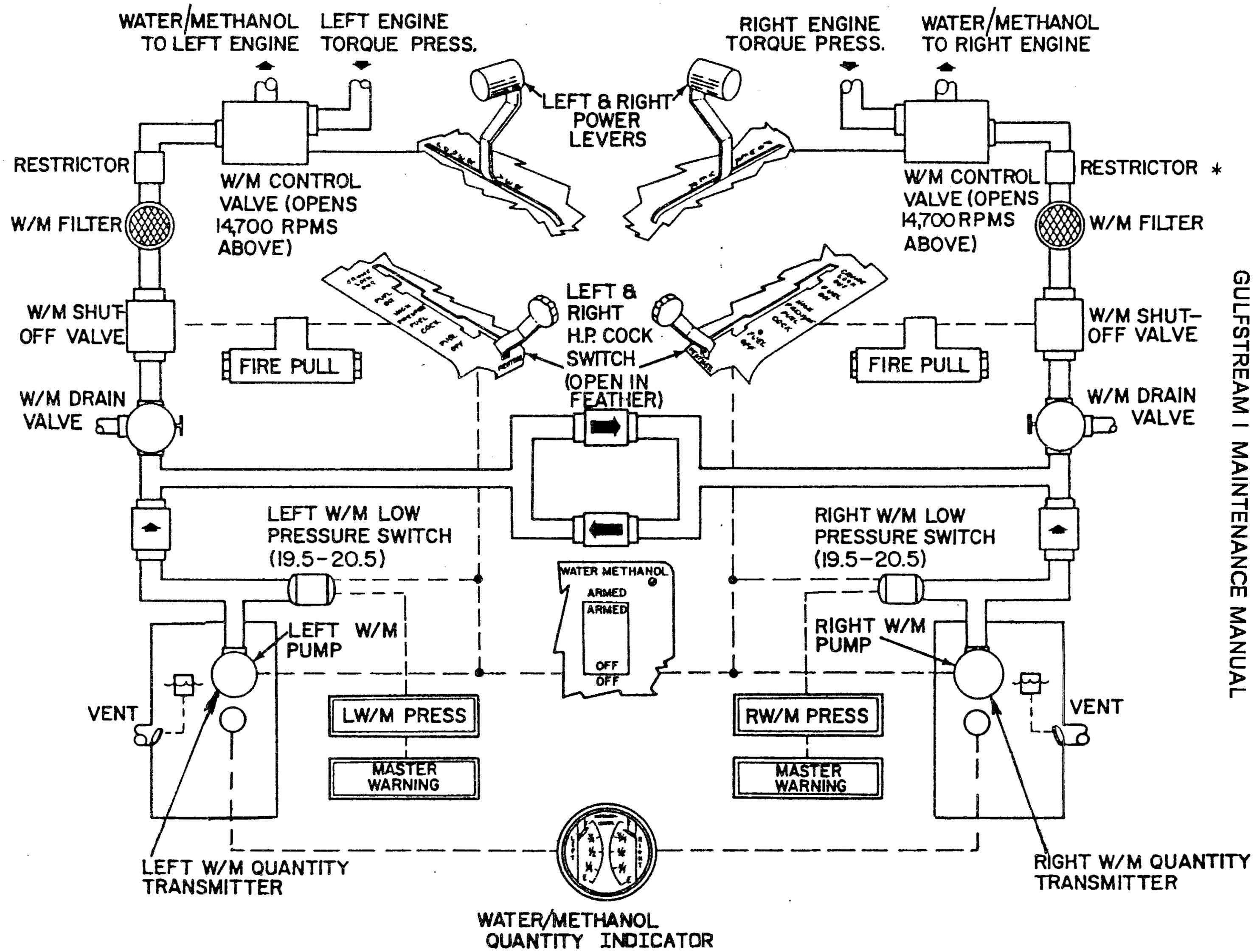


Water/Methanol System — Component Location (ASC 125)
Figure 4.



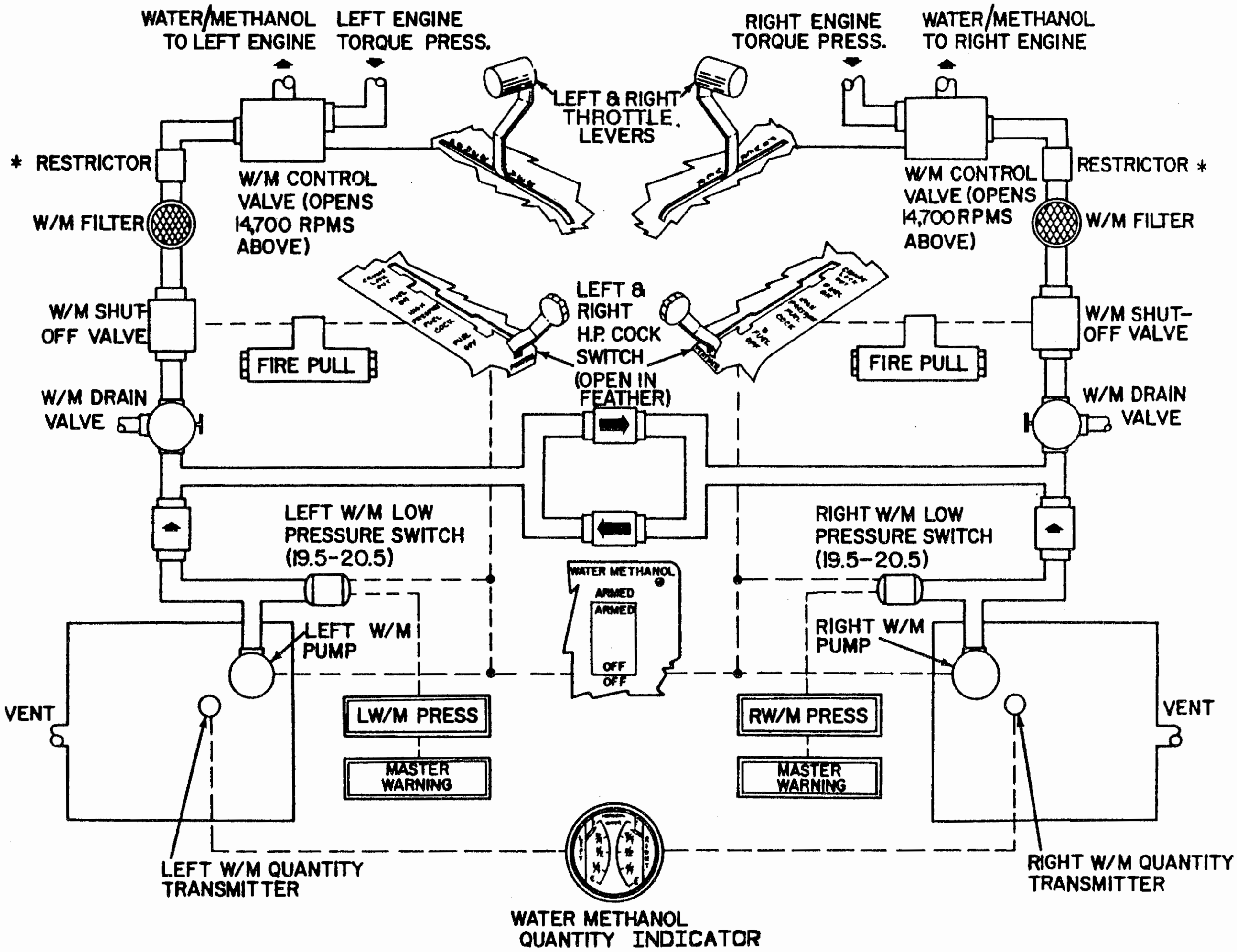
* A/C 1-164, 322 AND 323 NOT HAVING ASC 176A

Water/Methanol Injection System — Block Diagram
(Aircraft 1 - 106 and 114 not having ASC 165)
Figure 5.



* A/C 1-164, 322 AND 323 NOT HAVING ASC 176A

Water/Methanol Injection System — Block Diagram
(Aircraft 1 - 106 and 114 having ASC 165 and 107 - 200, 322 and 323)
Figure 6.



* A/C 1-164, 322 AND 323 NOT HAVING ASC 176A

Water/Methanol Injection System — Block Diagram
(Aircraft having ASC 125)
Figure 7.

WATER/METHANOL INJECTION SYSTEM — MAINTENANCE PRACTICES

1. Water/Methanol Injection System — Operational Test

NOTE: The following should be accomplished with water/methanol tanks less than 1/2 full.

A. LH System

(1) Pull L W/M PUMP circuit breaker.

(2) Place WATER/METHANOL switch in ARMED position. If system is functioning correctly, the LH water/methanol low-pressure warning light should come ON and there should be no transfer of fluid.

B. RH System

(1) Pull R W/M PUMP circuit breaker.

CAUTION: IF WARNING LIGHT COMES ON, THEN OFF, OR IF TRANSFER OF WATER METHANOL FLUID OCCURS, STOP THE CHECK IMMEDIATELY AND TROUBLE SHOOT SYSTEM.

(2) Place WATER/METHANOL switch in ARMED position. If system is functioning correctly, RH water/methanol low-pressure warning light should come ON and there should not be any transfer of fluid.

2. Water/Methanol System — Draining / Servicing

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Draining

(1) Gain access to drain valve as follows, (See Figure 201).

(a) On Aircraft 1 - 163, 322 and 323 not having ASC 178, access by removing panels lower aft nacelle. (LH, 29-JP-1 and RH, 30-LH-6).

(b) On Aircraft 1 - 163, 322 and 323 having ASC 178, and Aircraft 164 - 200, access by extending flaps (valve is on face of rear wing beam aft of each nacelle in flap well).

(2) Open main landing gear clamshell doors and install ground safety struts.

(3) Attach suitable length of hose to each drain valve.

(4) Attach suitable length of hose to each water/methanol filter drain fitting.

NOTE: If only one cell/tank is to be drained, attach drain hoses to opposite side drain valve and same side filter.

(5) Secure drain hoses into a clean container with appropriate capacity. Ensure hose end can be viewed.

(6) Open water/methanol filler caps so cell can be viewed during draining.

(7) Ensure both HP FUEL COCK levers are in FUEL OFF.

(8) Ensure POWER LEVERS are in IDLE.

(9) Apply electrical power to aircraft main and essential dc bus.

NOTE: If one cell/tank is to be drained, pull W/M PUMP circuit breaker for opposite cell/tank.

(10) Place WATER/METHANOL switch to ARMED, L/R W/M PRESS warning lights should come on and go off.

(11) Open drain valves to remove water/methanol from system.

CAUTION: DO NOT ALLOW PUMP TO RUN DRY.

(12) Place WATER/METHANOL switch to OFF as soon as cell/tank is empty. Install filler caps.

(13) Remove electrical power from aircraft.

(14) Remove hoses from drain valves, filters and install caps on filter when draining is complete.

NOTE: Preserve pumps if system is to be dry for 12 hours or more, see Water/Methanol Boost Pump - Preservation/Depreservation, Section 82-3-0.

B. Servicing

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

CAUTION: IF CELLS/TANKS NEED REFILLING, CARE SHOULD BE TAKEN NOT TO DRAG HOSE ACROSS WING AS DAMAGE TO DE-ICER BOOTS AND AIRCRAFT PAINT COULD RESULT. IT IS RECOMMENDED HOSE BE BROUGHT OVER TRAILING EDGE OF WING RATHER THAN LEADING EDGE.

(1) Check quantity in cells/tanks using cockpit indicator.

(2) Remove electrical power from aircraft.

(3) Ensure adequate supply of water/methanol mixture is available. See Rolls-Royce Dart Maintenance Manual M-Da7-G for water/methanol mixture requirements.

NOTE: Water methanol mixture remaining in aircraft cells/tanks may have become contaminated and contamination may be of a large enough quantity to affect specific gravity of water methanol mixture therefore, a specific gravity check of mixture should be made before servicing. See Water/Methanol Mixture - Field Test, this Section.

(4) Ensure valves and opened lines are closed.

CAUTION: BECAUSE OF LOCATION OF FUEL AND WATER METHANOL FILLER CAPS, BE CAREFUL NOT TO INADVERTENTLY PUT FUEL INTO WATER/METHANOL CELLS. ENGINE WILL BE OVERTORQUED IF FUEL IS USED INSTEAD OF WATER/METHANOL.

(5) Open filler caps.

CAUTION: IF CELLS/TANKS NEED REFILLING, CARE SHOULD BE TAKEN NOT TO DRAG HOSE ACROSS WING AS DAMAGE TO DE-ICER BOOTS AND AIRCRAFT PAINT COULD RESULT. IT IS RECOMMENDED HOSE BE BROUGHT OVER TRAILING EDGE OF WING RATHER THAN LEADING EDGE.

(6) Ground aircraft and ground water/methanol hose nozzle to aircraft structure through grounding jacks located inboard of filler caps.

CAUTION: OVERFLOWING CELLS/TANKS CREATE A SERIOUS FIRE HAZARD AND MUST BE AVOIDED. DAMAGE TO DE-ICER BOOTS COULD ALSO RESULT. SPILLED WATER METHANOL MUST BE WIPED UP IMMEDIATELY.

(7) Service cells/tanks approved mixture without overflowing.

NOTE: If maintenance was performed on system, fill cells/tanks 1/4 to 1/2 full. This will allow appropriate system tests to be performed. After tests are completed fill water/methanol cells/tanks.

(8) Remove service hose and ground cables from aircraft.

(9) Ensure filler caps are securely fastened/locked.

3. Water/Methanol Mixture — Field Test

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

NOTE: For information on specifications, preparation and storage, determination of non-volatile solids content of water, filtration and determination of silica content of water, refer to Rolls-Royce Dart Maintenance Manual M-Da7-G.

A. Drain a small quantity of water/methanol mixture into a clean glass container. Mixture can be obtained through filler cap or drain valve.

B. Ensure mixture meets the following requirements:

(1) Mixture shall be clear and free of sediment or suspended matter.

(2) Specific gravity of water/methanol mixture at 60°F (15.6°C) shall be not less than 0.9412 (18.8° API) and not more than 0.9445 (18.3° API). At mixture temperatures other than 60°F (15.6°C), specific gravity must be within the chart limits provided in the Rolls-Royce Dart Maintenance Manual M-Da7-G.

4. Water/Methanol System — Bleeding

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Gain access to drain valve as follows: (See Figure 201).

(1) On Aircraft 1 - 163, 322 and 323 not having ASC 178 access by removing panels lower aft nacelle (LH, 29-JP-1 and RH, 30-LH-6).

(2) On Aircraft 1 - 163, 322 and 323 having ASC 178, and Aircraft 164 - 200, access by extending flaps (valve is on rear wing beam aft of each nacelle in flap well).

B. Open main landing gear clamshell doors and install ground safety struts.

C. Ensure water/methanol cells/tanks are 1/2 to 3/4 full.

D. Attach suitable length of hose to each drain valve.

E. Attach suitable length of hose to each water/methanol filter drain fitting.

NOTE: If only one cell/tank is to be bled, attach drain hoses to opposite side drain valve and same side filter.

F. Secure drain hoses into a clean container with a 5 gallon or greater capacity. Ensure hose end can be viewed.

G. Pull L NAC FIRE EXT and R NAC FIRE EXT circuit breakers (Aircraft 1 - 200, 322 and 323 not having ASC 210) or FIRE EXT SHOT #1 and FIRE EXT SHOT #2 circuit breakers (Aircraft 1 - 200, 322 and 323 having ASC 210).

H. Ensure both HP FUEL COCK levers are in FUEL OFF.

I. Ensure both POWER LEVERS are in IDLE.

J. Apply electrical power to aircraft main and essential dc bus.

K. Pull both FIRE PULL T-handles.

L. Place WATER/METHANOL switch to ARMED, L/R W/M PRESS warning lights should come on and go off.

M. Individually open each drain valve to bleed air from system, close drain valve after air is removed.

N. Individually reset FIRE PULL T-handles to open shutoff valves. Pull FIRE PULL T-handles out when all air is bled off.

O. Place WATER/METHANOL switch to OFF

P. Reset FIRE PULL T-handles to normal.

Q. Remove electrical power from aircraft.

R. Depress circuit breakers pulled in Step G. above.

S. Remove hoses from drain valves and filters. Install caps on filter and tighten securely.

T. Inspect area for presents of foreign objects, damage, leaks and security of all attachments.

U. Install access panels or retract flaps.

V. Service water/methanol cells/tanks to full, see Water/Methanol System - Draining/Servicing, this Section.

W. Remove ground safety struts.

5. Water/Methanol System Check Valve — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Gain access to check valve as follows: (See Figure 202).

(a) Aircraft 1 - 200, 322 and 323 not having ASC 125, remove mid and outboard wing leading edges.

(b) Aircraft 1 - 200, 322 and 323 having ASC 125, remove landing light cover access panel 107-LH FIL-2 or 107-RH FIL-2, as required.

(2) Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, this Section.

(3) Loosen B-nuts until nuts are free of check valve.

(4) Remove check valve and note direction of flow arrow.

NOTE: If pump is to remain out of aircraft for periods in excess of twelve hours, the boost pump must be preserved, see Water/Methanol Pump - Preservation/Depreservation, Section 82-3-0.

B. Installation

(1) Install check valve with free flow arrow in proper direction.

(2) Ensure drain valve is closed.

(3) Service water/methanol cell/tank 1/4 to 1/2 full, see Water/Methanol System - Draining/Servicing, this Section.

(4) Bleed water/methanol system, inspect cells and pumps for leaks, see Water/Methanol System - Bleeding, this Section.

(5) Perform Water/Methanol System Check Valve - Operational Test and inspect for leaks during test, see this Section.

(6) Inspect area for presents of foreign objects, damage and security of all attachments.

(7) Install access panels as follows:

(a) On aircraft not having ASC 125, ensuring deice hoses are properly installed, install wing leading edges.

(b) On Aircraft having ASC 125, install landing light access panel.

(8) Service water/methanol cells/tanks to full, see Water/Methanol System - Draining/Servicing, this Section.

6. Water/Methanol System Check Valve — Operational Test

CAUTION: DO NOT PERFORM TEST WITH FULL WATER/METHANOL CELLS.

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Ensure water/methanol cells are 1/4 to 1/2 full.

B. Ensure POWER LEVERS are in IDLE.

C. Energize main and essential dc bus, pull L W/M PUMP circuit breaker.

CAUTION: IF WARNING LIGHT COMES ON AND THEN GOES OFF, OR IF TRANSFER OF FLUID OCCURS IN STEP D. OR F. BELOW, STOP CHECK IMMEDIATELY AND TROUBLE SHOOT SYSTEM.

D. Place WATER/METHANOL switch to ARMED, L W/M PRESS warning light must come on with no transfer of fluid (observe quantity indicator).

E. Depress L W/M PUMP circuit breaker, L W/M PRESS warning light should go out.

F. Pull R W/M PUMP circuit breaker, R W/M PRESS warning light must come with no transfer of fluid (observe quantity indicator).

G. Depress R W/M PUMP circuit breaker, R W/M PRESS warning light should go out.

H. Place WATER/METHANOL switch OFF.

I. De-energize main and essential dc bus.

7. Water/Methanol Drain Valve — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Place WATER/METHANOL switch OFF and remove electrical power.

(2) Gain access to drain valve as follows: (See Figure 201).

(a) On Aircraft 1 - 163, 322 and 323 not having ASC 178 access by removing panels lower aft nacelle. (LH, 29-JP-1 and RH, 30-LH-6 panels)

(b) On Aircraft 1 - 163, 322 and 323 having ASC 178, and Aircraft 164 - 200, access by extending flaps. Valve is on rear wing beam aft of each nacelle.

(3) Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, this Section.

(4) Remove drain valve.

NOTE: If pump is to remain out of aircraft for periods in excess of twelve hours, the boost pump must be preserved, see Water/Methanol Pump - Preservation/Depreservation, Section 82-3-0.

B. Installation

(1) Install new O-ring.

(2) Install drain valve and tighten jam nut.

(3) Service water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, this Section.

(4) Bleed water/methanol system, inspect valve for leaks, see Water/Methanol System - Bleeding, this Section.

(5) Inspect area for foreign objects, damage and security of all attachments.

(6) Install access panels or retract flaps.

8. Water/Methanol Cell/Tank Pump — Inspection

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, this Section.

B. Gain access to pump as follows: (See Figure 201 or Figure 202).

(1) Aircraft 1 - 200, 322 and 323 not having ASC 125, remove upper access panels 21/22-UPR-2/-3 and lower access panels 21/22-LWR-1. On Aircraft 1 - 106 and 114, remove upper access panel 21/22-UPR-4.

(2) Aircraft 1 - 200, 322 and 323 having ASC 125, remove landing light cover access panel 107-LH FIL-2 or 107-RH FIL-2, as required. Tank side panel may be removed, if desired.

C. Remove water/methanol boost pumps, see Water/Methanol Pump - Removal/Installation, Section 82-3-0.

NOTE: Preserve pumps if system is to be dry for 12 hours or more, see Water/Methanol Boost Pump - Preservation/Depreservation, Section 82-3-0.

D. Flush cells/tanks with fresh water and wipe away residue accumulated in cells/tanks.

E. Inspect cells/tanks, pumps and plumbing for cracks, foreign objects, evidence of corrosion, overheating and general condition. Inspect all electrical connections. Clean or replace all corroded components (See Figure 202).

NOTE: Before installing a pump that has been preserved, depreserve pump, see Water/Methanol Pump - Preservation/Depreservation, Section 82-3-0.

F. Install water/methanol pumps, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

G. Inspect area for presents of foreign objects, damage and security of all attachments.

(1) On Aircraft not having ASC 125, install upper wing access panels.

(2) On Aircraft having ASC 125, install tank side panel, if removed.

H. Ensure drain valve is closed.

I. Service water/methanol cells 1/4 to 1/2 full, see Water/Methanol System - Draining/Servicing, this Section.

J. Bleed water/methanol system, inspect cells and pumps for leaks, see Water/Methanol System - Bleeding, this Section.

K. Apply electrical power to aircraft and select WATER/METHANOL switch to ARMED.

L. Ensure L/R W/M PRESS warning light comes on momentarily and goes out.

M. Inspect cells/tanks and pumps for leaks.

N. Set WATER/METHANOL switch to OFF and remove electrical power.

O. Inspect area for presents of foreign objects, damage and security of all attachments.

P. Install access panels.

Q. Service water/methanol cells to full, see Water/Methanol System - Draining/Servicing, this Section.

9. Water/Methanol Lines — Pressure Test / Inspection

A. Remove water/methanol shutoff valve box covers

B. Pull L NAC FIRE EXT and R NAC FIRE EXT circuit breakers (Aircraft 1 - 200, 322 and 323 not having ASC 210) or FIRE EXT SHOT #1 and FIRE EXT SHOT #2 circuit breakers (Aircraft 1 - 200, 322 and 323 having ASC 210).

C. Apply electrical power to aircraft and pull FIRE PULL T-handles.

D. Lower flaps to TAKE OFF and open landing gear doors. Install ground safety struts.

E. Remove electrical power from aircraft and pull L/R W/M CONT circuit breakers.

F. With shutoff valve closed, disconnect electrical connector.

G. Gain access to pump as follows (See Figure 203 and Figure 204).

(1) Aircraft 1 - 200, 322 and 323 not having ASC 125, remove upper access panels 21/22-UPR-1/-2 access panels.

(2) Aircraft 1 - 200, 322 and 323 having ASC 125, remove landing light cover access panel 107-LH FIL-2 or 107-RH FIL-2, as required.

H. On Aircraft not having ASC 125, remove mid and outboard wing leading edges to gain access to water/methanol lines.

I. Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, this Section.

NOTE: If pump is to remain out of aircraft for periods in excess of twelve hours, the boost pump must be preserved, see Water/Methanol Pump - Preservation/Depreservation, Section 82-3-0.

J. Remove water/methanol pump, see Water/Methanol Pump - Removal/Installation, Section 82-3-0.

K. Remove check valves, see Water/Methanol Check Valve - Removal/Installation, this Section.

L. Inspect check valves,for corrosion, flapper separation and general condition.

M. Inspect inside of both lines for corrosion.

N. Install check valves with flapper hinge at top, see Water/Methanol Check Valve - Removal/Installation, this Section.

O. With pump removed, accomplish following pressure check:

   (1) Disconnect low pressure switch sensing line and cap line.

   (2) Place a suitable reducer, such as an AN919-21D, in each of the disconnected pump lines.

   (3) Attach a standard pressure regulator to reducer at left cell/tank.

   (4) Connect pressure source to regulator with a capacity in excess of 80 psi.

   (5) Pressurize lines to 80 $\pm$ 10 psi for 5 minutes.

   NOTE: There may be a slight leakage from open line at right cell/tank. Check valve is not a zero leak type. If leak rate exceeds 20 cc per minute, check valve should be removed and serviced in accordance with required inspection.

   (6) While pressure is applied, inspect entire length of water/methanol lines in leading edge or wing fillet, nacelle, flap well and in cells/tanks for leaks. If FLR-15 is removed, inspect crossfeed lines for leaks.

      (a) Any leakage at a connection will be opened for a detailed inspection.

      (b) Any leakage through a tube wall will be cause for replacement of that tube.

   (7) Repeat Steps O. (1) thru (6) for right side.

P. Install water/methanol pumps and connect lines, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0. Leave lower access panels off.

Q. Uncap and reconnect low pressure switch sensing line.

R. Service water/methanol cells/tanks to at least half full to submerge pumps and inspect for leaks during servicing, see Water/Methanol System - Draining/Servicing, this Section.

S. Apply electrical power to aircraft and reset FIRE PULL T-handles. Remove electrical power.

   NOTE: Inspect for leaks during bleeding.

T. Perform Water/Methanol System - Bleeding, see Water/Methanol System - Bleeding, this Section.

U. Service water/methanol cells/tanks to full, see Water/Methanol System - Draining/Servicing, this Section.

V. Inspect entire system for leaks.

W. Inspect area for foreign objects, damage and security of all attachments.

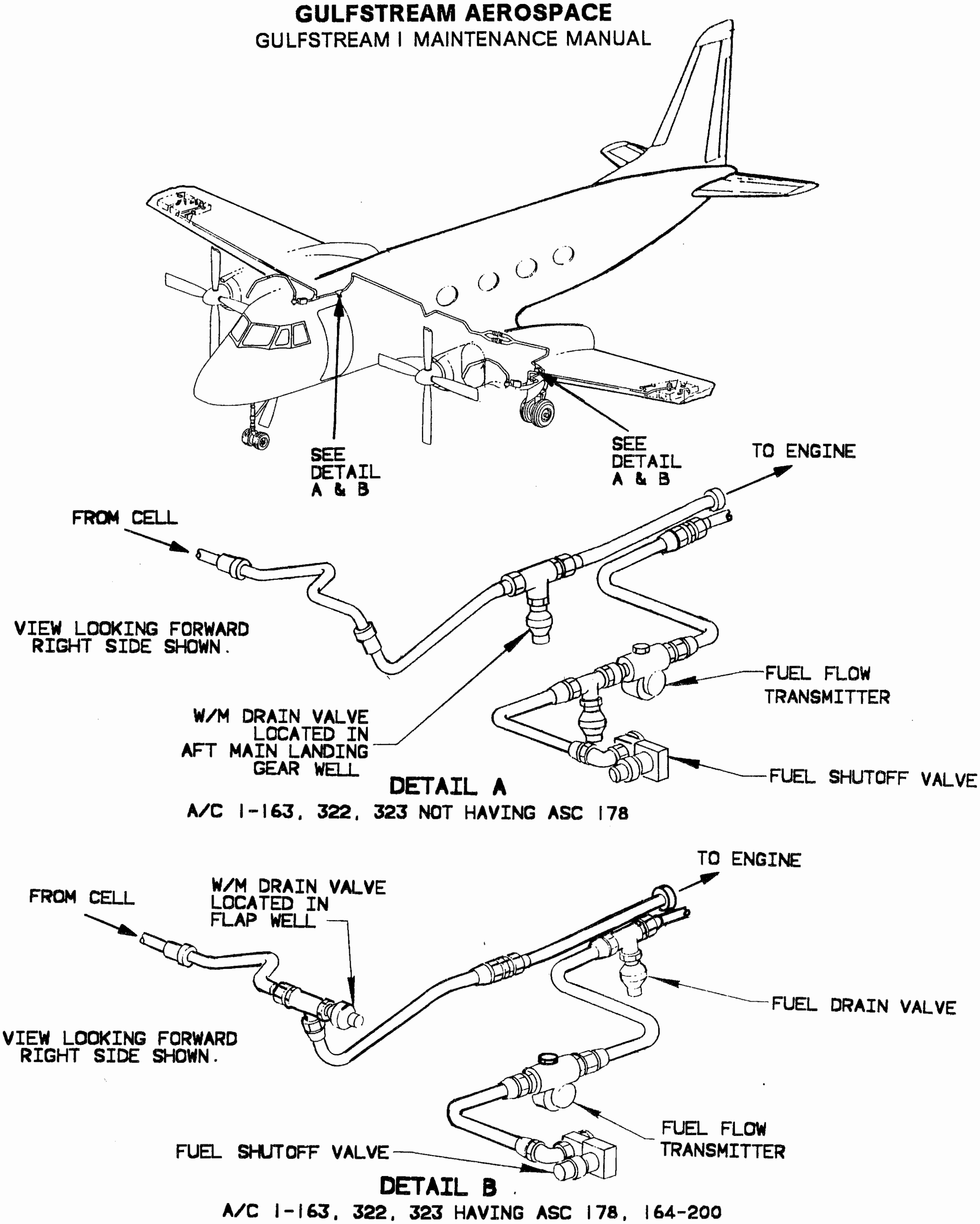
X. Install access panels as follows:

   (1) On aircraft not having ASC 125, install access panels 21/22-UPR-1 and 21/22-UPR-2, wing leading edges and water/methanol shutoff valve box covers.

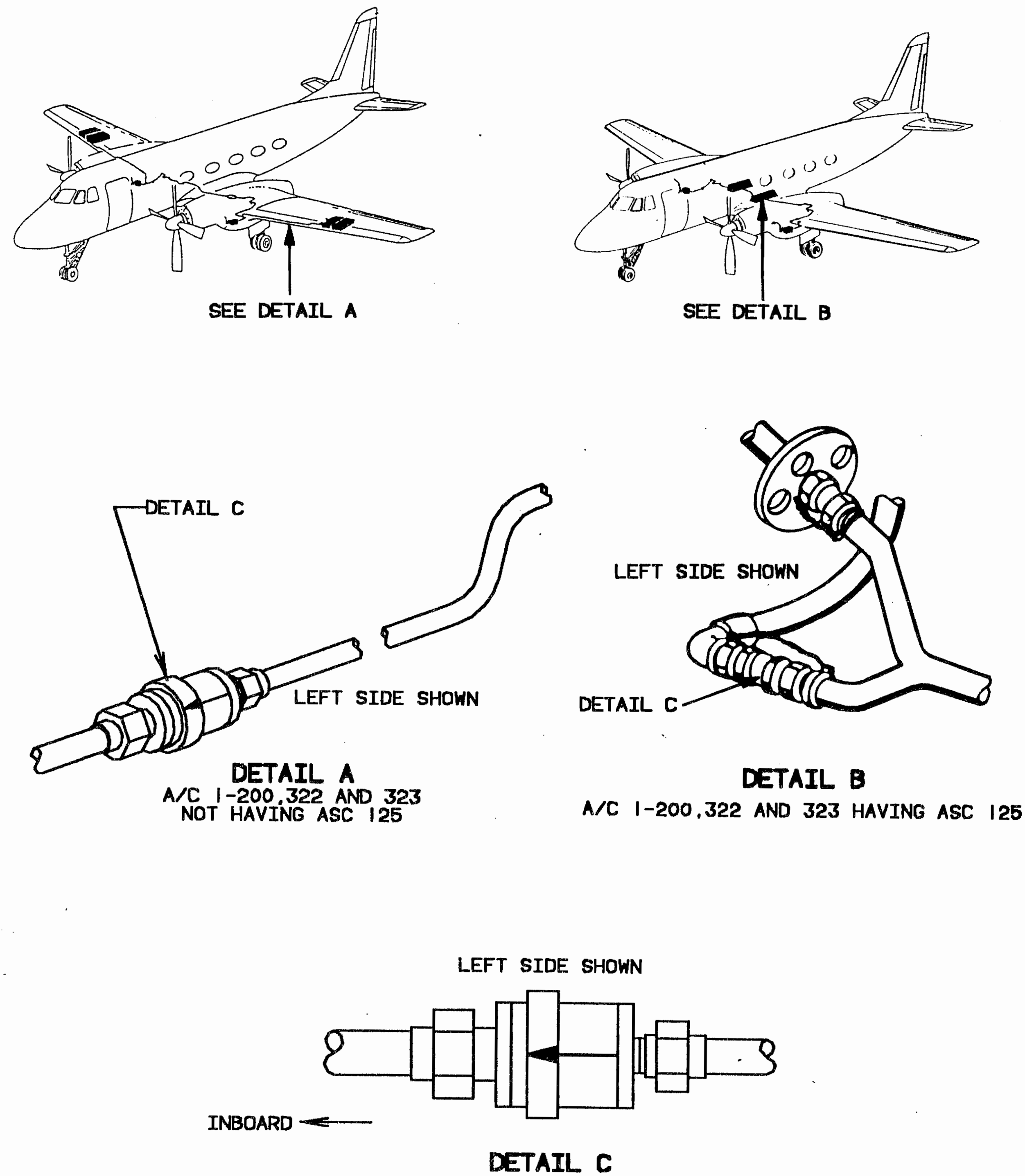
   (2) On Aircraft having ASC 125, install landing light access panel and water/methanol shutoff valve box covers.

Y. Depress circuit breakers pulled in Step B. above.

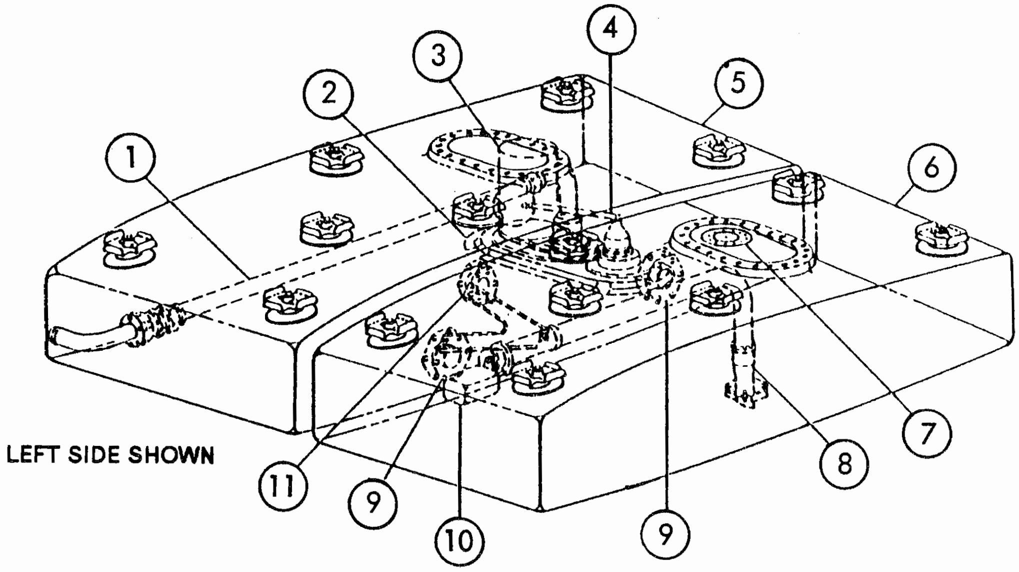
Z. Retract flaps and remove ground safety struts.



Water/Methanol Drain Valve
Figure 201.



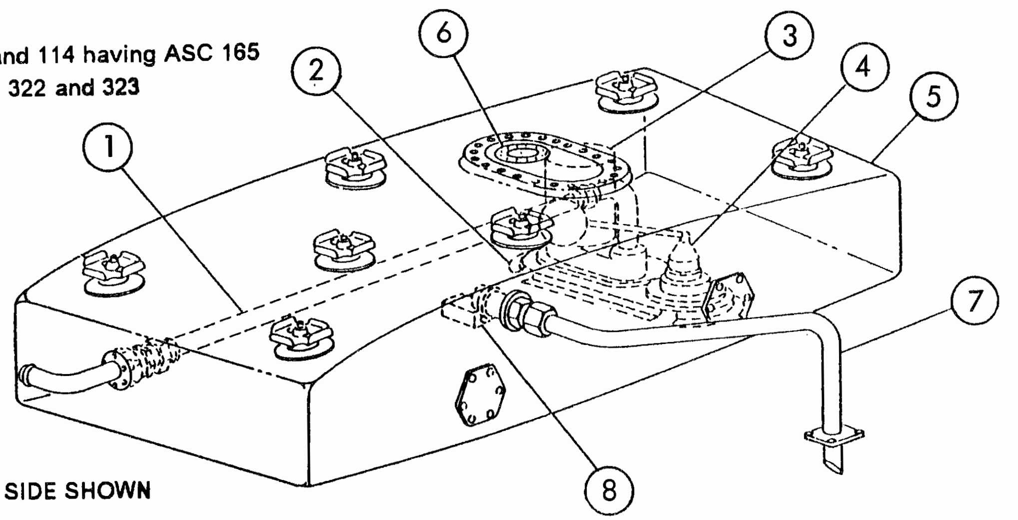
Water/Methanol Check Valve
Figure 202.



1. Water/Methanol Feed Line
2. Mounting Plate
3. Boost Pump
4. Quantity Transmitter
5. Inboard Bladder Cell
6. Outboard Bladeer Cell
7. Filler Cap
8. Vent Tube
9. Water/Methanol Interconnector
10. Vent Valve
11. Vent Interconnector

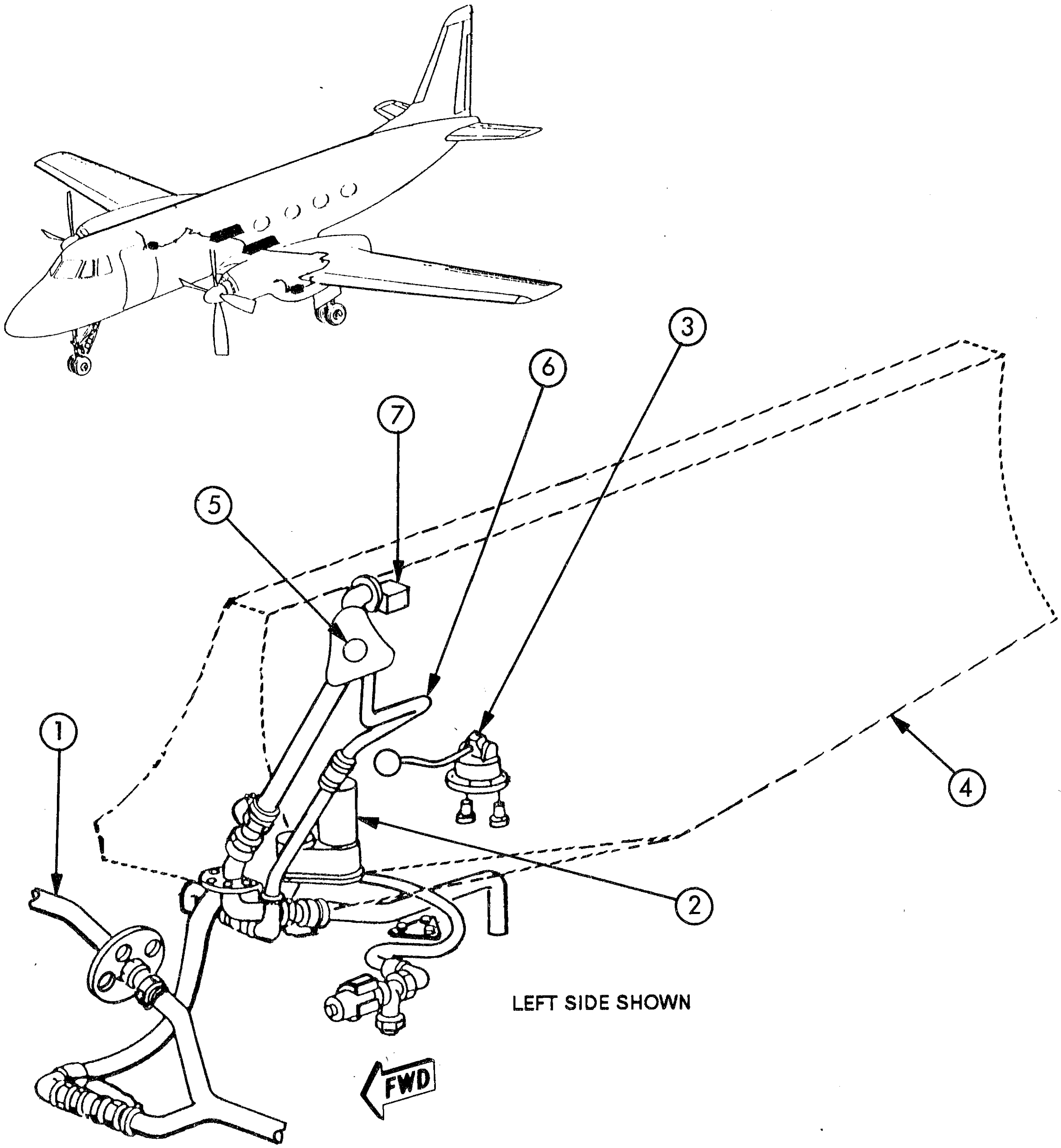
A/C 1-106, and 114 not having ASC 165

A/C 1-106, and 114 having ASC 165
and 107-200, 322 and 323



1. Water/Methanol Feed Line
2. Mounting Plate
3. Boost Pump
4. Quantity Transmitter
5. Bladeer Cell
6. Filler Cap
7. Vent Tube
8. Vent Valve

Water/Methanol Cell
(Aircraft 1 - 200, 322 and 323 not having ASC 125)
Figure 203.



1. Water/Methanol Feed Line
2. Boost Pump
3. Quantity Transmitter
4. Metal Tank
5. Filler Cap
6. Vent Tube
7. Vent Valve

Water/Methanol Injection Tank
(Aircraft Having ASC 125)
Figure 204.

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WATER/METHANOL CELLS/TANKS — DESCRIPTION / OPERATION

1. General

The water/methanol cell/tank system provides suitable storage facilities for the water/methanol carried aboard the aircraft. Though three separate cell/tank configurations are presently in use in the aircraft and their installation varies accordingly, basic description, operation and maintenance are similar. Differences between the three configurations will be reflected accordingly throughout this Section.

In utilizing this section, first determine which system is installed in the aircraft in question, and follow the applicable parts of this section.

WARNING: TO MINIMIZE FIRE HAZARDS, ENSURE THAT GROUNDING JACKS ARE CONNECTED BEFORE FILLING OR DRAINING.

WATER/METHANOL IS POISONOUS. USE IN WELL VENTILATED AREA. AVOID BREATHING VAPORS AND CONTACT WITH SKIN. IF SPILLED ON SKIN OR AIRCRAFT FLUSH IMMEDIATELY WITH FRESH WATER.

NOTE: If water/methanol system is to be inactive or if aircraft is to be laid up for a period in excess of 12 hours, it is recommended that the water/methanol fluid completely cover the boost pumps to prevent corrosion. The cells should be 1/2 full in order to cover the pumps. If the fluid is to be drained to facilitate work within the cells which requires more than 12 hours to complete, the pumps must be preserved. See Water/Methanol Pump - Preservation/Depreservation, Section 82-3-0.

2. Description

A. Bladder Cells (Aircraft not having ASC 125)

WARNING: BECAUSE OF THE LOCATION OF THE FUEL AND WATER/METHANOL FILLER CAPS, BE CAREFUL NOT TO INADVERTENTLY PUT FUEL IN THE WATER/METHANOL CELLS. THE ENGINE WILL BE OVERTORQUED IF FUEL, INSTEAD OF WATER/METHANOL, ENTERS ENGINE.

The left and right wing water/methanol cells consists of bladder type cells installed in the outer panels between front and rear beams. Two types of installations, i.e., single or dual cells per wing, are utilized (See Figure 1 and Figure 2). Differences between these installations are provided in later paragraphs. The water/methanol cells, vented to the atmosphere while the aircraft is at rest on the ground, are slightly pressurized by ram air during flight. The ram vent line has a 30° scarf cut to give a greater amount of ram air to water/methanol cells than fuel cells. This ram air provides pressure in the cells to prevent their collapse. Filling of the water/methanol cells is accomplished by the standard gravity method through a filler neck in each cell, which has a red, grey, red filler cap. The fuel cell grounding jack is used when filling the water/methanol cells The cells are held in the cavity by snap fasteners.

(1) Aircraft 1 - 106 and 114 not having ASC 165

These aircraft have two cells per wing, the inboard cell is located between Fuselage Station 314 1/4 and 331, and outboard cell is located between Fuselage Station 332 and 348 1/2. Each wing has a capacity of 42 gallons (330 pounds). The cells are joined together by leakproof interconnector fittings (two at the bottom, one at the top). The inboard cell is provided with an access cover in the upper and lower skins. The outboard cell has one access cover and water/methanol filler cap located in the upper skin. The non-icing ram vent tube, installed in the outboard cell, is equipped with a float actuated valve to prevent spillage during operation. A boost pump and quantity transmitter mounting base is attached to the lower skin of the inboard cell.

(2) Aircraft 1 - 106 and 114 having ASC 165 and 107 - 200, 322 and 323.

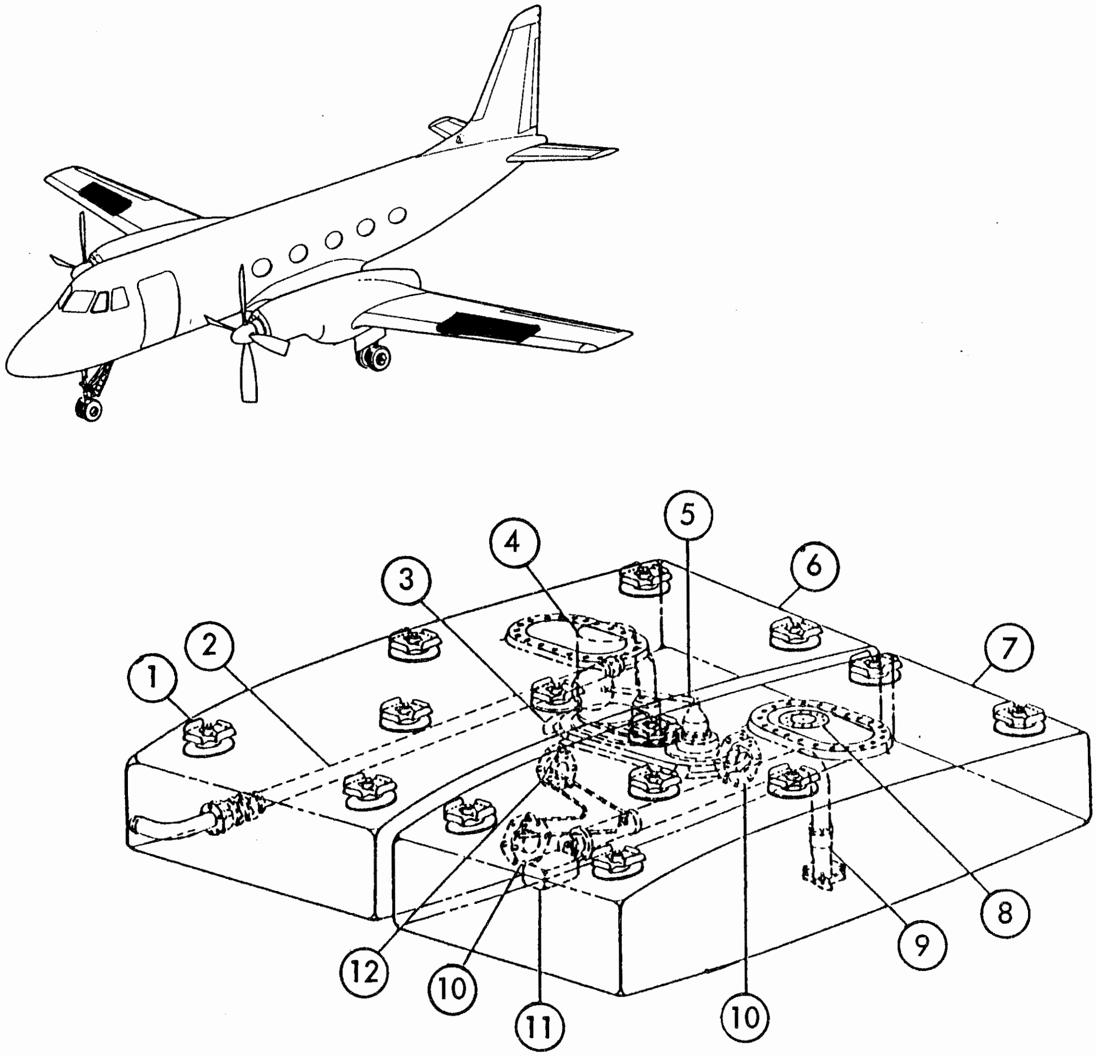
These aircraft have one cell per wing, the cell is located between Fuselage Station 313 1/4 and 331. Each cell has a capacity of 23 gallons (180 pounds). Each cell is provided with an access cover in the lower skin, and one access cover and water/methanol filler cap located in the upper skin. The non-icing ram vent tube is equipped with a float actuated valve to prevent spillage during

operation. A boost pump and quantity transmitter mounting base is attached to the lower skin of the cell.

B. Metal Tanks (Aircraft having ASC 125)

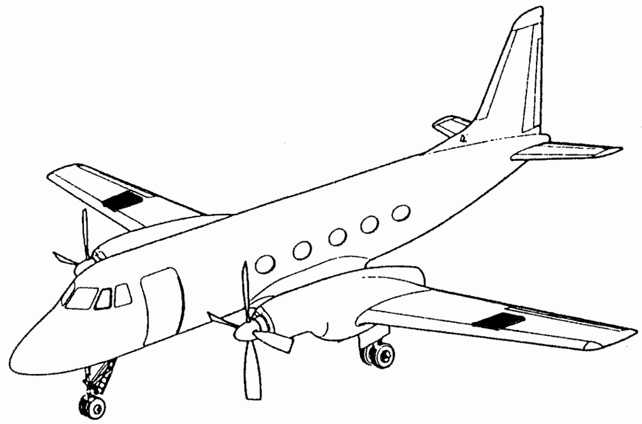
There are two integral aluminum reservoir tanks one located inside each aft wing/fuselage fillet between Fuselage Station 355 and 415 (See Figure 3). Each tank has a capacity of 23.7 U.S. gallons (186 pounds), thus the total capacity of the system is 47.4 U.S. gallons (372 pounds). The feed lines extend from the tanks outboard along the rear beam to the inboard end of the nacelles. Filler caps on these tanks are located in the fillet area, approximately over the rear wing beam and are painted red, grey, red.

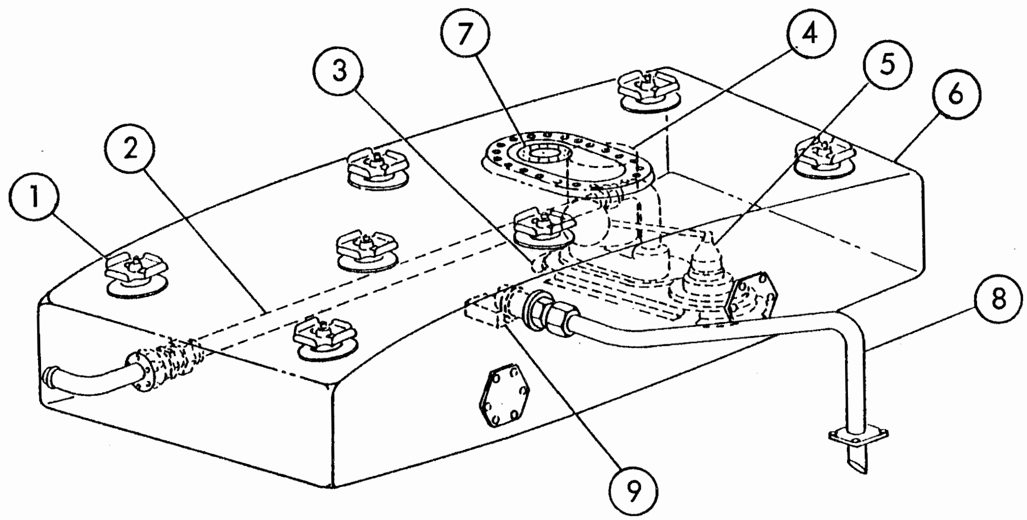
The tanks are vented to the atmosphere while the aircraft is on the ground. A vent pipe inside each tank vents the tank at all times to the bottom of each fuselage wing fillet at Fuselage Station 368. A float valve capping each vent pipe, prevents water/methanol discharge overboard upon aircraft maneuvering. Since bladder type cells are not used, scarf cut vents are not necessary. The boost pumps are located in the sump of each tank, aft of the rear wing beam and forward of each landing light. All feed lines and crossfeed lines are inboard of the nacelles. The crossflow check valves are inboard under the right pump. The low pressure warning switch is mounted on the inboard surface of the flap closure rib under each tank.



1. Snap Fastener
2. Water/Methanol Feed Line
3. Mounting Plate
4. Boost Pump
5. Quantity Transmitter
6. Inboard Bladder Cell
7. Outboard Bladeer Cell
8. Filler Cap
9. Vent Tube
10. Water/Methanol Interconnector
11. Vent Valve
12. Vent Interconnector

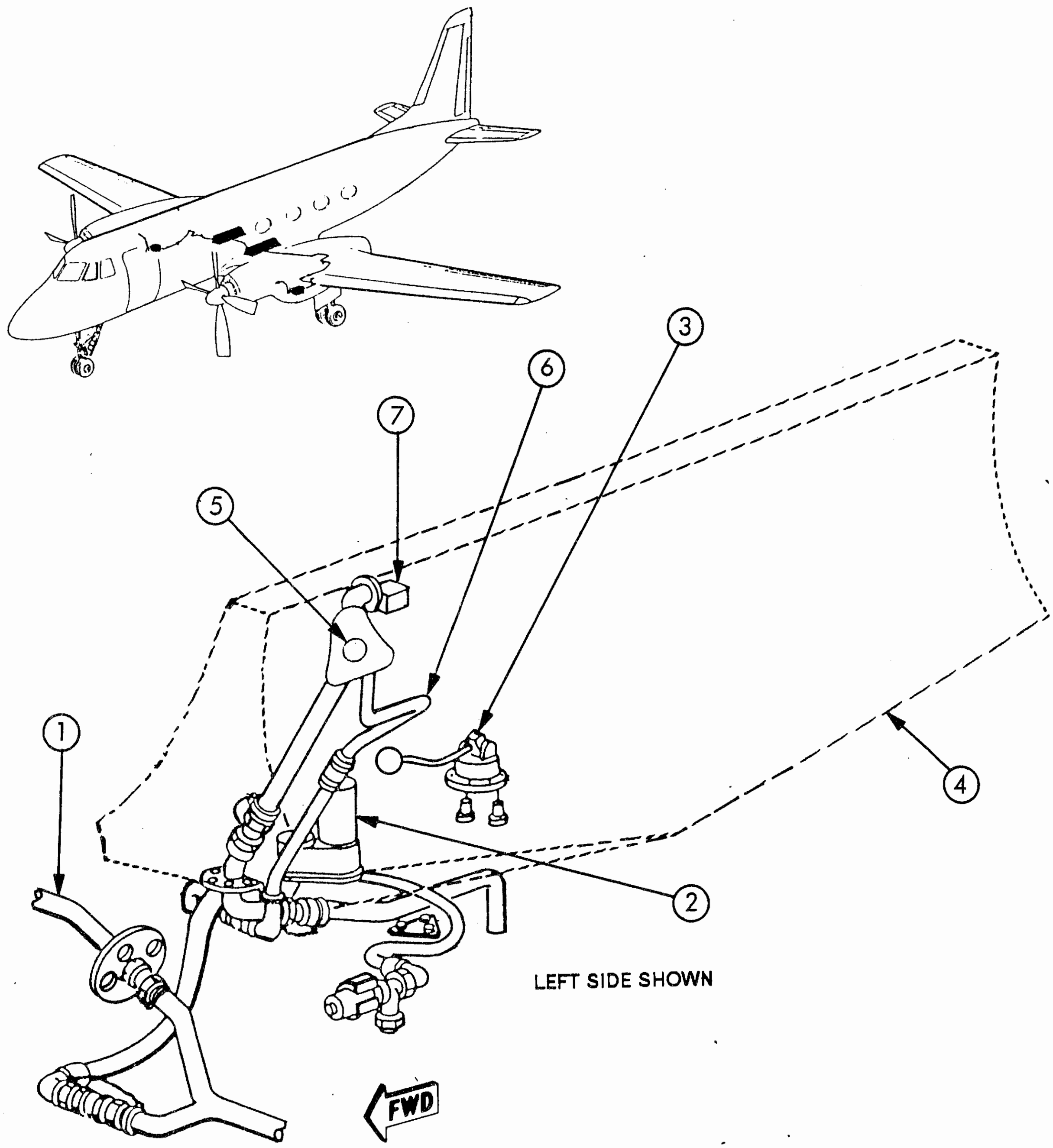
Water/Methanol Cell Installation
(Aircraft 1-106 and 114 not having ASC 165)
Figure 1.





1. Snap Fastener
2. Water/Methanol Feed Line
3. Mounting Plate
4. Boost Pump
5. Quantity Transmitter
6. Bladder Cell
7. Filler Cap
8. Vent Tube
9. Vent Valve

Water/Methanol Cell Installation
(Aircraft 1-106 and 114 having ASC 165 and 107-200, 322 and 323)
Figure 2.



1. Water/Methanol Feed Line
2. Boost Pump
3. Quantity Transmitter
4. Metal Tank
5. Filler Cap
6. Vent Tube
7. Vent Valve

Water/Methanol Tank Installation
(Aircraft having ASC 125)
Figure 3.

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WATER/METHANOL CELLS/TANKS — MAINTENANCE PRACTICES

1. Water/Methanol Cells — General Handling Precautions

NOTE: Additional handling procedures, as well as procedures for storage and repair of water/methanol bladder cells, are contained in Uniroyal Corporation Report FC-1473-73, Third Revision, June 1979, available from the Uniroyal Corporation, Engineered Systems Department, Mishawaka, Indiana 46544.

A. Keep cell in packaged carton until all preceding installation procedures have been accomplished.

B. Avoid contact with sharp or abrasive objects such as sharp edges of metal, metal shavings, concrete floors, etc.

C. Ensure outer wing panel water/methanol cell cavity is clean and free from sharp edges, metal chips and shavings.

D. Thoroughly clean and inspect inside of cavity before installing cells.

E. Store cells in a cool dry place.

F. Handle the cells carefully in temperatures below 4.5°C (40°F). In areas where temperatures below 4.5°C (40°F) exist, provisions should be made to heat the aircraft wing. The upper temperature limit is 71.1 °C (160°F).

CAUTION: AVOID SPILLING HYDROLUBE, LACQUER THINNER OR PAINT REMOVER ON CELLS. IF SPILLAGE OCCURS, WIPE CELLS CLEAN AS SOON AS POSSIBLE. ANY CELL THAT MAY HAVE BEEN WET WITH THESE FLUIDS FOR A PROLONGED PERIOD OF TIME SHOULD BE REMOVED FROM THE AIRCRAFT AND EXAMINED FOR DETERIORATION.

2. Water/Methanol Cell — Packaging

A. Pull cell out to full length and force the sides of cell in toward center, making cell as flat as possible.

B. Fold cell in middle, with wax paper between all surfaces, bringing ends together, then roll cell into tight bundle and tape in place.

C. Wrap cell in wax paper and then in soft padded material.

D. Place cell in a cardboard container small enough to keep cell from moving in carton when sealed. If carton is too large, protect cell by adding enough packing to eliminate movement.

E. The carton shall be strong enough to withstand normal handling by shipping companies.

F. The carton shall be stenciled or marked "HANDLE WITH CARE", "USE NO HOOKS", and "DO NOT STAPLE". Part number and drawing number should be stenciled on the outside.

3. Water/Methanol Filler Cap O-ring — Removal / Installation

A. Removal

(1) Remove electrical power from aircraft.

(2) Remove cap from aircraft fillerneck access panel.

(3) Seal filler cap access panel with tape or install spare filler cap.

(4) Remove pin from cap handle. (See Figure 201)

(5) Disassemble locking device; remove O-rings from cap assembly.

B. Installation

(1) Install new O-rings; assemble cap and install pin through handle.

(2) Install cap on aircraft fillerneck access panel.

4. Water/Methanol Cell — Removal / Installation

NOTE: This procedure applicable to Aircraft 1 - 106 and 114 having ASC 165 and 107 - 200, 322 and 323.

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Remove electrical power from aircraft.

(2) Drain water/methanol cells, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(3) Remove access panels 21/22-UPR-3 and 21/22-LWR-1.

(4) Remove outboard wing leading edge.

(5) Remove residual water/methanol from cell with lint free cloths, using proper precautions to prevent contact with skin.

(6) Remove vent and tubing from cell, see Water/Methanol Vent Valve - Removal/Installation, this Section.

(7) Working inside cell, disconnect feed line from pump and front beam connection; remove feed line through upper access hole (See Figure 202).

(8) Remove water/methanol boost pump, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

(9) Remove quantity transmitter, see Water/Methanol Quantity Transmitter - Removal/Installation, Section 82-6-0.

(10) Disconnect feed line Wiggins fitting from elbow on forward face of front beam.

(11) Remove outlet connection through upper access hole.

(12) Working at lower access hole, remove pump/quantity transmitter flange fitting.

(13) Disconnect snap hangers holding top portion of cell to wing upper skin.

(14) Collapse cell uniformly and roll lengthwise.

(15) Remove cell through upper access hole.

(16) Discard O-rings and gaskets.

(17) Inspect wing cavity for damage to liners. Repair or replace liners as required.

B. Installation

NOTE: Inspect all components for damage and/or corrosion prior to installation.

(1) Roll cell lengthwise and insert through access hole in upper wing skin.

CAUTION: WHEN INSTALLING CELLS AVOID CONTACT WITH SHARP OR ABRASIVE OBJECTS. ENSURE WING CELL CAVITY AREA IS CLEAN AND FREE FROM METAL CHIPS AND SHAVINGS.

(2) Guide cells into position and properly seat cell openings and cutouts at respective fitting attachment locations.

(3) Install snap hangers to holding brackets on upper wing skin.

(4) Install pump and quantity transmitter flange fitting.

(5) Install water/methanol feed outlet connection; use new O-ring and Stat-O-Seals

(6) Install quantity transmitter, see Water/Methanol Quantity Transmitter - Removal/Installation, Section 82-6-0.

(7) Install boost pump, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

(8) Insert feed line through upper access hole and connect to front beam outlet fitting then connect to pump.

CAUTION: WHEN INSTALLING VENT LINE ENSURE THAT SCARF CUT FACES FORWARD.

(9) Install vent valve and tubing.

(10) Install access panels on upper wing only.

(11) Service water/methanol cells and inspect for leaks during servicing, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(12) Bleed water/methanol cells and inspect for leaks, see Water/Methanol System - Bleeding, Section 82-0-1.

(13) Inspect area for foreign objects and damage.

(14) Install lower access panel and wing leading edges, ensure deice lines are connected.

5. Inboard / Outboard Water/Methanol Cell — Removal / Installation

NOTE: This procedure applicable to Aircraft 1 - 106 and 114 not having ASC 165.

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Remove electrical power from aircraft.

(a) Drain water/methanol cells, see Water/Methanol System - Draining/Servicing, Section 82-0-1. this Section.

(b) Remove bolts and washers at each interconnector; then remove collars.

(c) Disconnect snap hanger securing cell to wing upper skin.

(d) Collapse cell uniformly and roll lengthwise.

(e) Remove cell through upper access hole.

(f) Discard all O-rings and gaskets.

(2) Remove inboard cell as follows: (See Figure 202)

(a) Working inside inboard cell, disconnect feed line from pump and front beam connection; Remove feed line through upper access hole.

(b) Remove quantity transmitter, see Water/Methanol Quantity Transmitter - Removal/Installation, Section 82-6-0.

(c) Remove water/methanol boost pump, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

(d) Remove outboard wing leading edge.

(e) Disconnect feed line Wiggins fitting from elbow on forward face of front beam.

(f) Remove outlet connection through upper access hole.

(g) Working at lower access hole, remove pump/quantity transmitter flange fitting.

(h) Remove bolts and washers at each interconnector; then remove collars.

(i) Disconnect snap hangers securing cell to wing upper skin.

(j) Collapse cell uniformly and roll lengthwise.

(k) Remove cell through upper access hole.

(3) Discard all O-rings and gaskets.

(4) Inspect wing cavity for damage to liners. Repair or replace liners as required.

B. Installation

(1) Install inboard cell as follows:

NOTE: Inspect all components for damage and/or corrosion prior to installation.

(a) Collapse cells uniformly and roll lengthwise.

CAUTION: WHEN INSTALLING CELLS AVOID CONTACT WITH SHARP OR ABRASIVE OBJECTS. ENSURE WING CELL CAVITY AREA IS CLEAN AND FREE OF FOREIGN OBJECTS.

(b) Guide cells through access in upper skin into position, properly seat cell openings and cutouts at respective fitting attach locations.

(c) Connect snap hangers to holding brackets on wing upper skin.

(d) Install interconnector collars; use new O-ring and Stat-O-Seals.

(e) Install pump/quantity transmitter flange fitting.

(f) Using new O-ring, install water/methanol feed outlet connection.

(g) Install boost pump, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

(h) Install quantity transmitter, see Water/Methanol Quantity Transmitter - Removal/Installation, Section 82-6-0.

(i) Insert feed line through upper access and connect to front beam outlet fitting and then connect to pump.

CAUTION: WHEN INSTALLING VENT LINE ENSURE THAT SCARF CUT FACES FORWARD.

(2) Install outboard cell as follows:

NOTE: Inspect all components for damage and/or corrosion prior to installation.

(a) Collapse cells uniformly and roll lengthwise.

CAUTION: WHEN INSTALLING CELLS AVOID CONTACT WITH SHARP OR ABRASIVE OBJECTS. ENSURE WING CELL CAVITY AREA IS CLEAN AND FREE OF FOREIGN OBJECTS.

(b) Guide cells through access in upper skin into position, properly seat cell openings and cutouts at respective fitting attach locations.

(c) Connect snap hangers to holding brackets on wing upper skin.

(d) Install interconnector collars; use new O-ring and Stat-O-Seals.

(e) Install vent valve, tubing and bracket in outboard cell. See Water/Methanol Vent Valve - Removal/Installation, this Section.

(3) Inspect cells for foreign objects.

(4) Install access panels on upper wing only.

(5) Service water/methanol cells and inspect for leaks during servicing, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(6) Bleed water/methanol cells and inspect for leaks, see Water/Methanol System - Bleeding, Section 82-0-1.

(7) Inspect area for foreign objects and damage.

(8) Install lower access panel and wing leading edges, ensure deice lines are connected.

6. Water/Methanol Vent Valve (Dual Cell) — Removal / Installation

NOTE: This procedure applicable to Aircraft 1 - 106 and 114 not having ASC 165.

WARNING: WATER METHANOL IS TOXIC. USE IN WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Remove electrical power from aircraft.

(2) Drain water/methanol cells, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(3) Remove wing access panel 21/22-UPR-4.

(4) Remove bolt and washer holding vent tube to mounting bracket (See Figure 202).

(5) Disconnect vent tube at Wiggins coupling.

(6) Remove tube with vent valve.

(7) Remove Wiggins fitting assembly from both tubes and discard O-rings.

(8) Remove vent valve from tube. NOTE orientation of valve to tube.

(9) If required, remove vertical tube as follows:

(a) Remove bolts, backup washers and Stat-O-Seal washers holding vertical tube to lower wing cover.

(b) Lift and remove vertical tube. Discard O-ring and Stat-O-Seal washers.

B. Installation

(1) Inspect tubes for corrosion.

CAUTION: WHEN INSTALLING VENT LINE ENSURE THAT SCARF CUT FACES FORWARD.

(2) If removed, install vertical tube as follows:

(a) Install new O-ring in vertical tube flange.

(b) Using new Stat-O-Seal washers, position vertical tube on holding bracket.

CAUTION: ENSURE CELL CUTOUT AND TUBE FLANGE ARE PROPERLY SEATED ON MOUNTING BRACKET BEFORE TIGHTENING FASTENERS.

(c) Install and tighten washers and bolts.

(3) Install new O-rings in Wiggins coupling.

CAUTION: FLOAT MUST BE ON LOWER SIDE OF VENT VALVE.

(4) Install horizontal tube and valve. Ensure vent valve is correctly oriented.

(5) Connect vent tube at Wiggins coupling and safety-wire.

(6) Install washer and bolt holding vent tube to mounting bracket.

NOTE: Lifting float should close flapper valve.

(7) Check operation and position of float and flapper valve.

(8) Inspect area for foreign objects, damage and security of all attachments.

(9) Install access panel.

(10) Service water/methanol cells and inspect for leaks during servicing, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(11) Bleed water/methanol cells and inspect for leaks, see Water/Methanol System - Bleeding, Section 82-0-1.

7. Water/Methanol Vent Valve (Single Cell) — Removal / Installation

NOTE: This procedure applicable to Aircraft 1 - 106 and 114 having ASC 165 and 107 - 200, 322 and 323.

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Remove electrical power from aircraft.

(2) Drain water/methanol cells, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(3) Remove access panels 21/22-UPR-3 and 21/22-UPR-4.

(4) Remove bolt and washer from clamp that secures vent tube to bracket (See Figure 203).

(5) Unscrew B-nut at cell fitting and remove vent tube.

(6) Unscrew cell fitting retaining nut. Remove backup washer.

(7) Remove vent valve. **NOTE** orientation of valve to cell.

B. Installation

(1) Inspect vent tube for corrosion and damage.

CAUTION: FLOAT MUST BE ON LOWER SIDE OF FLAPPER VALVE.

(2) Install vent valve and washer in cutout inside cell. Ensure vent valve is correctly oriented.

(3) Through access hole outboard of cell, install backup washer and retaining nut.

CAUTION: WHEN INSTALLING VENT LINE ENSURE THAT SCARF CUT FACES FORWARD.

(4) Position vent tube and tighten B-nut to valve.

(5) Clamp vent tube to mounting bracket and secure.

NOTE: Lifting float should close flapper valve.

(6) Check operation and position of float and flapper valve.

(7) Inspect area for foreign objects, damage and security of all attachments.

(8) Install access panels.

(9) Service water/methanol cells and inspect for leaks during servicing, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(10) Bleed water/methanol cells and inspect for leaks, see Water/Methanol System - Bleeding, Section 82-0-1.

8. Water/Methanol Vent Valve (Metal Tank) — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Remove landing light panel access panel 107-LH FIL-2 or 107-RH FIL-2.

(2) Drain water/methanol tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(3) Remove Wiggins couplings and bonding jumper from overboard line, (See Figure 204).

(4) Remove water/methanol boost pump, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

(5) Through tank cutout for water/methanol pump, remove Wiggins coupling joining vent line.

(6) Remove clamp securing vent tube in tank.

(7) Remove upper vent tube and float through pump outlet.

(8) Remove vent assembly and gasket through vent outlet.

(9) Remove vent valve. **NOTE** orientation of valve to cell.

(10) Discard O-rings and gasket.

B. Installation:

(1) Inspect vent tubes for corrosion and damage.

CAUTION: FLOAT MUST BE ON LOWER SIDE OF FLAPPER VALVE.

(2) Install vent valve using new gasket and Stat-O-Seal. Ensure vent valve is correctly oriented.

CAUTION: WHEN INSTALLING VENT LINE ENSURE THAT SCARF CUT FACES FORWARD.

(3) Install upper vent tube and float through pump outlet.

(4) Install clamp securing vent tube.

(5) Install new O-rings in Wiggins coupling.

(6) Connect Wiggins coupling to vent tube.

(7) Install water/methanol boost pump, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

(8) Install Wiggins coupling with new O-rings and bonding jumper on overboard line.

(9) Service water/methanol tanks and check for leaks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(10) Inspect area for foreign objects, security of all attachments.

(11) Install access panel.

9. Water/Methanol Tank — Cleaning / Internal Surface Protection

NOTE: Aircraft having ASC 125 have aluminum water/methanol tanks which develop internal corrosion. Following procedure is a method of cleaning and surface reprotection which will provide a more servicable tank.

A. Locate aircraft in a place where handling of paints and solvents will not be a problem.

B. Drain water/methanol tank completely, see Water/Methanol System Draining/ Servicing, Section 82-0-1.

C. Remove landing light access panel 107-LH FIL-2 or 107-RH FIL-2.

D. Remove power from aircraft, set all system switches to OFF and disconnect batteries at battery quick disconnects.

E. Remove water/methanol pump, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

F. Remove vent valve, see Water/Methanol Vent Valve - Removal/Installation, this Section.

G. Remove water/methanol quantity transmitter, see Water/Methanol Quantity Transmitter - Removal/Installation, Section 82-6-0.

H. Inspect tank for location of any corrosion.

NOTE: If visual inspection discloses an extreme corrosion problem and a repair scheme is needed, contact Gulfstream Aerospace, Technical Operations Department.

I. Fabricate and install temporary cover plates and gaskets for tank openings uncovered by removal of components.

J. Disconnect all lines at tank and plug ports as necessary.

K. Provide a drain plug on lowest temporary cover plate.

L. Remove tank filler cap and clean tank or tanks as follows:

(1) Mix 7 gallons of Turco W. O. #1 with 21 gallons of fresh water in a suitable container.

(2) Fill water/methanol tank with Turco solution using a pump or gravity head. Direct discharge from transfer hose against top portion of tank to flush area above filler neck.

(3) Let solution stand in tank for 10 minutes. Solution may be agitated with an air hose to increase washing action.

NOTE: Avoid excessive spillage of solution to minimize external clean up.

(4) Drain Turco solution and fill tank with fresh water using same procedures used in Steps L.(2) and L.(3) above.

(5) Drain tank, remove cover plate from pump opening and air dry tank with warm air (approximately 150°F max) for 2 hours.

M. Reprotect tank as follows:

(1) Install pump opening cover plate and fill tank with Alodine No. 1200. Follow procedure used in Steps L.(2) and L.(3) above except tank shall be drained as soon as full level is reached.

NOTE: Alodine shall be mixed 0.5 - 2.5 oz/gal, ph 1.5 - 2.0, at a temperature of 65° - 95°F. and conform to MIL-C-5541B, class 1a, for application.

(2) Fill tank with fresh water as in Step L.(4) above and agitate for 10 minutes.

(3) Drain tank, remove cover plate from pump opening and air dry tank with warm air (approximately 150°F max) for 2 hours.

(4) When dry, spray inner surface of tank with Polyurethane Tank Coating in accordance with MIL-C-27725.

NOTE: Because of tank shape, and size of openings through which to work, there is possibility that small surface areas will not receive a coating of polyurethane, but since original tank was only alodined any polyurethane coating will be a large improvement.

If tanks are removed from aircraft a quantity of polyurethane finish could be put in tank and tank rotated until all surfaces are coated.

(5) After coating, remove all cover plates and permit the finish to dry for 30 to 40 minutes prior to handling.

NOTE: Some drainage of excess Polyurethane may be experienced after spraying, consequently, drip pans should be placed to catch any discharge.

Finish will be fully cured after 14 days but tank may be returned to service after 16 hours.

N. Install quantity transmitter, see Water/Methanol Quantity Transmitter - Removal/Installation, Section 82-6-0.

O. Install vent valve, see Water/Methanol Vent Valve - Removal/Installation, this Section.

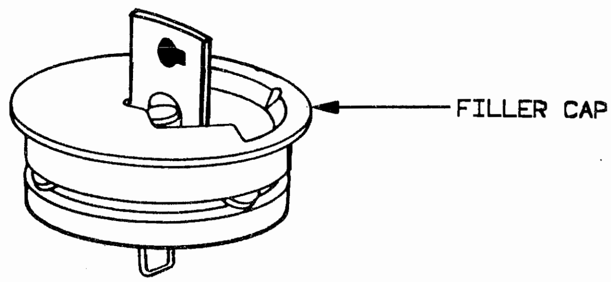
P. Install water/methanol boost pump, see Water/Methanol Boost Pump - Removal/Installation, Section 82-3-0.

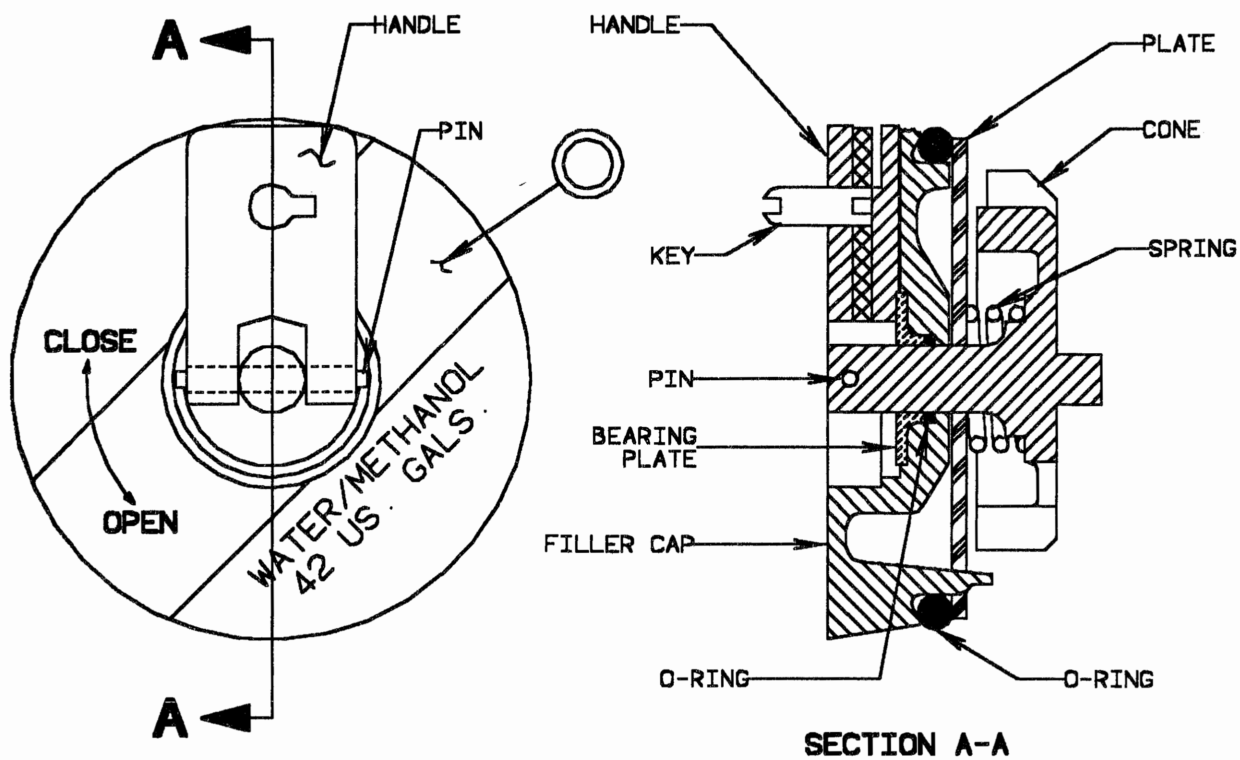
Q. Service water/methanol tanks and inspect for leaks during servicing, see Water/Methanol System Draining Servicing, Section 82-0-1.

R. Bleed water/methanol cells and inspect for leaks, see Water/Methanol System - Bleeding, Section 82-0-1.

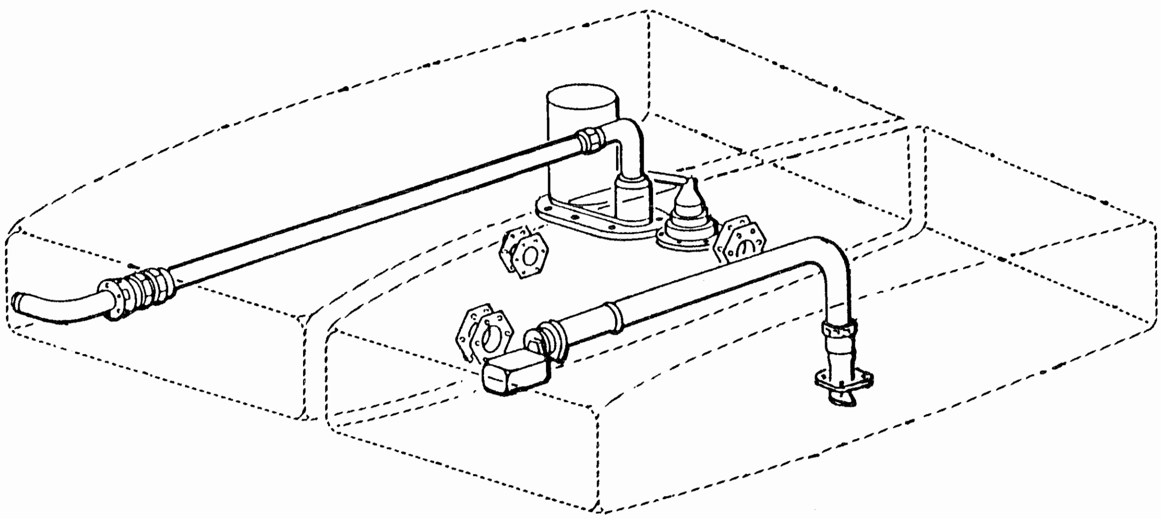
S. Inspect area for of foreign objects, damage and security of all attachments.

T. Install access panels.

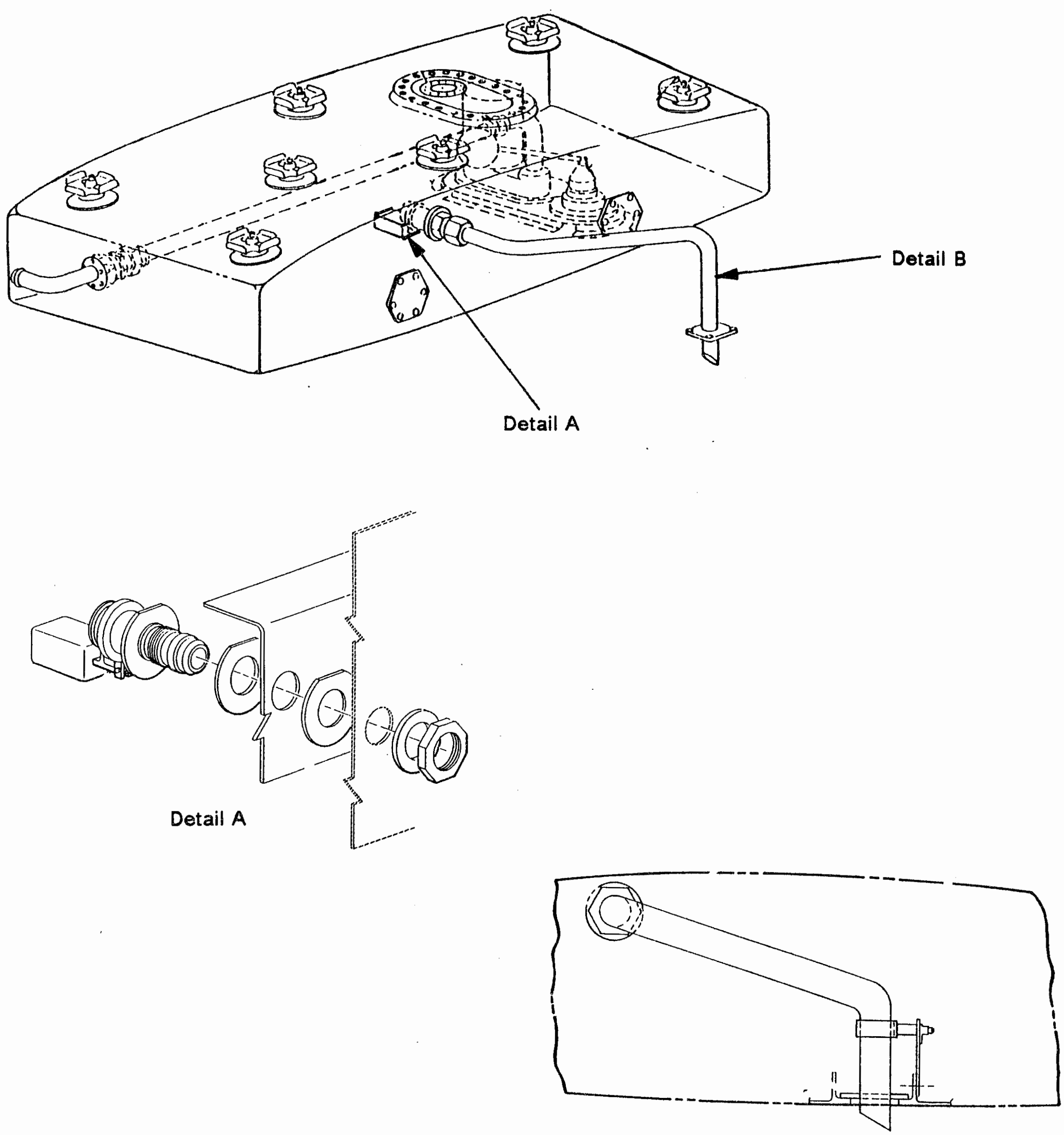




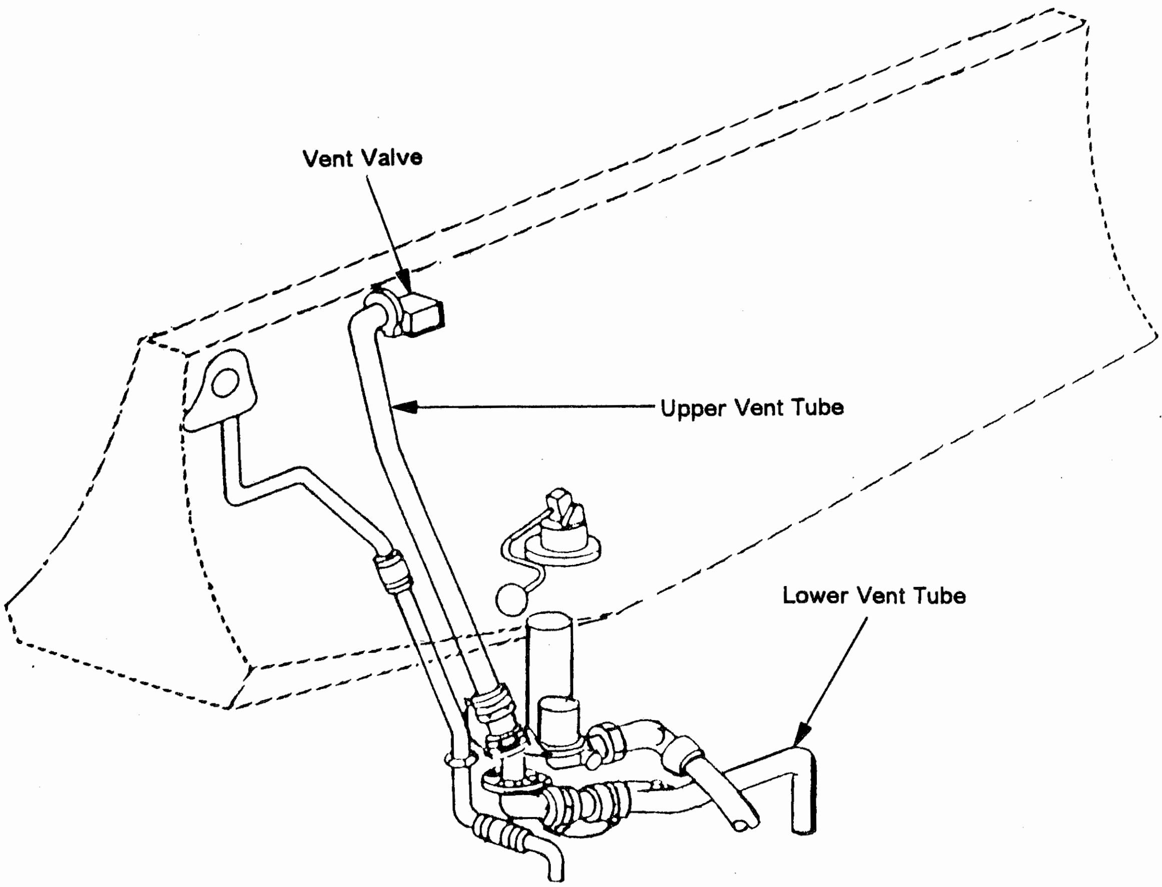
Water/Methanol Filler cap
Figure 201.



Water/Methanol Vent System
(Aircraft 1-106 and 114 not having ASC 165)
Figure 202.



Water/Methanol Vent System
(Aircraft 1-106 and 114 having ASC 165 and 107-200, 322 and 323)
Figure 203.



Water/Methanol Vent System
(Aircraft Having ASC 125)
Figure 204.

WATER/METHANOL INJECTION CONTROL SYSTEM — DESCRIPTION / OPERATION

1. General

The water/methanol injection control system consists of the necessary components installed in the aircraft to deliver the fluid from the tank system to the engine when required. Though two separate configurations of the water/methanol injection system are presently in use in aircraft and their installation varies accordingly, basic description, operation and maintenance are the same. Differences between the two configurations will be reflected accordingly throughout this section.

In utilizing this section, first ascertain whether the aircraft in question has ASC 125 installed or not and then follow the applicable parts of this section.

2. Description

The water/methanol injection control system consists of boost pumps, pressure switches, shutoff valves and all necessary controls. The system supplies fluid to the water/methanol control (metering) unit at a required rate and pressure and also provides for immediate cutoff of fluid if there is an engine fire, engine failure or automatic propeller feathering. A boost pump is installed in each system. The low-pressure switch and sensing line are tapped in each feed line upstream of the flow check valve. When feed line pressure falls below 19.5 psi the pressure switch closes and causes the L and/or R W/M PRESS warning lights in the master warning system to come on. The shutoff valves control the flow of water/methanol from the cells/tanks to the control unit and are among the components operated by FIRE PULL T-handles. Each valve is supplied power from the 28 volt essential dc bus through a 5A circuit breaker, on the left circuit breaker panel.

3. Normal Operation

The system is controlled by a single four pole ARMED/OFF switch located on cockpit lower overhead panel. The pumps are powered and valves are energized open through their respective relays when the WATER/METHANOL switch is set to ARMED. These actions supply water/methanol under pressure to the water/methanol control units on the engines. Placing the switch to OFF, deactivates pumps, closes shutoff valves and turns off warning lights. When a FIRE PULL T-handle is pulled or when manual feathering of an engine occurs, the appropriate valve control relay is de-energized thus allowing the valve to close. The boost pump within each water/methanol cell/tank then continues to operate and supply fluid to remaining engine by the crossfeed line.

When the system is required to restore power, it should always be set to ARMED with engines operating at 14,000 rpm or less. This is done by setting the WATER/METHANOL switch to ARMED and checking that the W/M PRESS lights blink and then remain OFF.

When power levers are advanced to takeoff power (15,000 rpm), torque pressure gradually rises to the water/methanol check pressure (WMCP) (See engine calibration card in cockpit). Injection of water/methanol is controlled by a metering unit, which is sensitive to propeller shaft torque through torque meter oil pressure, ambient pressure through two opposed capsules and engine speed by interconnection with throttle and rpm controls.

For flight operation at weights which necessitate use of wet takeoff power for compliance with airworthiness performance, total water/methanol supply at start of takeoff must not be less than the minimum required quantity (refer to Flight Manual) and system must be operating for takeoff, approach, and landing as appropriate. When a safe altitude is reached after takeoff, power levers are retarded to 14,200 rpm and the WATER/METHANOL switch is set to OFF.

4. Electrical Operation

As previously stated, the pumps are controlled by the WATER/METHANOL switch. With the switch OFF, valves are energized closed. When switch is ARMED (with FIRE PULL T-handles in the normal position and HIGH PRESSURE (HP) FUEL COCK levers are in a position other than FEATHER), the following occurs:

A. Power comes from the essential dc bus through a 5A circuit breaker through the arming switch to both the right and left water/methanol pump relays (See Figure 1).

B. The main dc bus power is waiting at the contacts of the pump relays.

C. As each relay coil is actuated, A1 and A2 make contact and allow this power to operate left and right pumps.

D. In addition, a second circuit from the arming switch goes through the HP FUEL COCK lever switches (L and R), through the FIRE PULL T-handle switches (L and R), to the right and left water/methanol valve relays.

E. The coils shift the armatures from B3 to power the right and left water/methanol valves open. These are motor operated, limit switch controlled, shutoff valves.

If a FIRE PULL T-handle is pulled as a result of a nacelle fire, the appropriate valve relay is de-energized. With the relay in the relaxed position the valve is energized to close. This action stops the flow of water/methanol to that particular engine. Both pumps continue to run and fluid from both cells/tanks is available to the operative engine through the crossfeed line. Similar action occurs when a HP FUEL COCK lever is moved to the FEATHER position.

A third circuit from the arming switch goes to the L and R W/M PRESS warning switches. If water/methanol pressure is 20.5 psi or over, B and A break contact and the warning light goes out. If pressure drops below 19.5 psi then B and A make contact and send power to L or R W/M PRESS warning light in the master warning system.

5. Crossfeed Operation

The crossfeed function of the water/methanol system comes into operation automatically if either one of two pumps fail. In normal configuration, both pumps are supplying water/methanol to their respective engines. Pressure in the crossfeed line is also built up, but theoretically water/methanol does not crossfeed because pressure forces are opposing one another. Several factors enter into the unequal use of water/methanol. The two most important factors are, unequal requirements of the engines and production tolerances on boost pump pressure.

Engine requirements cannot be corrected, but production tolerances on the pumps are compensated for by two opposing direction crossflow check valves, which are connected in parallel. These valves have a 2.5 psi cracking pressure and a 3 psi full flow. Under normal operation, the crossfeed line has no flow because the crossflow check valve cracking pressure is equal to the boost pump tolerance.

If one pump fails, pressure falls off so that 2.5 psi is attained on one valve or the other depending on which pump failed. The water/methanol mixture/fluid now crossflows from the cell/tank with operating pump, and feeds both engines. If one engine fails, the remaining engine is fed from the cell/tank on that side. As the fluid in the cell/tank is depleted, the pressure on that side drops off and the crossflow check valve opens allowing fluid from the opposite tank/cell to feed the remaining engine.

6. Fire Pull T-Handle Operation

One function of FIRE PULL T-handle operation is to automatically shut off flow of water/methanol to the affected engine, eliminating possibility of inflammable fluid from entering a fire area.

7. High Pressure Fuel Cock Lever Operation

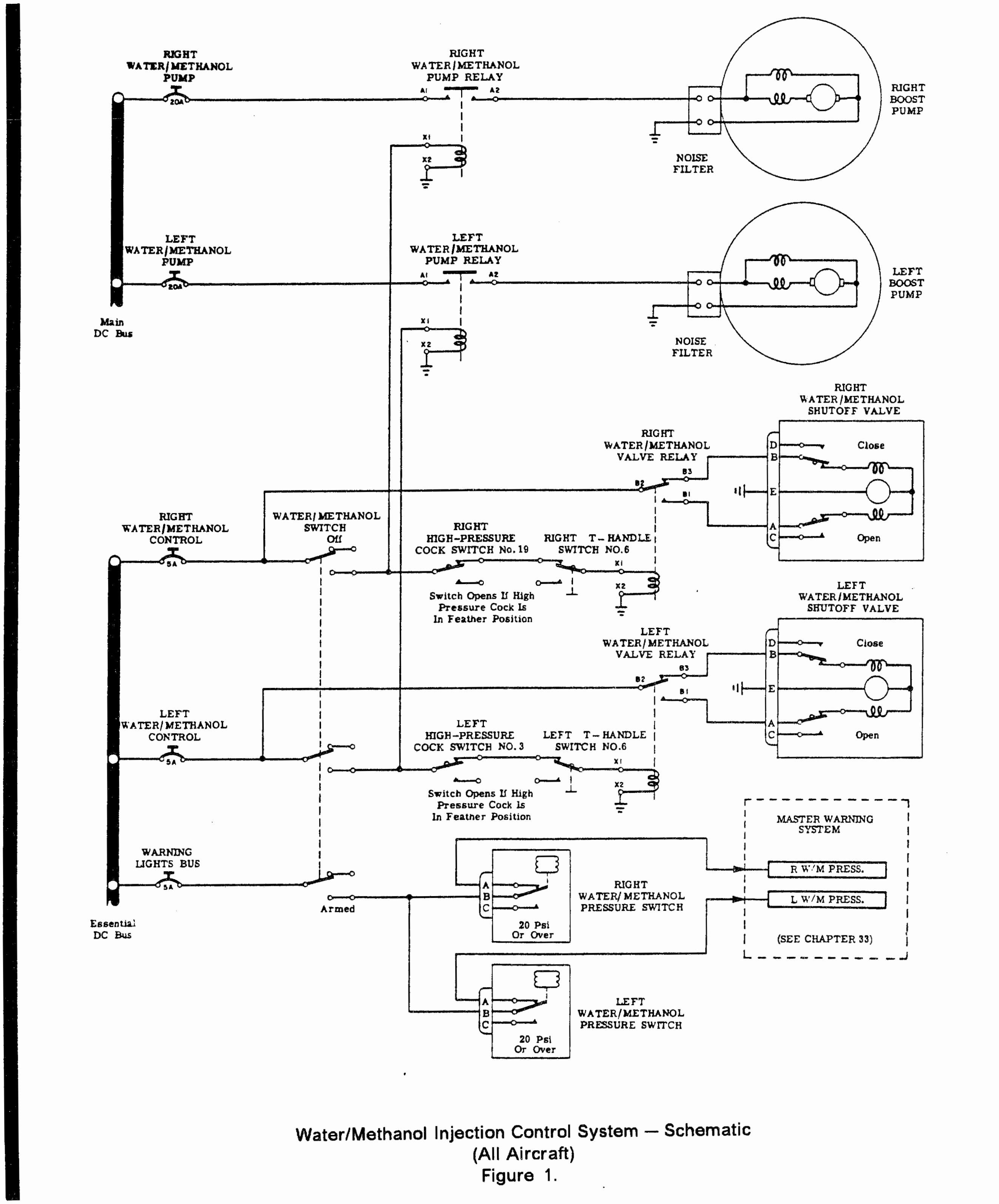
When the HP FUEL COCK lever is placed in FEATHER position, if the WATER/METHANOL switch is ARMED, the shutoff valve to that engine closes, preventing flow to a feathered engine. If the valve is closed (switch off), and subsequently armed after the HP FUEL COCK lever is moved to FEATHER, the valve will not open to that engine.

8. Emergency DC Operation

Water/methanol is not available to the engines in emergency dc operation due to the fact that water/methanol pumps are not operative.

The water/methanol shutoff valves are operative for fire protection purposes only (FIRE PULL T-handle shutoff or HP FUEL COCK lever to FEATHER).

The L and R W/M PRESS warning lights (master warning system) are operative within the emergency dc operation capability of the master warning system.



Water/Methanol Injection Control System — Schematic
(All Aircraft)
Figure 1.

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WATER/METHANOL INJECTION CONTROL SYSTEM — MAINTENANCE PRACTICES

1. Water/Methanol Low-Pressure Switch — Functional Test

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

CAUTION: ENSURE ELECTRICAL POWER IS OFF.

A. Gain access by removing following panels.

(1) Aircraft 1 - 200, 322 and 323 not having ASC 125, remove upper access panel 21/22-UPR-2.

(2) Aircraft 1 - 200, 322 and 323 having ASC 125, remove landing light cover access panel 107-LH FIL-1.

B. Remove and cap pressure sensing line to low pressure switch. (See Figure 201).

C. Connect air supply with pressure gage to switch.

D. Pull L W/M PUMP and R W/M PUMP circuit breakers.

CAUTION: ENSURE CIRCUIT BREAKERS ARE PULLED AND LINE FROM PRESSURE SWITCH HAS PLUG INSTALLED BEFORE APPLYING ELECTRICAL POWER.

E. Energize main and essential dc bus.

F. Place WATER/METHANOL switch to ARMED. W/M PRESS warning light for side being checked should come on.

G. Slowly apply increasing air pressure to switch. W/M PRESS warning light for side being checked should go out before pressure reaches 20.5 psi.

H. Increase pressure to 23 psi.

I. Slowly decrease pressure. W/M PRESS warning light for side being checked should come on before 19.5 psi is reached.

J. Place WATER/METHANOL switch OFF.

K. Disconnect air supply with air pressure gage and remove plug from pressure sensing line.

L. Connect pressure sensing line to pressure switch.

M. Depress L W/M PUMP and R W/M PUMP circuit breakers.

N. Place WATER/METHANOL switch to ARMED. Check for leaks.

O. De-energize main and essential dc bus.

P. Inspect area for foreign objects and damage.

Q. Install access panel.

NOTE: Repeat procedure for opposite pressure switch if both are to be checked.

2. Water/Methanol Low-Pressure Switch — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Remove electrical power from aircraft.

(2) Gain access by removing following panels.

(a) Aircraft 1 - 200, 322 and 323 not having ASC 125, remove upper access panels 21/22-UPR-2.

(b) Aircraft 1 - 200, 322 and 323 having ASC 125, remove landing light cover access panel 107-LH FIL-1.

(3) Disconnect pressure sensing line (See Figure 201).

(4) Disconnect electrical connector.

(5) Remove low-pressure switch.

(6) Remove fitting and discard O-ring.

B. Installation

(1) Install fitting with new O-ring.

(2) Install pressure switch.

(3) Connect electrical connector and lockwire.

(4) Inspect area for of foreign objects and damage.

(5) Perform Water/Methanol Low Pressure Switch - Functional Test and check for leaks, see this Section.

(6) Install access panel.

3. Water/Methanol Crossflow Check Valve — Operational Test

NOTE: The following procedures represent alternate methods of checking the crossflow check valve and either may be used.

Both procedures below will identify a crossflow check valve failed in the closed position. Abnormal uneven use of water/methanol between right and left cells/tanks, could indicate a crossflow check valve failed in open position.

A. With engine operating;

(1) Start left engine using normal procedures. See GI Aircraft Flight Manual Engine Starting Procedures.

(2) Check engine oil temperature - must be at least 65°C.

(3) Set fuel datum to 50%.

(4) Pull L W/M PUMP circuit breaker.

(5) Place WATER/METHANOL switch to ARMED (below 14,000 rpm). L W/M PRESS warning light should be on and R W/M PRESS warning light should be out.

(6) Slowly open LH POWER LEVER and observe increase in torque at water injection cut in.

(7) With power lever fully open, torque meter pressure should read +7, −0 of corrected data plate water/methanol check pressure. This verifies crossflow check valve with flow direction towards left engine is open.

(8) Reduce POWER LEVER back to IDLE.

(9) Place WATER/METHANOL switch OFF.

(10) Reset L W/M PUMP circuit breaker.

(11) Shut down engine using normal procedures. See GI Aircraft Flight Manual, Engine Shutdown Procedures.

(12) Repeat procedure above utilizing RH engine and pulling R W/M PUMP circuit breaker to check opposite crossflow check valve.

B. Without engines operating;

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

(1) Gain access to LH water/methanol drain valve, or LH filter assembly and attach a suitable length of hose to drain port/fitting, with free end of hose directed into a clean container.

(2) Pull L W/M PUMP circuit breaker.

(3) Apply electrical power.

(4) Place WATER/METHANOL switch to ARMED.

(5) Repeat procedure above utilizing RH engine and pulling R W/M PUMP

NOTE: The following step verifies crossflow check valve with flow direction towards left engine is open.

(6) Ensure one of the following occurs.

(a) Open LH drain valve and ensure water/methanol mixture is flowing under pressure from hose.

(b) If hose is on filter drain, ensure water/methanol mixture is flowing under pressure from hose when shutoff valve opens.

(7) Place WATER/METHANOL switch OFF.

(8) Reset L W/M PUMP circuit breaker.

(9) Repeat procedure above utilizing RH drain valve/filter drain fitting and pulling R W/M PUMP circuit breaker to check opposite crossflow check valve.

(10) Remove electrical power from aircraft.

(11) Remove hose from aircraft. Ensure drain valve is closed or filter drain fitting cap is installed and tightened securely.

(12) Inspect areas for foreign objects, damage and security of all attachments.

(13) Install all panels removed.

4. Water/Methanol Crossflow Check Valve — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Remove electrical power from aircraft.

(2) Drain water/methanol cells/tanks. See Water/Methanol - Draining/Servicing, Section 82-0-1.

(3) Gain access to crossflow valve tube assembly as follows (See Figure 202).

(a) Aircraft 1 - 200, 322 and 323 not having ASC 125, extend flaps to TAKE-OFF.

(b) Aircraft 1 - 200, 322 and 323 having ASC 125, remove landing light cover access panel 107-RH FIL-2.

(4) Loosen B-nuts at check valve.

(5) Remove check valve, note direction of flow arrows.

B. Installation

CAUTION: INSTALL VALVES SO FREE FLOW ARROWS WILL BE IN OPPOSITE DIRECTIONS TO EACH OTHER.

(1) Inspect tubes (including flared ends) for corrosion and damage.

(2) Install check valve.

(3) Service water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(4) Bleed water/methanol system and check for leaks during bleeding, see Water/Methanol System - Bleeding, Section 82-0-1.

(5) Inspect area for foreign objects and damage.

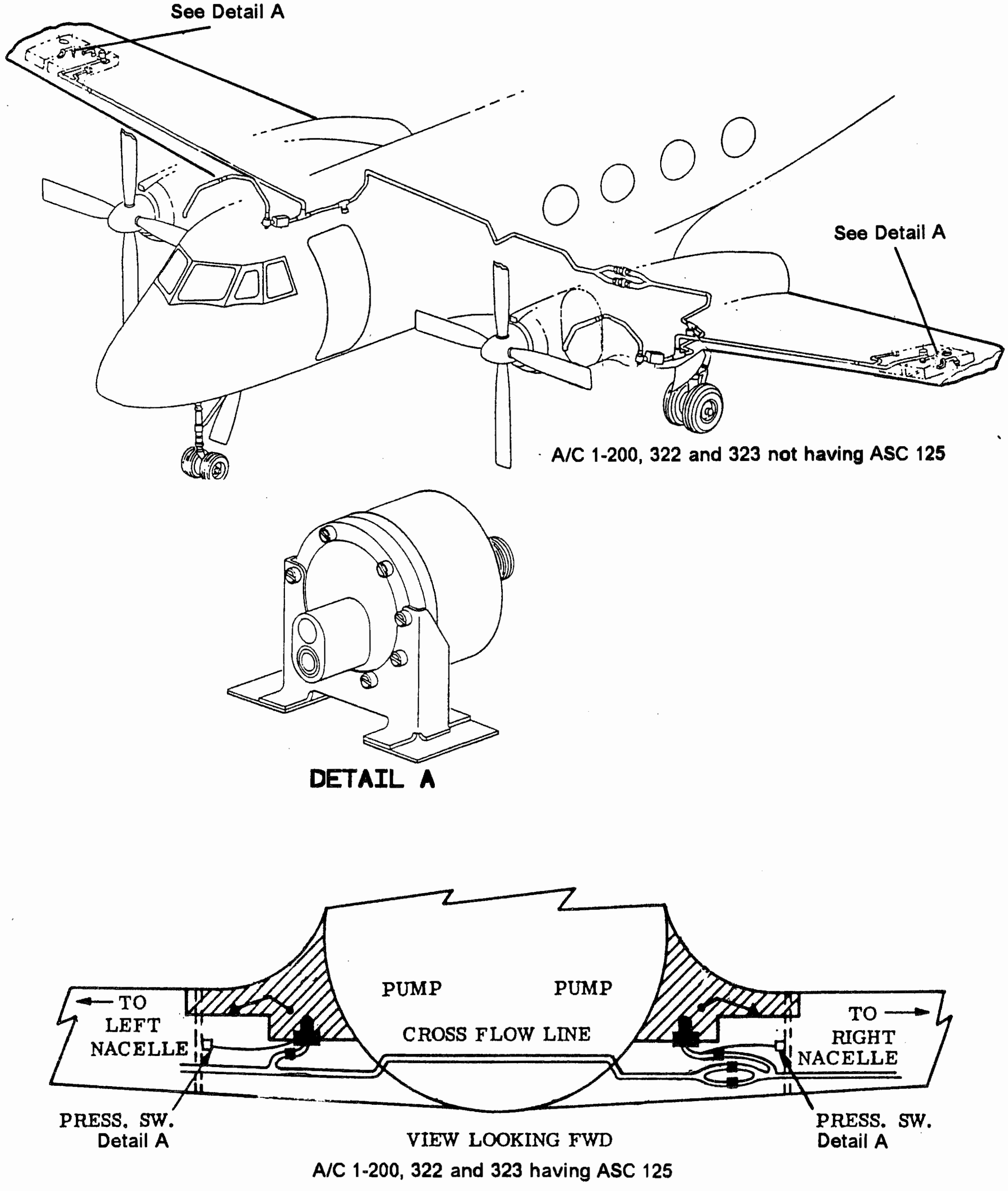
(6) Retract flaps or install access panel.

5. Water/Methanol Control Unit — Removal / Installation

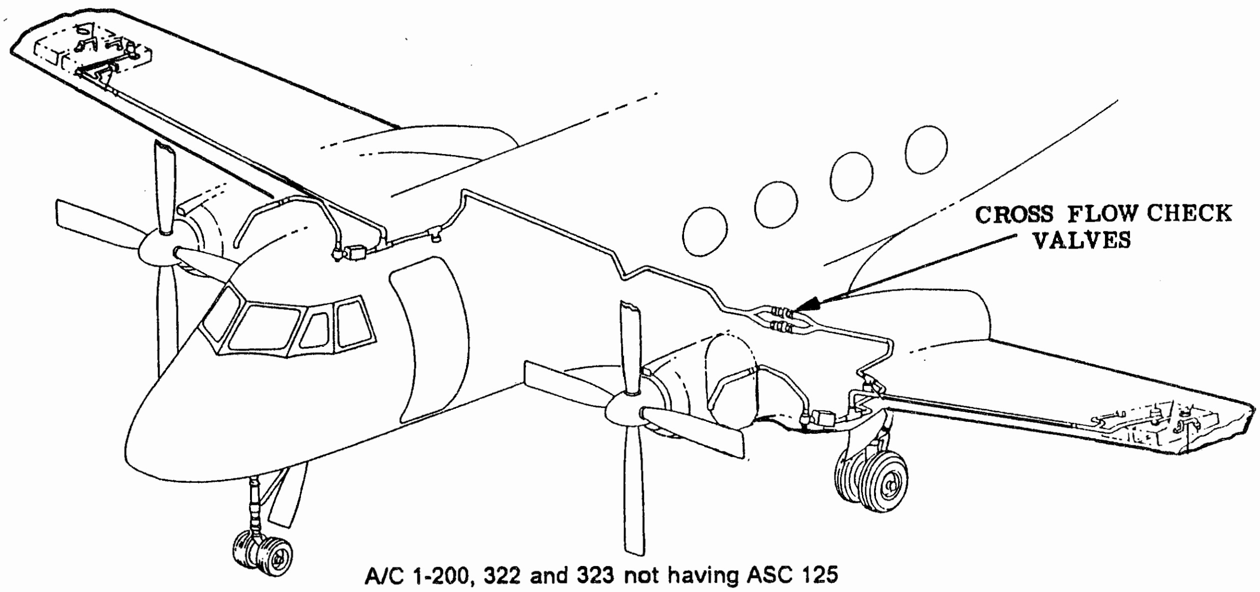
A. Refer to Rolls-Royce Dart Maintenance manual (M-Da7-G)

6. Water/Methanol Control Unit — Adjustment / Check

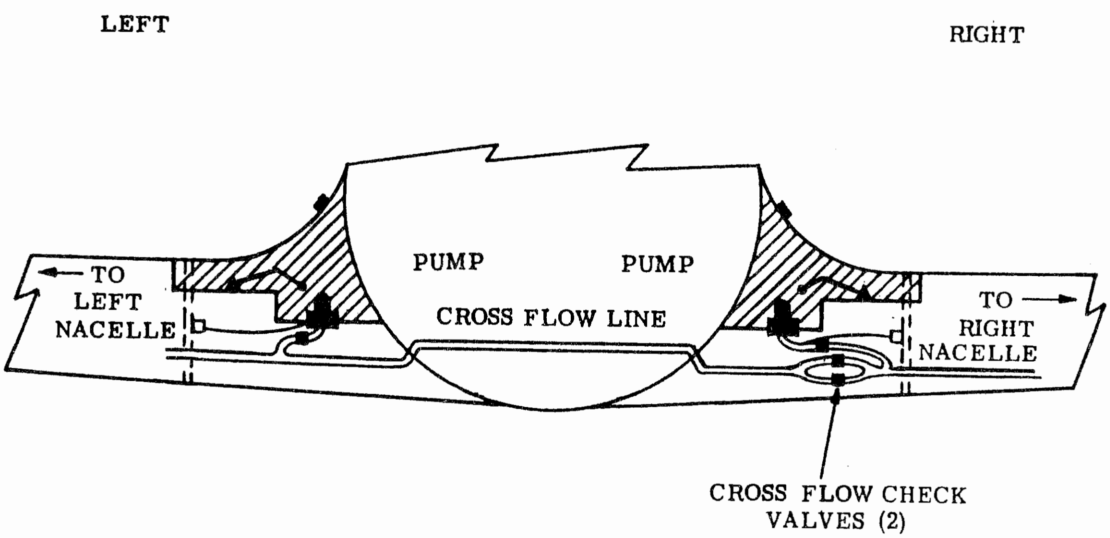
A. Refer to Rolls-Royce Dart Maintenance manual (M-Da7-G)



Water/Methanol Pressure Switch
Figure 201.



A/C 1-200, 322 and 323 not having ASC 125



VIEW LOOKING FWD
A/C 1-200, 322 and 323 having ASC 125

Water/Methanol Crossflow Check Valve
(Aircraft having ASC 125)
Figure 202.

WATER/METHANOL INJECTION SYSTEM BOOST PUMPS — DESCRIPTION / OPERATION

1. General

The water/methanol injection boost pumps function to deliver the water/methanol mixture from the tank, metered by the control devices, to the system in sufficient quantity and under sufficient pressure for utilization by the engine when required. Each pump is a positive displacement type pump driven by a compound wound, 28V dc motor, capable of delivering up to a maximum of 720 gallons per hour. Though two separate configurations of the water injection system are presently in use in the aircraft and their pump installation varies accordingly, basic description, operation and maintenance are the same. Differences between the two configurations will be reflected accordingly throughout this Section.

In utilizing this section, first ascertain whether the aircraft in question has ASC 125 installed or not, and then follow the applicable parts of this section.

A. Aircraft 1 - 200, 322 and 323 not having ASC 125.

One water/methanol boost pump is installed in each wing as follows: Aircraft 1 - 106 and 114 in inboard cell, Aircraft 1 - 106 and 114 having ASC 165 and 107 - 200, 322 and 323 in single cell. The electrical leads of each pump extend to a radio noise filter located in the left and right wing dry bay between Wing Stations 307 and 313. The pumps are accessible through access holes located in the outer panel upper and lower covers between Wing Stations 313 and 331.

B. Aircraft 1 - 200, 322 and 323 having ASC 125.

One water/methanol boost pump is installed in each water/methanol tank. The electrical leads of each pump extend to a radio noise filter mounted in the inboard edge of landing light bracket at Fuselage Station 379.5 and approximately BL 24. The pumps are located at Fuselage Station 373.5 and approximately BL 34 and are mounted forward of landing light bracket and aft of rear beam. Access to the pump is through landing light access panel.

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WATER/METHANOL INJECTION SYSTEM BOOST PUMPS — MAINTENANCE PRACTICES

1. Water/Methanol Boost Pump — Pressure Check

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Remove electrical power from aircraft.

B. Ensure each tank has a minimum of 5 gallons of water/methanol.

C. Ensure water/methanol shutoff valve is closed.

D. Remove water/methanol filter drain cap.

E. Connect 0-30 psi direct reading pressure gage to water/methanol filter drain fitting.

F. Energize main and essential dc bus. Maintain a minimum 27V dc.

NOTE: Lower voltage will result in reduced pump pressure.

G. Pull opposite pump circuit breaker.

H. Ensure POWER LEVERS are in IDLE.

I. Place WATER/METHANOL switch in ARMED position and note pressure reading on gage. Zero flow pressure should read from 21-24 psi.

J. W/M PRESS warning light for pump being checked should come on momentarily and go out. Opposite pump W/M PRESS warning light should be on.

K. Place WATER/METHANOL switch in OFF.

L. Remove electrical power from aircraft.

M. Disconnect gage line at drain fitting. Install cap and tighten securely.

2. Water/Methanol Boost Pump — Preservation / Depreservation

NOTE: Preservation of water/methanol boost pump must be accomplished if either of the following applies:

If for any reason, pumps are removed from aircraft after having been immersed or operated in water/methanol mixture and are to remain out of the aircraft for periods in excess of 12 hours.

If water/methanol mixture is to be drained to facilitate work in cell/tank and more than 12 hours will elapse before refilling.

A. Preservation

(1) Completely flush pump, both internally and externally, with absolute methanol.

(2) Apply a liberal coat of high grade preserving oil (Cosmoline No. 1091, Type 3 MIL-C-16173B, Grade 3) to pump both internally and externally.

NOTE: A temporary method to preserve the pump (and other components) during maintenance is submerge removed components in a clean sealable container with approved water/methanol mixture.

B. Depreservation

(1) Drain excess preserving oil from pump internal.

(2) Completely flush pump, both internally and externally with absolute methanol to remove all preserving oil.

3. Water/Methanol Boost Pump — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(2) Gain access to pump as follows: (See Figure 201 or Figure 202)

(a) Aircraft 1 - 200, 322 and 323 not having ASC 125, remove upper access panels 21/22-UPR-2/-3 and lower access panels 21/22-LWR-1.

(b) Aircraft 1 - 200, 322 and 323 having ASC 125, remove landing light cover access panel 107-LH FIL-2 or 107-RH FIL-2, as required. Tank side panel may be removed, if desired.

(3) On Aircraft having ASC 125, disconnect pressure sensing line to low pressure switch (left side only).

NOTE: For ease of installation, a string should be connected to electrical leads before pulling them out of conduit.

(4) Disconnect electrical leads at radio noise filter.

(5) Disconnect feed line at discharge fitting on pump.

NOTE: On Aircraft not having ASC 125, loosen rigid feed tube B-nut at wing front beam feed-thru fitting.

(6) Remove bolts and washers securing pump.

(7) Remove bonding jumper and remove pump.

(8) Remove discharge fitting.

NOTE: If pump is to remain out of aircraft for periods in excess of 12 hours, perform Water/Methanol Boost Pump - Preservation/Depreservation, See this Section.

B. Installation

NOTE: Before installing a pump that has been preserved, depreserve pump, see Water/Methanol Boost Pump - Preservation/Depreservation, See this Section.

(1) Inspect area for foreign objects.

(2) Install discharge fitting with new O-ring on pump. Leave fitting loose.

(3) Install pump with new gasket.

(4) Connect bonding jumper.

(5) Connect fuel feed line to discharge fitting. If loosened, tighten B-nut on feed line wing front beam feed-thru fitting. Tighten discharge fitting.

(6) Route and connect electrical leads to radio noise filter.

(7) Safety-wire as required.

(8) On Aircraft having ASC 125, connect pressure sensing line to low pressure switch (left side only).

(9) Inspect area for foreign objects, damage and security of all attachments.

(10) Install following access panels:

(a) Aircraft not having ASC 125, install 21/22-UPR-3 panel.

(b) Aircraft having ASC 125, install tank side panel, if removed.

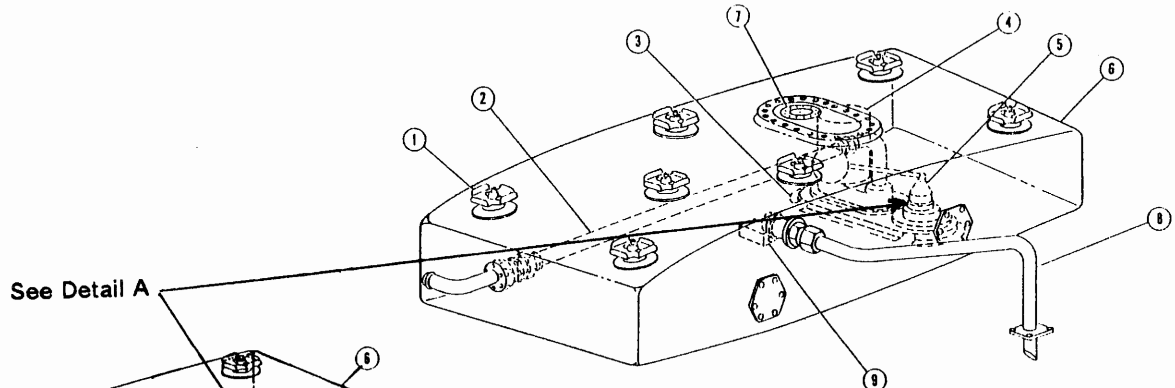
(11) Perform Water/Methanol Boost Pump - Pressure Check and inspect for leaks during test, see this Section.

(12) Install following upper access panels:

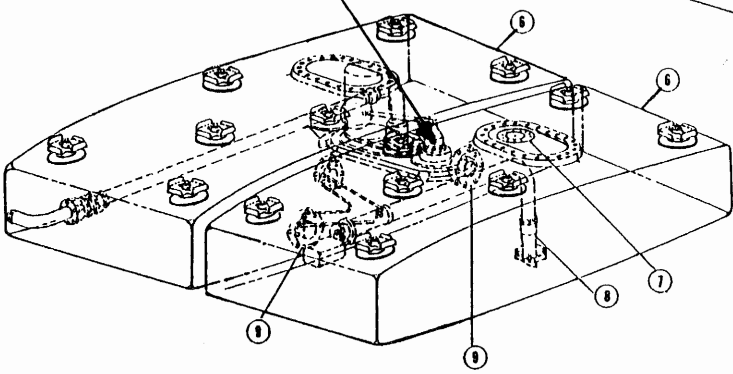
(a) Aircraft not having ASC 125, install 21/22-UPR-2/-3 and 21/22-LWR-1 access panels.

(b) Aircraft having ASC 125, install 107-LH FIL-2 or 107-RH FIL-2 access panel.

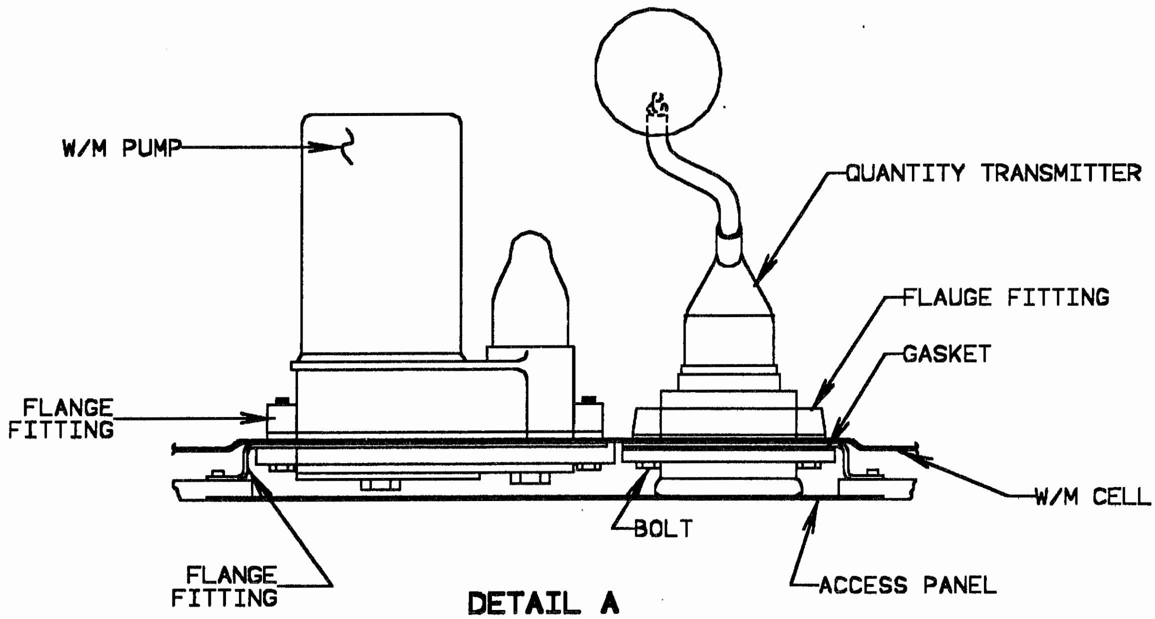
(13) Service water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.



A/C 1-106, and 114 having ASC 165
and 107-200, 322 and 323



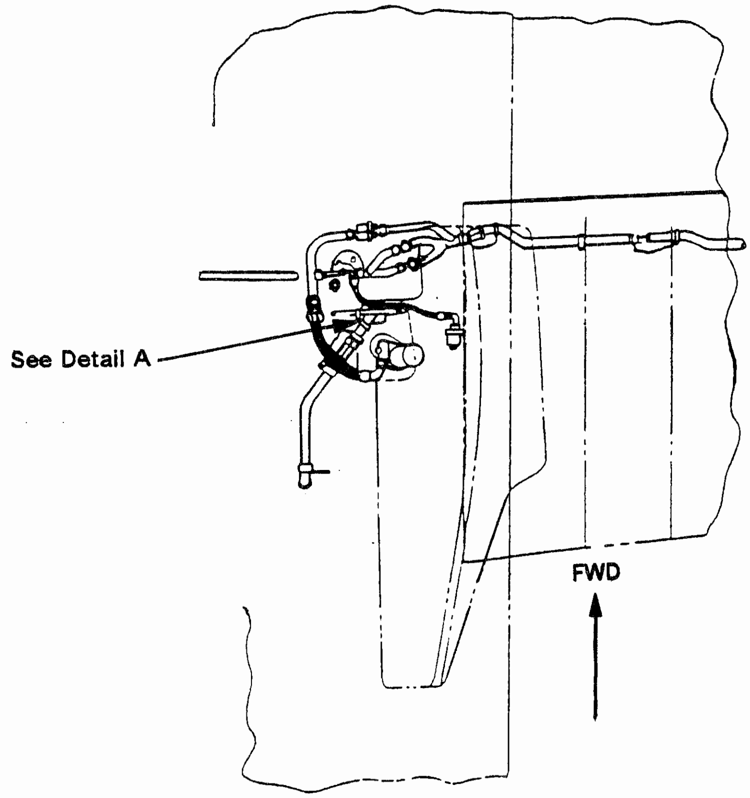
A/C 1-106, and 114 not having ASC 165

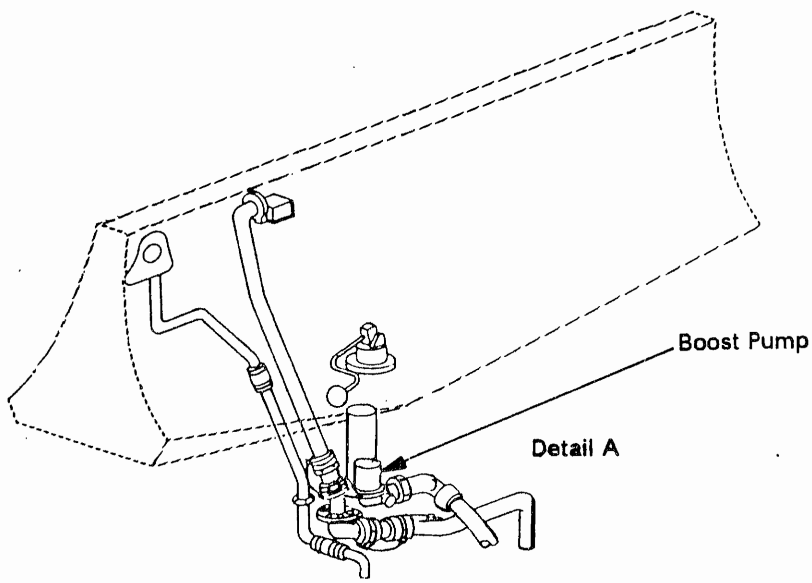


DETAIL A

1. Snap Fastener
2. Water/Methanol Feed Line
3. Mounting Plate
4. Boost Pump
5. Quantity Transmitter
6. Bladder Ce.l
7. Filler Cap
8. Vent Tube
9. Vent Valve

Water/Methanol Boost Pump
(Aircraft not having ASC 125)
Figure 201.





Water/Methanol Boost Pump
(Aircraft having ASC 125)
Figure 202.

WATER/METHANOL INJECTION SYSTEM SHUTOFF VALVES — DESCRIPTION / OPERATION

NOTE: Since the water/methanol shutoff valves and installation are the same for Aircraft not having and having ASC 125, the following information will apply to both.

1. Description

Each electric motor driven, sliding gate, shutoff valve is installed within a protective box and is bracketed to the left side of the nacelle structure. A flag lever and OPEN and CLOSED markings are provided on each valve for visual indication of the sliding gate position. Power for valve comes from essential dc bus 5A W/M CONT circuit breaker. Access to the valves is provided through main landing gear clamshell doors on the left side of each nacelle and by removing protective box cover.

NOTE: Valve P/N AV16B1453B4 is interchangeable with main fuel shutoff and fuel crossfeed valves.

For logistics and spares reasons it is suggested that, upon overhaul of the valve, P/N 16B1453B the valve be brought to the P/N AV16B1453B4 standard. All future procurement of these valves from General Controls will be to this standard only, since this component incorporates a product improvement of General Controls which installs Belleville springs in the actuator assembly which preloads the motor bearing.

The P/N AV16B1453B4 standard is as follows:

- Line Thermal Relief 30-55 psi both directions. Relief Valve Assembly No. 102913 AW.
- Body Relief 30-55 psi. Relief Valve Assembly No. 103852 BF.
- Actuator Assembly No. 102842 ME.

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WATER/METHANOL INJECTION SYSTEM SHUTOFF VALVES — MAINTENANCE PRACTICES

1. Water/Methanol Shutoff Valve — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

BEFORE WORKING IN ANY WHEEL WELL ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Removal

(1) Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(2) Gain access to water/methanol shutoff valve and filter assembly (See Figure 201) through main landing gear clamshell doors in each nacelle. Install ground safety strut in doors.

(3) Remove protective box cover.

(4) Disconnect B-nut and electrical connector from valve.

(5) Remove outlet line from water/methanol filter.

(6) Remove valve and filter assembly from protective box as an assembly.

(7) Remove hardware securing valve, filter assembly and line adapter. Remove and discard O-rings.

NOTE: If pump is to remain out of aircraft for periods in excess of twelve hours, the boost pump must be preserved, see Water/Methanol Pump - Preservation/Depreservation, Section 82-3-0.

B. Installation

(1) Install valve on line adapter and filter assembly, using new O-rings.

(2) Install valve and filter assembly in protective box.

(3) Connect outlet line to filter assembly.

(4) Connect B-nut and electrical connector to valve and safety wire as required.

(5) Inspect area for foreign objects, damage and security of all attachments.

(6) Service water/methanol cell/tanks see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(7) Energize main and essential dc bus.

(8) Bleed water/methanol system, and inspect for leaks during system bleed, see Water/Methanol System - Bleeding, Section 82-0-1.

(9) Perform Water/Methanol Shutoff Valve - Operational Test, see this Section.

2. Water/Methanol Shutoff Valve — Operational Test

WARNING: BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Gain access to water/methanol shutoff valves (See Figure 201) through main landing gear clamshell doors. Install ground safety strut.

B. Remove protective box cover from valve enclosure.

C. Ensure POWER LEVERS are in IDLE.

D. Pull L/R NAC FIRE EXT circuit breakers (Aircraft 1 - 200, 322 and 323 not having ASC 210) or FIRE EXT SHOT #1 and FIRE EXT SHOT #2 (Aircraft 1 - 200, 322 and 323 having ASC 210)

E. Energize main and essential dc bus.

F. Place WATER/METHANOL switch to ARMED, W/M PRESS warning lights should come on momentarily and go out.

G. Check that valve flag moves to OPEN and that there are no leaks.

H. Pull applicable FIRE PULL T-handle. Valve should close.

I. Reset FIRE PULL T-handle. Valve should open.

J. Place appropriate HP FUEL COCK lever to FEATHER, valve should close.

K. Place appropriate HP FUEL COCK lever to FUEL OFF, valve should open.

L. Place WATER/METHANOL switch to OFF. Valve should close.

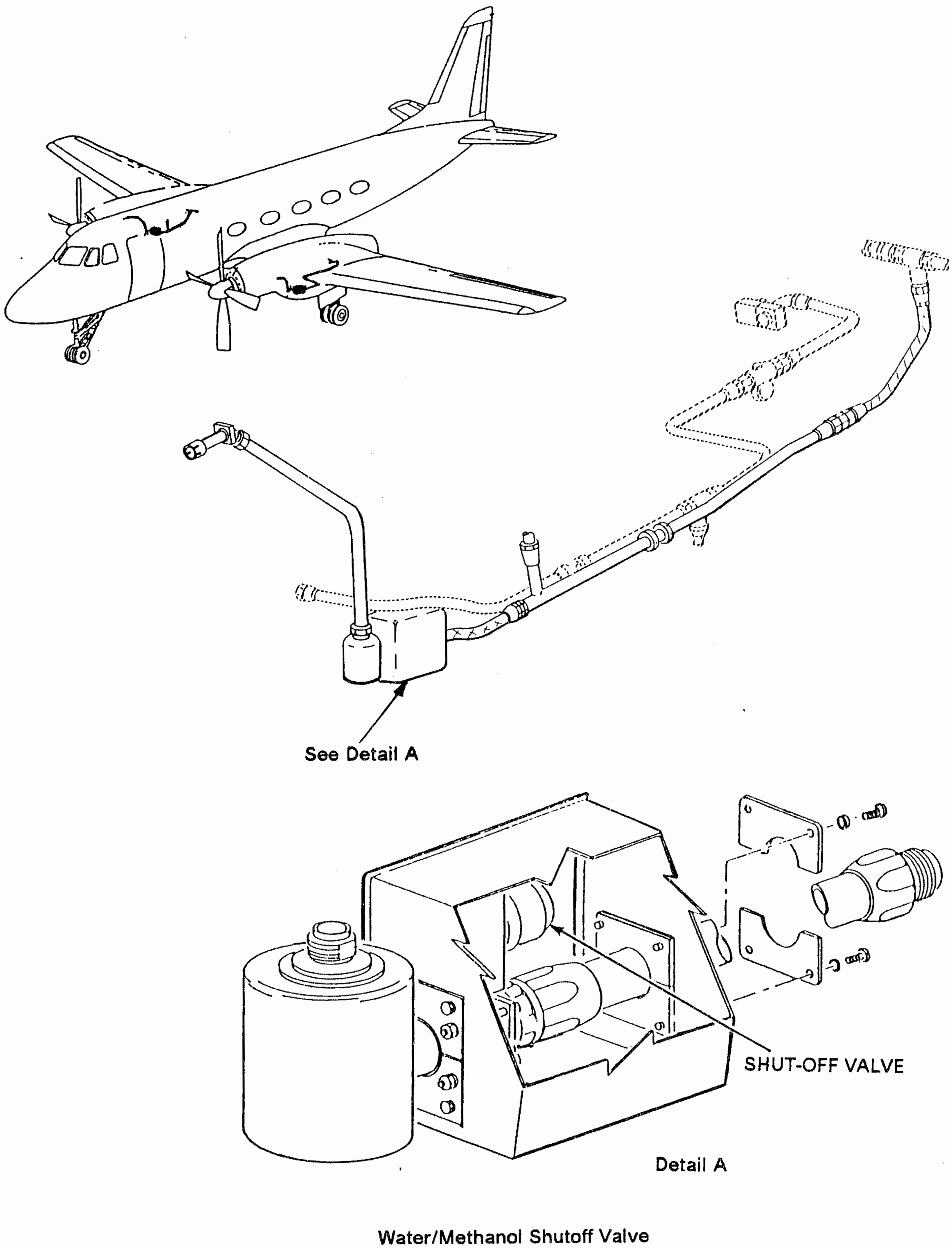
M. Close circuit breakers opened in Step D. above.

N. De-energize main and essential dc bus.

O. Inspect area for presents of foreign objects, damage and security of all attachments.

P. Install protective box cover.

Q. Remove ground safety strut from landing gear doors.



Water/Methanol Shutoff Valve
Figure 201.

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WATER/METHANOL INJECTION SYSTEM FILTERS — DESCRIPTION / OPERATION

NOTE: Since the water/methanol filters and installation are the same for Aircraft not having and having ASC 125, the following information will apply to both.

1. **Description**

The water/methanol system filters are located in the forward section of each main gear wheel well. They are part of the shutoff valve and filter assemblies. The inlet port of the filter is attached to the shutoff valve.

Tubing between the water/methanol cells/tanks and the shutoff valve carries the water/methanol mixture to the filter inlet port.

The filter outlet port is connected by tubing directly to the engine. Each filter contains a stack assembly made of 14 No. 20 micron filters, which trap any contamination which may be in the tanks and prevent it from entering the engine. A spring.loaded ball will unseat and bypass the fluid if the filter should become clogged. Access to the filter is gained by opening the main landing gear clamshell doors.

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WATER/METHANOL INJECTION SYSTEM FILTERS — MAINTENANCE PRACTICES

1. Water/Methanol Filter Assembly — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Removal

(1) Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(2) Install ground safety strut in clamshell doors.

(3) Gain access to water/methanol shutoff valve and filter assembly through main landing gear clamshell doors in each nacelle. (See Figure 201).

(4) Remove protective box cover.

(5) Disconnect B-nut and electrical connector from valve.

(6) Remove outlet line from water/methanol filter.

(7) Remove valve and filter assembly from protective box as an assembly.

(8) Remove hardware securing valve, filter assembly and line adapter. Remove and discard O-rings.

NOTE: If pump is to remain out of aircraft for periods in excess of twelve hours, the boost pump must be preserved, see Water/Methanol Pump - Preservation/Depreservation Section 82-3-0.

B. Installation

(1) Install valve on line adapter and filter assembly, using new O-rings.

(2) Install valve and filter assembly in protective box.

(3) Connect outlet line to filter assembly.

(4) Connect B-nut and electrical connector to valve, and safety wire as required.

(5) Inspect area for foreign objects, damage and security of all attachments.

(6) Service water/methanol cell/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(7) Bleed water/methanol system, and inspect for leaks during system bleed, see Water/Methanol System - Bleeding, Section 82-0-1.

(8) Perform Water/Methanol Shutoff Valve - Operational Test, see Section 82-4-0.

(9) Remove ground safety strut from clamshell doors.

2. Water/Methanol Filter — Inspection

WARNING: WATER/METHANOL IS TOXIC. USE IN A WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON SKIN OR AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

BEFORE WORKING IN ANY WHEEL WELL, ENSURE THAT ALL LANDING GEAR AND LANDING GEAR DOOR SAFETY DEVICES ARE INSTALLED.

A. Install ground safety strut.

B. Gain access to filter assembly (See Figure 201). through main landing gear clamshell doors in each nacelle.

C. Place WATER/METHANOL switch OFF. Ensure water/methanol valves are closed.

D. Provide a suitable container to drain water/methanol from filter assembly.

E. Remove cap from drain fitting and drain filter.

F. Rotate cover assembly 10 - 20° in each direction to loosen cover assembly O-ring.

G. Press in cover assembly and remove retaining ring.

H. Remove cover assembly. If spring-loaded cover will not come out, repeat Step F. above.

I. Thread safety wire through holes provided in bottom of stack assembly.

J. Pull stack assembly from filter housing.

K. Remove O-rings from stack and cover assembly.

L. Clean stack assembly using ultrasonic cleaning method.

M. Inspect for stack assembly for cleanliness, damaged filters, broken or distorted springs and sticking or mutilated check ball.

NOTE: Replace stack assembly or individual parts if defects are evident.

N. Install new O-rings on stack assembly and cover.

O. Install stack assembly in filter housing and remove safety wire.

P. Install cover and retaining ring.

Q. Install drain fitting cap and tighten securely.

R. Inspect area for foreign objects and damage.

S. Energize main and essential bus.

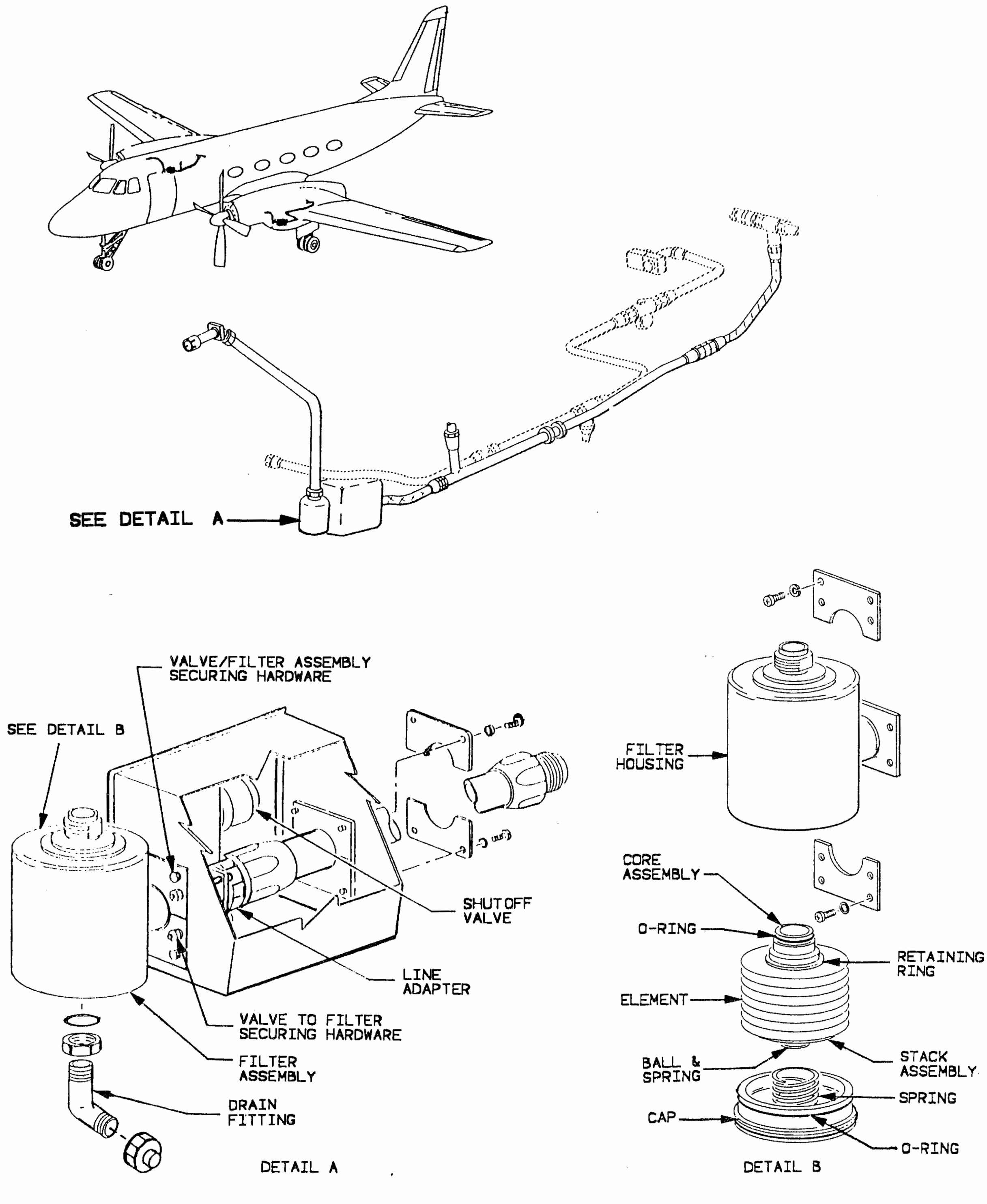
T. Ensure POWER LEVERS are in IDLE.

U. Place WATER/METHANOL switch to ARMED. W/M PRESS warning lights should come on momentarily and go out.

V. Inspect filter assembly for leaks.

W. Place WATER/METHANOL switch to OFF.

X. Remove electrical power from aircraft and remove ground safety struts.



Water/Methanol Filter
Figure 201.

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WATER/METHANOL QUANTITY INDICATING SYSTEM — DESCRIPTION / OPERATION

1. General Description

The water/methanol quantity indicating system provides the crew with indication of the amount of fluid in the cells/tanks. Though two separate types of water/methanol cell/tank systems are installed, their associated indicating systems are similar. Differences between the two configurations will be reflected accordingly through this section.

In utilizing this section, first ascertain whether the aircraft in question has ASC 125 installed or not, and then follow the applicable parts of this section.

2. Description

A. Aircraft not having ASC 125 (increased fuel load)

The quantity indicating system consists of a dual quantity indicator and two float actuated cell quantity transmitter units. The system is supplied power from the 28 volt main dc bus through a 5A circuit breaker located on the left circuit breaker panel. The indicator, mounted in the center instrument panel, provides for visual indication of water/methanol level in each cell. The indicator dials are provided with the markings reading EMPTY 1/4, 1/2, 3/4 and FULL. A float type quantity transmitter is mounted adjacent to the water/methanol boost pump in each wing system. Access to transmitter is made through access holes in outer wing panel upper and lower access panels between Fuselage Station 313 and 331.

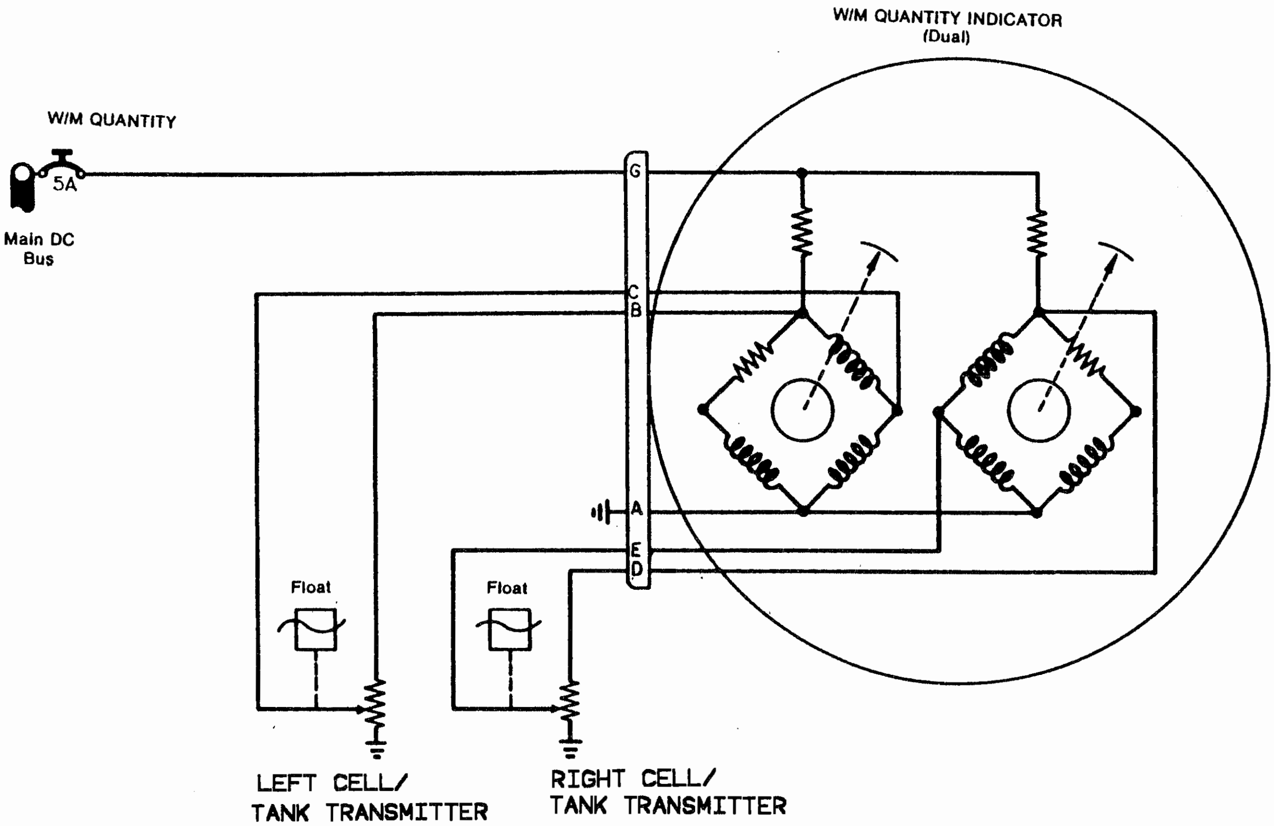
B. Aircraft having ASC 125 (increased fuel load)

The quantity indicating system consists of a dual quantity indicator and two float actuated tank quantity transmitter units. The system is supplied power from the 28 volt main dc bus through a 5A circuit breaker located on the left circuit breaker panel. The indicator, mounted in the center instrument panel, provides for visual indication of water/methanol level in each tank. The indicator dials are provided with the markings reading EMPTY, 1/4, 1/2, 3/4 and FULL. A float type quantity transmitter is mounted at Fuselage Station 381.5 and approximately BL 48. Access to the transmitter is through the access panel around the landing lights on the underside of fillet area. The units are located next to the pump in each integral tank.

3. Operation

NOTE: Operation of quantity system is identical for all aircraft.

System operation is based on the voltage change induced by a float operated potentiometer type transmitter. As the float rises or lowers, output voltage will vary accordingly. (See Figure 1) This output voltage is transmitted to quantity indicator by two wires. The indicator consists of a pointer rotor which is surrounded by three coils. The varying voltages transmitted to these coils cause the magnetic field flux produced around each coil to change, thus positioning the pointer rotor. A magnet is installed within the indicator to pull the pointer off scale when system is de-energized. Power comes from the main dc bus 5A W/M QUAN circuit breaker. As a result, the system is inoperative in emergency dc operation.



Quantity Indicating System — Schematic
Figure 1.

WATER/METHANOL QUANTITY INDICATING SYSTEM — MAINTENANCE PRACTICES

1. Water/Methanol Quantity Indicator — Removal / Installation

CAUTION: ENSURE ELECTRICAL POWER IS OFF.

A. Removal

(1) Remove power from aircraft.

(2) Remove electrical connector.

(3) Remove indicator.

B. Installation

(1) Inspect connector and general area for foreign objects and damage.

(2) Connect electrical connector.

(3) Install indicator.

(4) Apply electrical power to aircraft.

(5) Check for proper water/methanol quantity indication, based on amount of water/methanol mixture visibly checked in cells/tanks.

NOTE: Water/Methanol Quantity Indication - Operational Test should be accomplished the next time cells/tanks are empty.

2. Water/Methanol Quantity Indication — Operational Test

WARNING: WATER/METHANOL IS TOXIC. USE IN WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

B. Energize main and essential dc bus.

C. With water/methanol cells/tanks empty, indicator should read empty.

D. Add approved water/methanol mixture to each cell/tank in amounts specified and note indicator readings as follows:

A/C 1-106 and 114 NOT HAVING ASC 165	A/C 1-106 and 114 HAVING ASC 165 and 107-200, 322 and 323	A/C 1-200, 322 and 323 HAVING ASC 125A	INDICATOR READING
10 1/2 GAL.	5 3/4 GAL.	6 GAL.	1/4
21 GAL.	11 1/2 GAL.	11 7/8 GAL.	1/2
31 1/2 GAL.	17 1/2 GAL.	17 3/4 GAL.	3/4
42 GAL.	23 GAL.	23 3/4 GAL.	FULL

E. Remove power from aircraft.

3. Water/Methanol Quantity Transmitter — Removal / Installation

WARNING: WATER/METHANOL IS TOXIC. USE IN WELL VENTILATED AREA AND AVOID INHALATION OF FUMES OR CONTACT WITH SKIN. IF LIQUID IS SPILLED ON AIRCRAFT, IMMEDIATELY FLUSH WITH FRESH WATER.

A. Removal

(1) Drain water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(2) Remove electrical power from aircraft.

(3) Gain access to transmitter (See Figure 201 or Figure 202) as follows:

(a) On Aircraft not having ASC 125, remove lower wing access panel 21/22-LWR-1.

(b) On Aircraft having ASC 125, remove access panel 107-LH FIL-2 or 107-RH FIL-2 as appropriate.

(4) Disconnect electrical leads at Dage connectors.

(5) Disconnect bonding jumper.

(6) Remove transmitter.

NOTE: If pump is to remain out of aircraft for periods in excess of twelve hours, the boost pump must be preserved, see Water/Methanol Pump - Preservation/Depreservation, Section 82-3-0.

B. Installation

(1) Install transmitter with new gasket.

(2) Connect bonding jumper and electrical leads.

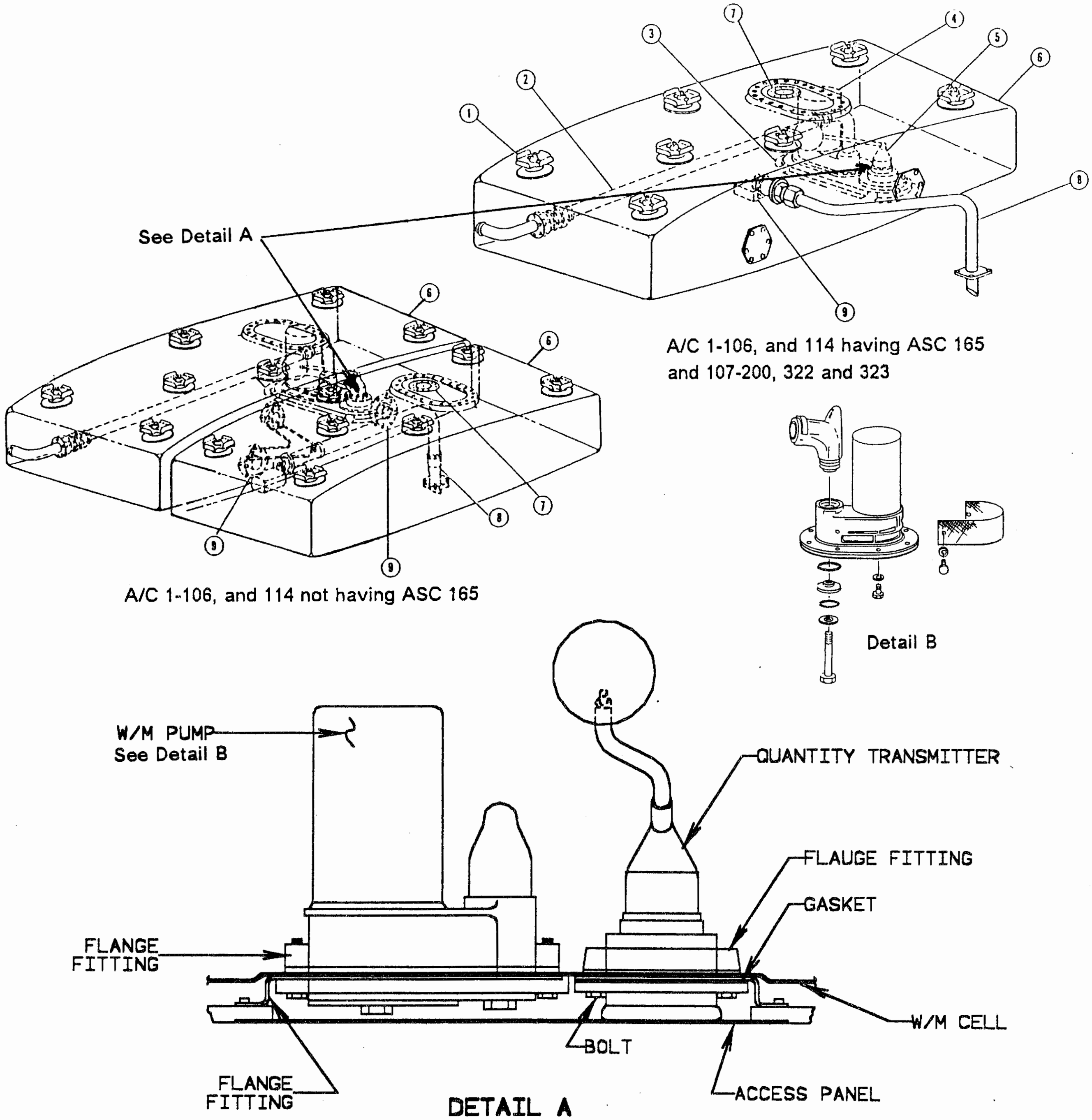
(3) Service water/methanol cells/tanks, see Water/Methanol System - Draining/Servicing, Section 82-0-1.

(4) Perform Water/Methanol Quantity Indication - Operational Test and inspect for leaks as cell/tank is being serviced, this Section.

NOTE: After cells/tanks are serviced to 1/2 full, select WATER/METHANOL switch to ARMED and check quantity transmitter for leaks.

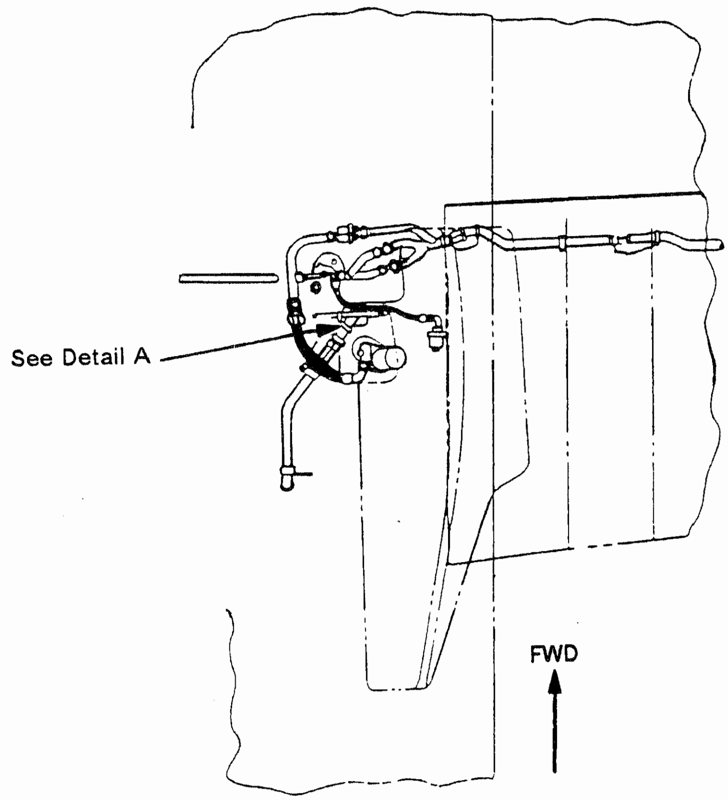
(5) Inspect area for foreign objects, damage and security of all attachments.

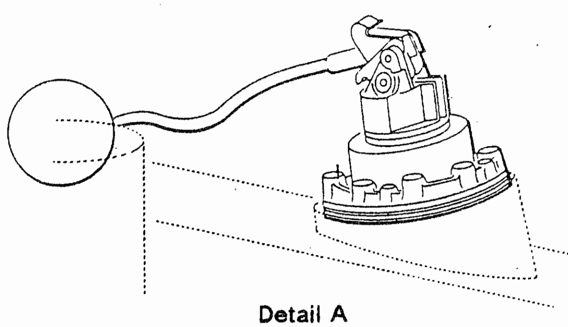
(6) Install access panels.



1. Snap Fastener
2. Water/Methanol Feed Line
3. Mounting Plate
4. Boost Pump
5. Quantity Transmitter
6. Bladder Cell
7. Filler Cap
8. Vent Tube
9. Vent Valve

Water/Methanol Quantity Transmitter
(Aircraft not having ASC 125)
Figure 201.





Detail A

Water/Methanol Quantity Transmitter
(Aircraft having ASC 125)
Figure 202.

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ACCESSORY GEARBOX

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CHAPTER 83

ACCESSORY GEARBOX

ACCESSORY GEARBOXES — DESCRIPTION / OPERATION

1. General

The accessory gearbox installations are basically similar, the main difference being the addition of a propeller brake unit to the left gearbox assembly and a blower assembly (supercharger) to the right gearbox. The accessory gearbox consists primarily of a high strength, light alloy main casing and front cover containing trains of straight spur gears and drives to operate the various accessories. Except for the generator and hydraulic pump (both have adapter and shear point built in) the final drive to each accessory is provided by a quill incorporating a safety shear-neck which protects the gearbox from damage in the event of a seizure or failure of the accessory. The system of quills and shear-necks is such that seizure of any one accessory will cause its associated shear-neck to break without damage to any other component. The momentary sum of the shear-neck break torque and the full load torque of all remaining accessories will not overstrain any part of the drive shaft system or the engine power takeoff. It is ensured that the seizure of any individual accessory can occur without interfering with the continued operation of any other accessory.

Again, the system is such that should a complete seizure of the entire gearbox occur, the shear-neck provided in the input drive shaft, will break at a torque within the static torque capacity of the engine power takeoff, and also within the proof torque capacity of any other component in the input drive shaft.

Means are provided which ensure that neither a single accessory seizure, nor a complete gearbox seizure will interfere with the continued operation of the engine.

The front cover contains the fillercap and has adapter pads for the mounting of a generator, a hydraulic pump, a synchronizer alternator, and the drive shaft housing. The main casing has provisions for mounting an alternator, a tachometer generator and the blower assembly (supercharger).

Accessory drive gearbox is mounted aft of the firewall and has a three point mounting comprised of two rear mounting pads on each side of the main casing. Each rear mounting pad has two dowels and four 7/16 inch BSF tapped holes and a single forward mounting comprised of a bracket, housing a Ferrobestos bushing, which supports the forward end of the input drive tunnel providing fore and aft location. Mounting brackets to fit these pads are supplied by Gulfstream Aerospace. They provide positive transverse location on one side only, leaving the other side free to move transversely. The function of the accessory gearboxes is to provide mounting bases for the various aircraft accessories and through the gearbox drive, gears and quills, transmit the necessary engine power required to drive them.

The left installation, as shown in Figure 1, comprises a gearbox type PTG.14/7 or PTG.14/30, and gearbox drive type GD.20/3 driving the Dowty Rotol synchronizer alternator, the N.Y. Airbrake hydraulic pump, the Jack and Heintz or Bendix 9 KW dc generator, the Smiths tachometer generator, the Lucas Rotax 11 KVA alternator. The Dunlop propeller brake unit is installed on Aircraft 1 - 148, 322 and 323 only.

The right installation, as shown in Figure 2, comprises a gearbox type PTG. 14/8 or PTG. 14/31, and a gearbox drive type GD.20/3 driving the Dowty Rotol synchronizer alternator, the N.Y. Airbrake hydraulic pump, the Jack and Heintz or Bendix 9 KW dc generator, the Godfrey blower type 15 MK1452A, the Smiths tachometer generator, and the Lucas Rotax 11 KVA alternator.

Gearboxes type PTG. 14/30 and 14/31 (Dowty Rotol GB 2483) provide for continuous gearbox oil lubrication of the hydraulic pump final drive gear shafts, as apposed to type PTG. 14/7 and PTG. 14/8 gearboxes where the pump shafts were lubricated on installation and requires lubricating at periodic intervals. The new drive shaft installed (in accordance with New York Air Brake Bulletin 109A) on the hydraulic pump, incorporates a seal which prevents gearbox oil from migrating into hydraulic pump.

The following table is a list of the ratio of the accessory drive to gearbox input drive and to engine rpm and direction of rotation:

Accessory	Ratio to Input Drive	Ratio to Engine Drive	Rotation Looking at Pad
Sync alternator	1.0780	0.3717	clockwise
Hydraulic pump	0.7700	0.2655	clockwise
9 Kw DC generator	1.6448	0.3672	clockwise
Godfrey blower	2.3329	0.8044	counterclockwise
Tach generator	0.7250	0.2500	clockwise
11 KVA alternator	1.5922	0.5491	counterclockwise
Input drive shaft	----	0.3448	clockwise

The gearbox drive is the double universal joint type. The universal joint is fitted to compensate for minor errors in alignment between engine and gearbox, while axial movement is allowed for by a serrated joint fork which is free to slide in a serrated shaft. The complete drive assembly is dynamically balanced to ensure vibration free running. The major portion of the power absorbed by the equipment is demanded by the Godfrey blower, Lucas Rotax 11 KVA alternator, Jack and Heintz or Bendix dc generator and N.Y. Airbrake hydraulic pump. In addition to these, the Dowty Rotol synchronizer alternator and the Smiths tachometer generator demand negligible power.

All these accessories are included in the right installation, the left installation being the same, except for the omission of the blower assembly and the addition of a Dowty Rotol propeller brake (Aircraft 1 -148, 322 and 323) which forms part of the gearbox type PTG.14/7.

2. **Alternator Adapter/Attachment Ring**

(See Figure 3)

The alternator adapter provides for the mounting of an alternator utilizing the Manacle Clamp type attachment flange. The adapter is ring shaped, flanged and spigoted to fit the large bore in the rear wall of the main casing. Eight nuts and plain and spring washers secure the adapter to the casing. The rear of the adapter contains an undrilled tapered flange, which forms a seat for the alternator attachment ring. A locating peg screwed into the flange insures correct positioning of the alternator. This adapter is a permanent part of the gearbox. It is not removed during disassembly. The attachment ring (manacle clamp type) secures the alternator to the alternator adapter. It consists of eight links, a groove is machined to fit over the tapered edges of the alternator and alternator adapter. The ring of links is secured with a plunger unit, nut, and link bolt.

CAUTION: REPAIR OF ATTACHMENT RINGS BY MERELY CHANGING LINKS IS NOT RECOMMENDED.

3. **Generator Adapter/Attachment Ring**

(See Figure 3)

The generator adapter is a ring shaped light alloy casing, spigoted to seat in the gearbox adapter. The mating flange is similar to that of the gearbox adapter, it is tapered to conform to the attachment ring. Six studs on the front machined face of the generator adapter enable its attachment to the generator. Attachment ring is similar to the alternator attachment ring, except that the component parts are smaller.

4. **Dunlop Propeller Brake**

Installed on Aircraft 1-148, 322 and 323 only. (See Figure 3 and Figure 4 only)

This hydraulically operated, caliper type brake unit is installed to prevent windmilling of the propeller after the left engine is shut down as a safety feature for the protection of the passengers and flight personnel and to expedite boarding of and alighting from, the aircraft. The unit consists of two opposing piston actuated brake pads in a caliper body. The body also houses four return springs, a feed pipe and an air bleed fitting. The propeller brake handle is located on the forward end of the console. Pulling the propeller brake handle up will permit operating fluid to be supplied from the auxiliary hydraulic system or the accumulator system, simultane-

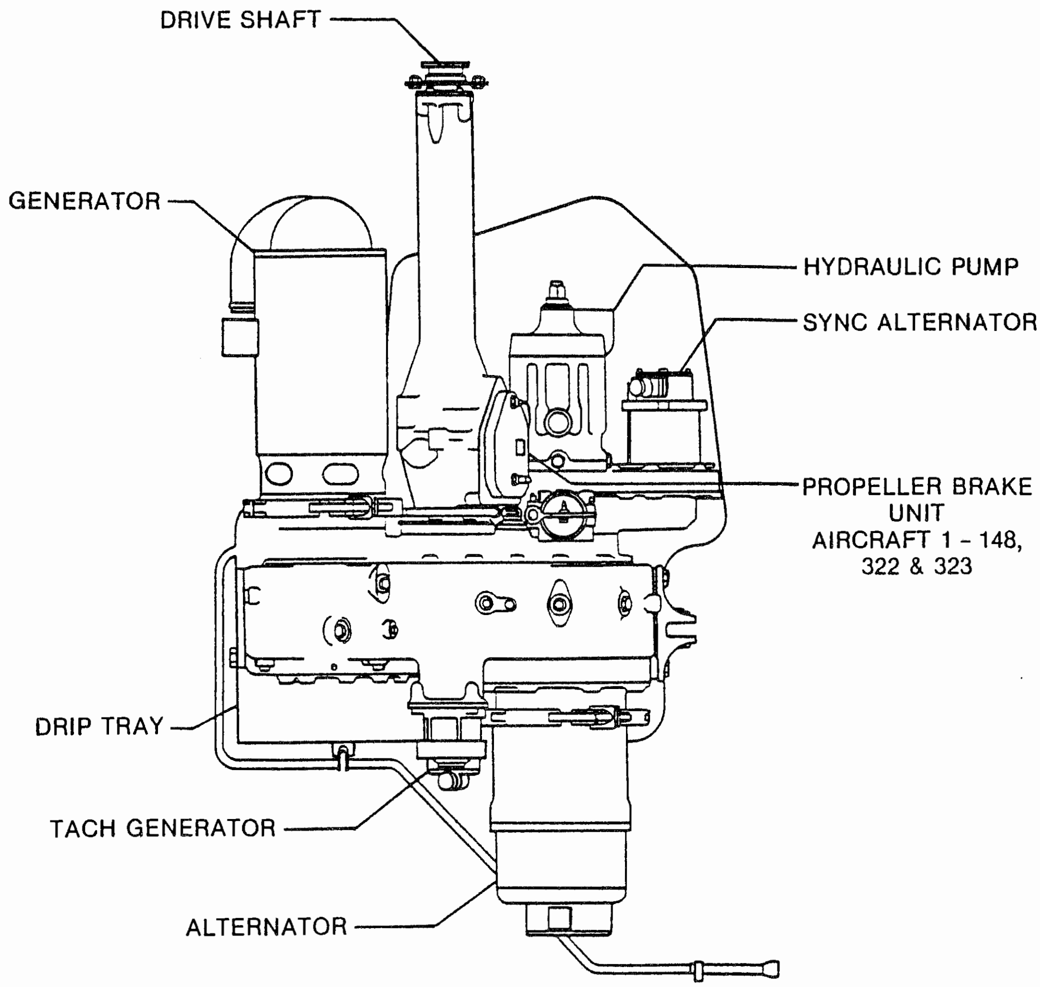
ously to the rear of each piston through the feed pipe and interconnecting drilled passages in the cylinder blocks. This causes the pistons to move against the pressure of the return springs, carrying with them the pad assemblies which then clamp the revolving brake disc applying the brake. Upon release of of operating pressure, the return springs return the pistons and pad assemblies to the off position. To release the brake, return the propeller brake handle to its stowed position.

CAUTION: DO NOT APPLY PROPELLER BRAKE UNTIL ENGINE RPM IS BELOW 2000.

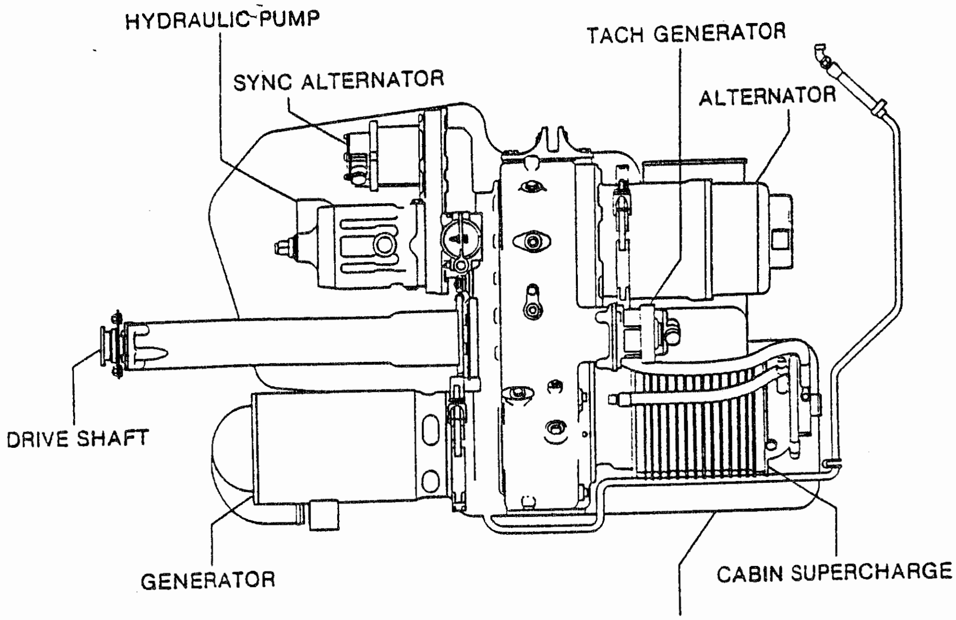
A switch operated by the propeller brake handle prevents starting of the left engine if the propeller brake is on. This switch is in series with the left engine ground starting system and allows only the starter system to activate if the propeller brake handle is in the off (in) position (See Starting, Chapter 80 for switching circuit.

5. Gearbox Low Pressure Warning System

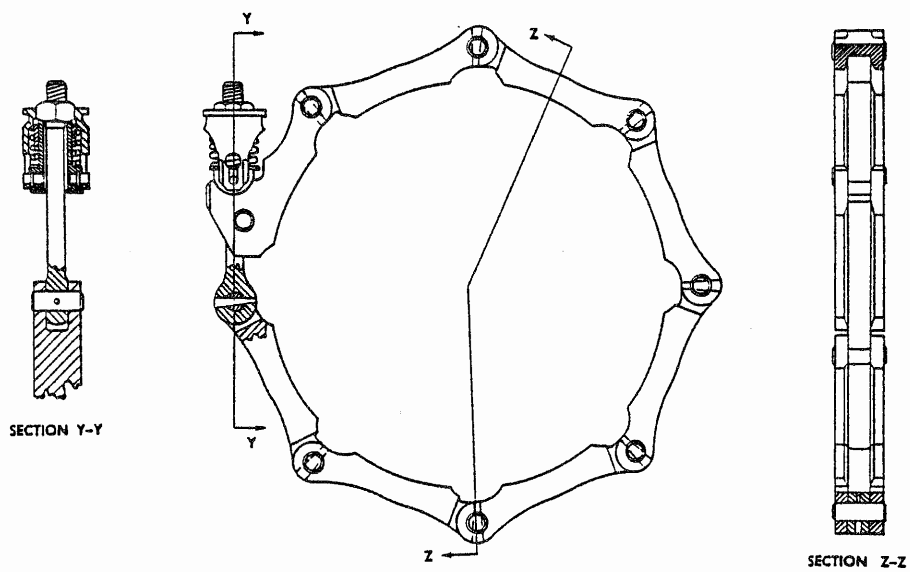
This is an optional installation on Aircraft 1 - 86 and 114. It is standard equipment on Aircraft 87 - 200, 322 and 323. This system is covered in detail in an other section of this chapter.



Left Gearbox Assembly
Figure 1.

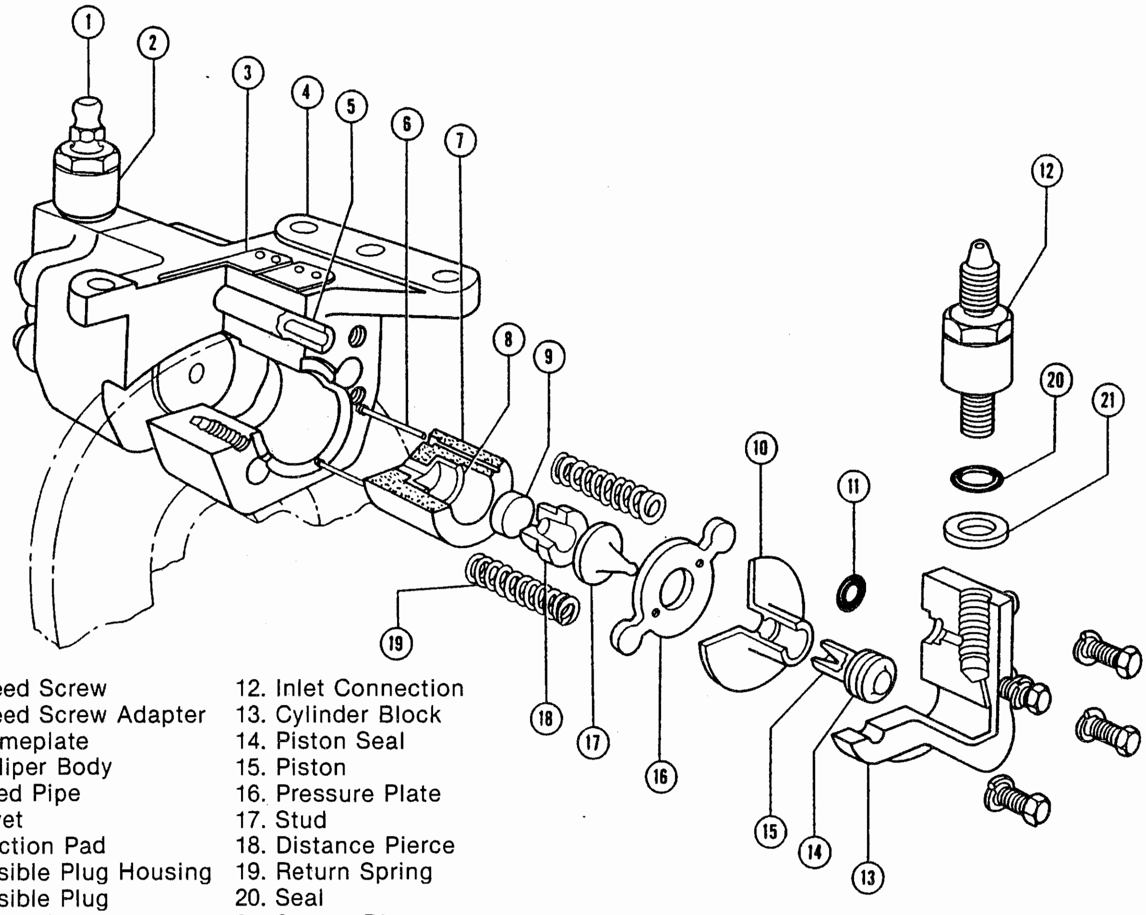


Right Gearbox Assembly
Figure 2.



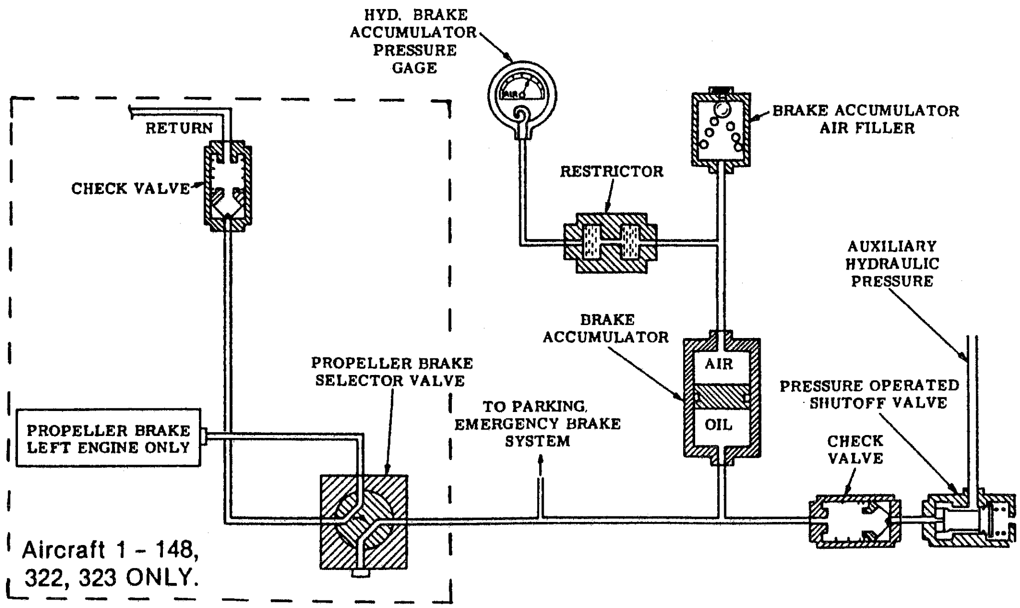
NOTE: The generator attachment ring components are smaller than the alternator attachment rings.

Alternator Attachment Ring
Figure 3.



1. Bleed Screw
2. Bleed Screw Adapter
3. Nameplate
4. Caliper Body
5. Feed Pipe
6. Rivet
7. Friction Pad
8. Fusible Plug Housing
9. Fusible Plug
10. Guide Plate
11. "O" Ring
12. Inlet Connection
13. Cylinder Block
14. Piston Seal
15. Piston
16. Pressure Plate
17. Stud
18. Distance Pierce
19. Return Spring
20. Seal
21. Spacer Ring

Propeller Brake Unit
Figure 4.



Propeller Brake — Hydraulic Schematic
Figure 5.

ACCESSORY GEARBOXES — MAINTENANCE PRACTICES

1. **General**

The accessory gearboxes require very little servicing other than occasional topping with oil, periodic greasing of the universal joints and serrations of the gearbox drives, lubrication of the Ferrodo Ferrobestos brush supporting the front end of the drive shaft housing and on Aircraft not having ASC 201. each gearbox mounting link (2 places). Oil is supplied through a filler near the top of the front cover. Level is checked by a graduated dipstick which passes through the top wall of the main casing and is retained by a bayonet fitting arrangement or a double-leaf spring retainer assembly. A centrifugal breather system allows satisfactory breathing of the gearbox under all conditions of flight.

A. Gearbox Oil Change

If it becomes necessary to drain the gearbox during service, unlock and remove both plugs (one near each bottom corner of the main casing front wall), and permit all oil to drain into a drip pan using a hose and suitable container.

B. Gearbox Driveshaft Lubrication

NOTE: Maintain pressure on lubricating gun until clean grease is visible around the oil sealing ring between the spider shoulder and the bearing cup on all the arms. If this does not occur, the drive shaft must be removed, disassembled, inspected and cleaned. If it is in satisfactory condition, it can be installed.

Check gearbox drive universal joints for play or wear

Dowty Rotol recommends Marfak HD3, Aeroshell 5 or Andok 260 in place of DTD 825 (MIL-G-3278A). However, the grease (MIL-G-3278A) in a new or overhauled shaft must be removed before the introduction of Marfak HD3 or Andok 260. See Chapter 12 or Dowty Rotol Maintenance Manual 865A/1 for approved lubricants.

Dowty Rotol has issued Mod GB1917 which will be stamped on the D/S modplate so that units lubricated with Aeroshell 5, Andok 260 or Marfak 3 HD may be identified.

Grease is to be pumped into the joint until fresh grease oozes out at each of the four seals. Should the new grease fail to exude from any of the seals, the indication is that the existing grease has hardened. In such cases, the gearbox drive is to be removed and the universal joints dismantled. All traces of the old grease is to be removed, and if the joint is satisfactory, it may be rebuilt, filled with fresh grease and returned to service. To avoid disturbing balance, all component parts are to be rebuilt in the positions which they originally occupied.

To prevent a service center from inadvertently mixing non-compatible greases, it is recommended that all operators enter the type of grease they use on their universal joints in the log books. (Mod GB1917)

C. Lubricants.

See Chapter 12 of this manual or to Dowty Rotol Maintenance Manual 865A/1 approved lubricants.

D. Oil Servicing

See Chapter 12 of this manual or to Dowty Rotol Maintenance Manual 865A/1 approved lubricants.

The oil level in the gearbox must be checked at the times specified in Chapter 5. To add oil proceed as follows:

E. Inspection / Check

NOTE: Whenever an accessory is removed from a gearbox, the drive shaft should be inspected, the fit to the mating drive checked and the drive shaft immersed in clean gearbox oil before reassembly.

An accessory drive shaft which has been, or is suspected of having been, subjected to an overload must not be returned to service, even through it passes the normal overload inspection.

Refer to Dowty Rotol Maintenance Manual 865A/1 for Inspection/Check requirements.

2. **Accessory Gearbox — Removal / Installation**

CAUTION: ENSURE ELECTRICAL POWER IS OFF.

AN ACCESSORY DRIVE SHAFT WHICH HAS BEEN, OR IS SUSPECTED OF HAVING BEEN, SUBJECTED TO AN OVERLOAD MUST NOT BE RETURNED TO SERVICE, EVEN THROUGH IT PASSES THE NORMAL OVERLOAD INSPECTION.

A. Removal

(1) Open top cowling on engine.

(2) Remove nacelle accessory section top access panel.

(3) Disconnect hydraulic lines and drain line at hydraulic pump, install protective caps on lines and fittings.

(4) Disconnect fire extinguisher line from gearbox drive shaft housing.

(5) Disconnect drain lines from gearbox casing, install protective caps on lines and fittings.

(6) If installed, disconnect hydraulic line from prop brake.(left nacelle only)

(7) Disconnect blower drain line. (Right gearbox only)

(8) Remove tach generator from gearbox with electrical connectors attached, secure and stow clear of gearbox, retain quill shaft.

(9) Disconnect electrical connections and cooling air ducts from alternator. (right gearbox only)

(10) Tag and disconnect electrical leads from dc generator terminal block.

(11) Disconnect electrical connector from generator overheat switch.

(12) Disconnect generator cooling air ducts at couplings nearest to generator.

(13) Disconnect electrical connector from synchronizer alternator.

(14) Disconnect fire detector wire at left aft bulkhead connector. Discard copper sealing washer and cap fittings.

NOTE: Steps (15) thru (18) applicable to right gearbox only.

(15) Remove bolts attaching blower outlet duct to blower and pull duct clear of blower.

(16) Remove sleeve and clamps from blower inlet duct as follows:

NOTE: Blower and inlet duct are connected and remain with gearbox during removal.

(a) Loosen screws on two worm clamps on inlet duct and remove clamps.

(b) Slide sleeve on inlet duct.

(17) Disconnect short flex line to differential pressure sensor at blower inlet duct fitting.

(18) Disconnect link supporting drip tray from rear of blower.

(19) Disconnect differential pressure sensor line from sensor, cap fitting.

(20) Connect hoisting sling to gearbox, take up slack.

(21) Remove and retain bolts and washers attaching gearbox mounting bracket to support assemblies on left and right sides of gearbox.

(22) Remove nuts. washers and bolts that support drive shaft assembly at aft side of firewall.

(23) Loosen bolts at firewall plate.

(24) Remove nuts. washers and bolts that secure gearbox mounting bracket to support assemblies.

(25) Raise gearbox slightly, if installed, disconnect gearbox low oil pressure switch line at gearbox.

(26) Carefully remove gearbox from aircraft.

NOTE: At operators discretion, 600 hour replacement of blower outlet duct seals may be performed at this time.

B. Installation

NOTE: Monitor accessory gearbox oil pressure during initial engine run.

If a replacement gearbox is being installed, following equipment should be installed prior to gearbox installation. (See appropriate Chapter for procedure)

LEFT GEARBOX	RIGHT GEARBOX
Prop brake (if applicable)	Hydraulic pump
Hydraulic pump	DC generator
Alternator	Synchronizer Alternator
DC generator	Gearbox mounting brackets.
Synchronizer alternator	Blower
Gearbox mounting brackets	

(1) Inspect gearbox support link, bearings, and attaching hardware for excessive wear. Replace worn components.

(2) Lubricate Ferrodo bushings at forward end of drive shaft with graphite grease. (See Chapter 12 for approved lubricant)

(3) Carefully place gearbox into position with Ferrodo bushing plate in full aft position.

(4) With gearbox slightly raised, connect low oil pressure switch line to gearbox (if installed).

(5) Install and secure bolts, washers and nuts attaching gearbox bracket to support assemblies (left and right sides).

(6) Align driveshaft with gearbox input driveshaft. Install bolts, washers and nuts in drive shaft assembly side of firewall.

(7) Install and tighten all nuts and bolts at firewall plate. Bolt heads should be located in engine compartment.

(8) Torque nuts securing driveshaft together, to 95-100 inch pounds. Lock tab washers.

(9) Remove sling from gearbox.

(10) Lubricate support link bearings, two points. (See Chapter 12 for lubricant (Aircraft w/o ASC 210)

(11) Connect electrical connector to synchronizer alternator.

(12) Install new copper sealing washer and connect fire detector wire at bulkhead behind blower installation torque to 50-70 inch pounds and safety wire.

(13) Connect link supporting drip tray from rear of blower.

(14) Connect rigid differential pressure sensor line to sensor at firewall.

(15) Connect short flex line to differential pressure sensor at blower inlet duct fitting.

(16) Connect blower drain line.

(17) Install blower outlet duct on blower.

(18) Install blower inlet ducting to inlet scoop as follows:

(19) Slide sleeve over inlet duct and scoop duct.

(20) Position two worm clamps over sleeve and tighten screws.

NOTE: Inlet duct is attached to blower and needs only to be attached to scoop duct.

Steps (13) thru (18) applicable to right gearbox only.

(21) Connect cooling air ducts to dc generator.

(22) Remove tags and connect electrical leads to generator terminal block. Torque hardware to: Large leads to 95-110 inch-pounds and Small leads to 30-40 inch-pounds

(23) Connect electrical connector to generator overheat switch.

(24) Install alternator.

(25) Connect cooling air duct and electrical connectors to alternator.

(26) Lubricate tach generator quill shaft splines with gearbox oil, install tach generator and torque nuts to 95-110 inch pounds.

(27) If installed, connect hydraulic line to prop brake. (Left gearbox only)

(28) Connect drain lines to gearbox casing.

(29) Connect fire extinguisher line to gearbox drive shaft housing.

(30) Connect hydraulic line and drain line to hydraulic pump.

(31) Fill gearbox with six pints of oil. Secure cap and dip stick.

(32) Bleed propeller brake unit, if installed, see this Section. (Left gearbox only)

WARNING: DO NOT OPERATE ENGINE ABOVE IDLE SPEEDS WITH COWLING/ACCESS PANEL OPEN.

(33) Check for leaks at engine idle.

(34) Perform operational test on all systems that was disturbed. See appropriate Section for procedures.

(35) Check gearbox oil level and Service if needed.

(36) Inspect area for presents of foreign objects, leaks and security of all attachments.

(37) Secure all cowlings and access panels.

3. Accessory Gearbox Drive Shaft — Removal / Installation

A. Removal

(1) Gain access to accessory gearbox drive shaft by opening top engine cowl.

(2) Remove nuts and washers securing bell housing over forward universal coupling and slide bell housing aft.

(3) Unlock tab washers and remove nuts, tab washers and bolts that secure drive shaft universals to gearbox coupling and engine power coupling flange.

NOTE: Discard tab washers.

B. Installation

(1) Inspect and lubricate drive shaft universals.

NOTE: Install drive shaft with splined shaft end toward engine.

(2) Connect drive shaft universals to gearbox coupling and engine power coupling flanges and secure with nuts, tab washers and bolts.

(3) Torque nuts to 95-110 inch-pounds. Lock tab washers.

(4) Install bell housing and secure with nuts and washers.

(5) Inspect area for presents of foreign objects, security of all attachments.

(6) Close top engine cowl.

4. Left Propeller Brake Assembly — Removal / Installation

A. Removal

(1) Remove left engine accessory section access panel.

(2) Disconnect hydraulic line from assembly.

(3) Remove assembly.

B. Installation

(1) Install assembly using hardware previously removed, connect hydraulic line.

(2) Bleed brake unit see this Section.

(3) Inspect area for presents of foreign objects, security of all attachments and damage.

(4) Check operation of brake unit as follows:

(a) Apply electrical power to aircraft.

(b) Perform engine crank cycle.

(c) After de-selecting crank cycle apply propeller brake. Ensure propeller slows fairly rapid.

(d) Reset propeller brake handle and remove electrical power

(e) Inspect area for presents of foreign objects, security of all attachments.

(f) install accessory section access panel.

5. Dunlop Propeller Brake Assembly — Priming / Bleeding

(Aircraft 1 - 148, 322 and 323.)

NOTE: On installation, or after any operation in which the hydraulic system has been disturbed, the brake unit should be bled.

Protect friction pads from contamination and from discharging fluid.

A Rubber tube should be attached to the bleed screw for the bleeding operation. The tube should be long enough to direct the discharging fluid away from the brake unit.

A. Remove accessory section access panel.

B. Apply electrical power turn on auxiliary hydraulic pump.

C. Slowly loosen bleed screw.

D. Prime and bleed brake unit by operating brake control lever several times.

E. Tighten bleed screw when discharging fluid if free of air.

F. Turn off auxiliary hydraulic pump and remove power from aircraft.

G. Inspect area for presents of foreign objects, security of all attachments and damage.

H. Install accessory gearbox access panel.

6. Generator and Alternator Attachment Ring Assemblies Repair

Repair/replacement of worn or damaged links on the generator attachment ring or on the alternator attachment ring are to be accomplished in accordance with Dowty Rotol Accessory Gearbox Overhaul Manual 83-10-3 and Dowty Rotol Accessory Drive Repair Manual 925.

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(5) Check Dunlop brake unit for damage and for security of attachment.
(6) Remove tachometer generator and lubricate with oil, Specification D. ENG. RD. 2487. Remove generator and lubricate drive splines with grease, Speicification MIL-G-3278A.
(7) Remove hydraulic pump quill shaft and lubricate with grease, Specification MIL-G-3278A.

G. After 150 Hours.
Remove and inspect equipment.

H. Every 300 Hours.
(1) Make pressure check on oil low-pressure switch (if ASC 68A is installed).
(a) Disconnect pressure sensing line at switch and cap sensing line, by working through accessory gearbox access panel.
(b) Connect a tee (AN804-4), with a nut (AN924-4), to switch.
(c) Connect 0 to 30 psi direct reading pressure gage to one leg of tee and connect pressure source to other leg.
(d) Energize main and essential dc buses; low oil pressure warning lights and master caution light glow.
(e) Apply air pressure to switch; respective warning lights must go out at reading of 23 ± 1 psi.
(f) Continue to increase pressure to 26 psi.
(g) Decrease pressure until warning lights glow, at or before reading of 20 psi.

10. **Generator and Alternator Attachment Ring Assemblies Repair**

The following repair is recommended for the replacement of worn or damaged links on the generator attachment ring (ROTOL part No. G25796) or on the alternator attachment ring (ROTOL part No. G18469).

A. Inspect ring assembly to ascertain that it is suitable for repair.
B. Remove taper and fulcrum pins at both ends of defective link(s), and remove link(s) from assembly.
C. Assemble new link(s) to ring assembly, and install new fulcrum pins, is necessary. (See figure 201)
D. Check ring assembly on appropriate bluing plate gage. (See chart).

NOTE: Contact between the ring assembly and gage must be 100 percent. If this does not occur, substitute new link(s) until 100 percent contact is obtained.

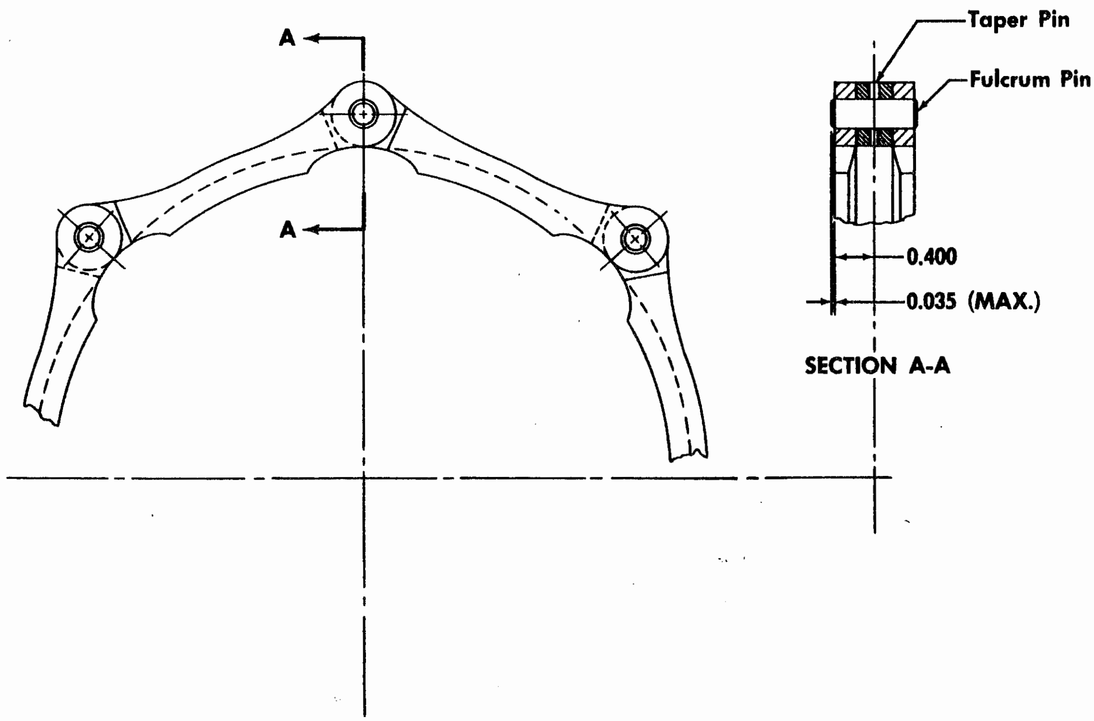
RELATED COMPONENTS AND BLUING PLATES

GENERATOR AND ALTERNATOR ATTACHMENT RING ASSEMBLIES

RING ASSEMBLY	ASA NO.	LINKS P/N	LINKS QTY	TAPER PINS P/N	TAPER PINS QTY	FULCRUMS PART NO.	BLUING PLATES* GAGE NO.
Alternator G18469	ASA 1447/2	G18462 G18463	7 1	SP. 29/E8	8	G18464/1	GT103235/1
Generator G25796	ASA 1447/3	G24198 G24199	6 1	SP.29/E6	7	G18573/1	GT103235/6

*Use base Part No. GT10324 with all bluing plates.

E. Ream taper-pin holes with a taper-pin reamer, and fit new taper-pins as specified.
F. Countersink smaller diameter taper-pin hole 0.025 inch deep at 45 degrees.
G. Install taper-pins and peen small end of taper-pin into countersunk hole so that pin locks in place.
H. Dress both ends of taper-pin flush with link(s), and centerpunch larger end of pin for identification.
I. Mark ASA 1447/2 on new alternator ring assembly link(s) near part number, and Mark ASA 1447/3 on new generator ring assembly link(s).
J. Carefully inspect ring assembly and pass for further service if found satisfactory.



Generator and Alternator Attachment Ring Repair
Figure 201

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM — GENERAL

1. **Description — General**

This section will be separated into four parts:

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM — DESCRIPTION / OPERATION

1. **General**

This section will discuss the accessory gearbox low oil pressure warning system for the Gulfstream I and effectivity will be noted within each Section for configuration changes to the system.

2. **Description**

A. Aircraft 1 - 86 and 114 having ASC 68A and Aircraft 87 - 135, 322 and 323.

This system is installed as standard equipment in Aircraft 87 - 135, 322 and 323. It is an optional change for earlier aircraft. The purpose of the system is to provide a means of warning the crew when the accessory gearbox oil pressure drops below 20 psi. The 20 psi oil pressure was selected to provide maximum protection during higher engine rpm ranges. Gearbox oil pressure is proportional to engine rpm, for this reason Dowty Rotol specifications dictate that oil pressure must not be below 20 psi above 12,000 rpm. The system utilizes a pressure switch for each engine-driven gearbox which will complete an electrical circuit to a warning light in the master caution light display. The two warning lights are labeled L GEARBOX PRESS and R. GEARBOX PRESS and are located in the outboard row of warning lights. The pressure switches are located on the frame just aft of each gearbox in the upper section of each nacelle.

B. Aircraft 1 - 135 and 114 having ASC 68B and Aircraft 136 - 200.

This system is installed on Aircraft 136 - 200 and is an optional ASC on Aircraft 1 - 135, 322 and 323.

ASC 68A was designed to warn the crew when the gearbox oil pressure dropped below 20 psi. This setting was selected to provide maximum protection in higher engine rpm ranges such as cruise.

The gearbox oil pressure is proportional to engine rpm. For this reason specifications published by Dowty Rotol dictate that oil pressure should not be below 20 psi at engine speeds above 12,000 rpm. This was the warning point selected for ASC 68A. Since the system was installed, there have been several cases of the warning system operating during descent and landing, when the engine is normally below 12,000 rpm, with the gearbox oil pressure dropping below 20 psi. The warning in this case is inadvertent and annoying. ASC 68B prevents this situation by disarming the system when the power lever is at 12,000 rpm or below. This action prevents the annoyance without endangering the basic purpose of the system.

The system uses pressure switch for each gearbox, a microswitch in the pedestal for each power lever, and a warning light, part of the master warning system for each gearbox. The pressure switch, mounted on a frame aft of the gearbox is set to open at 23 psi and above and close at 20 psi and below. The power lever switch is set to close at a setting of 12,000 rpm and remain closed above this setting.

3. **Emergency DC Operation — Effect on System**

A. Aircraft 1 - 120, 322 and 323 having ASC 68B and Aircraft 136 - 200.

The left gearbox pressure warning light will be operative within the limitations of the master warning system during emergency dc operation.

B. Aircraft 121 - 200

Both gearbox pressure warning lights will be operative within the limitations of the master warning system during emergency dc operation.

4. **Operation**

(See Figure 1 or Figure 2)

A. Aircraft 1 - 86 and 114 having ASC 68A and Aircraft 87 - 120.

Since the right and left gearbox pressure systems are wired differently they will be discussed separately.

The right gearbox pressure switch is wired so that it completes a ground to the appropriate circuit of the master warning system. This is necessary since the warning capsule utilized is a negative switching capsule. It has a common positive feed from the master warning circuit breaker, which is fed from the

main dc bus. When the pressure drops below 20 psi in the gearbox oil system, the right pressure switch closes. This places a ground on pin C, and completes a circuit to the master warning control box. This box in turn converts the information into a signal for the R GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 $\pm$ 1 psi turning off its respective light.

The left gearbox pressure switch is wired so that a circuit from the essential dc bus through the L OIL TEMP circuit breaker exists. This circuit passes through the left gearbox pressure switch to the appropriate circuit of the master warning system. This is necessary since the warning capsule utilized is a positive switching capsule. When the pressure drops below 20 psi in the gearbox oil system, the left pressure switch closes. This places a positive feed on pin C, and completes a circuit to the master warning control box. This box in turn converts the information into a signal for the L GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 $\pm$ 1 psi turning off its respective light.

WARNING: ILLUMINATION OF GEARBOX PRESSURE DURING MAINTENANCE REQUIRES IMMEDIATE SHUT-DOWN OF ENGINE, UNLESS ENGINE RPM IS LESS THAN 12,000 RPM. GEARBOX SHOULD BE CHECKED FOR PROPER OIL LEVEL AND LEAKS.

B. Aircraft 1 - 120 having ASC 68B.

Both gearbox pressure warning lights will be operative within the limitations of the master warning system during emergency dc operation.

The gearbox low oil pressure system operates as described in paragraph 4. A. except with the addition of pedestal microswitches which opens each circuit when the respective lever is below the 12,000 RPM position.

C. Aircraft 121 - 135 not having ASC 68B and Aircraft 136 - 200.

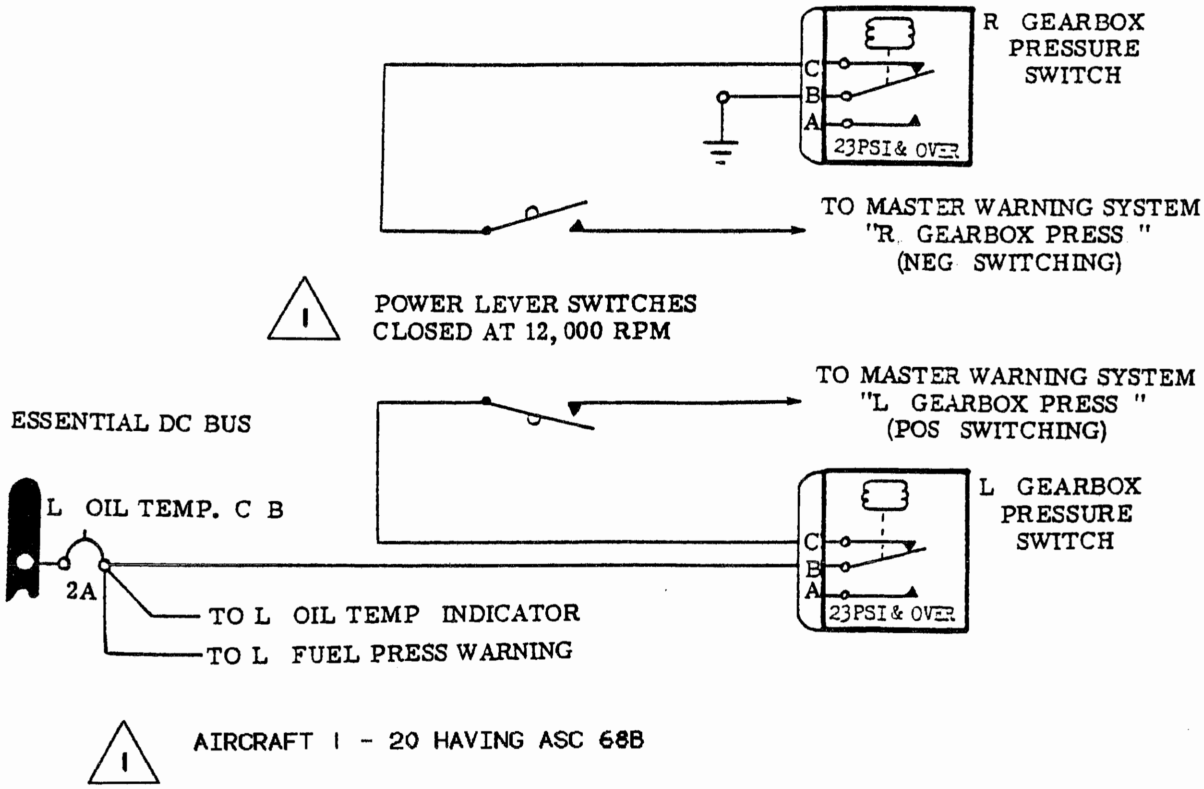
Since the right and left gearbox pressure systems are identical only the left system will be described.

The left gearbox pressure switch is wired so that a circuit from the essential dc bus through the L OIL TEMP circuit breaker exists. This circuit passes through the left gearbox pressure switch and the left power lever switch to the appropriate circuit of the master warning system. When the pressure drops below 20 psi in the gearbox oil system, the left pressure switch closes. This places a positive feed on pin C, and if the engine is operating above 12,000 rpm, completes a circuit to the master warning light system and turns on the L GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 $\pm$ 1 psi turns off its respective light.

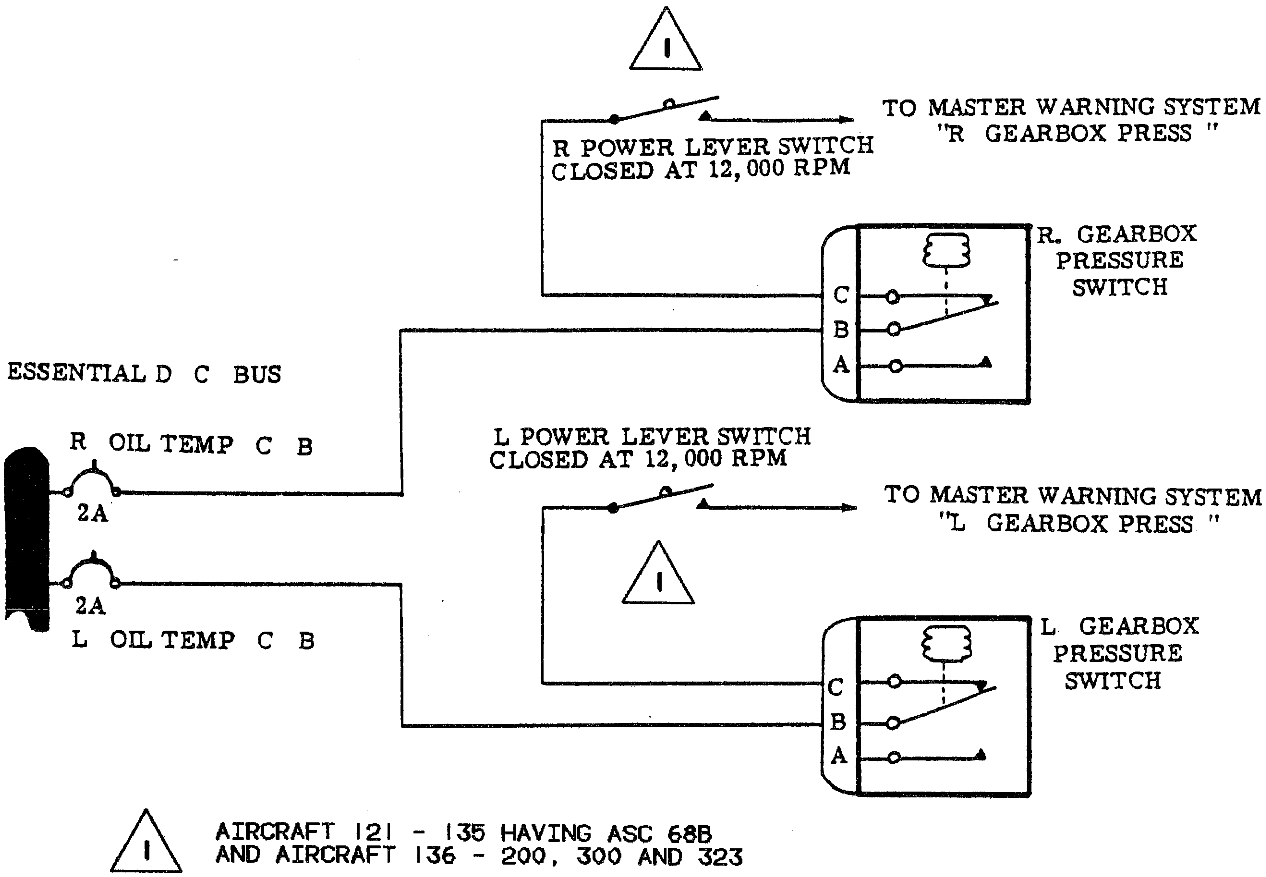
WARNING: ILLUMINATION OF GEARBOX PRESSURE DURING MAINTENANCE REQUIRES IMMEDIATE SHUT-DOWN OF ENGINE, UNLESS ENGINE RPM IS LESS THAN 12,000 RPM. GEARBOX SHOULD BE CHECKED FOR PROPER OIL LEVEL AND LEAKS.

D. Aircraft 121 - 135 having ASC 68B and Aircraft 136 - 200.

The gearbox low oil pressure system operates as described in paragraph 4. A. thru 4. C. except with the addition of pedestal microswitches which opens each circuit when the respective lever is below the 12,000 RPM position.



Gearbox Low Oil Pressure Warning Circuit — Schematic (Aircraft 1 - 120)
Figure 1.



Gearbox Low Oil Pressure Warning Circuit — Schematic (Aircraft 121 - 200, 322 and 323)
Figure 2.

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM — MAINTENANCE PRACTICES

1. **Pressure Warning Switch — Operational Test**

This procedure applicable to Aircraft 1 - 135 having ASC 86B and Aircraft 136 - 200.

To perform an operation test of the gearbox low oil pressure warning system daily, before first flight proceed as follows:

A. Energize main and essential dc bus.

B. Move right power lever (with engine dead) forward to a point where gearbox low pressure light comes on, then return to idle.

C. Repeat procedure for left system (with engine not operating) forward to a point where gearbox low pressure light comes on, then return to idle.

NOTE: Daily operational test of this system is required.

2. **Gearbox Low Oil Pressure System — Functional Test**

NOTE: Aircraft 1 - 135, 322 and 323 having ASC 68B and Aircraft 136 - 200 having gearbox low oil pressure system with power lever cutoff feature. Aircraft 1 - 120, 322 and 323 having ASC 68A and Aircraft 121 135 having gearbox low oil pressure system without power lever cutoff feature.

A. Remove accessory gearbox access panel to gain access to pressure switch.

B. Disconnect pressure line from elbow on switch.

C. Install protective cap on line.

D. Connect pressure source to switch with air hose, valve and pressure gage.

CAUTION: DO NOT APPLY PRESSURE TO SWITCH AT THIS TIME.

E. Energize main and essential dc bus, with power lever retarded (If applicable), observe that associated GEARBOX light on master warning lights panel is off.

F. Advance throttle until GEARBOX PRESS light comes on. (If applicable)

G. Slowly apply increasing pressure to switch, GEARBOX light should go out when 22.0 to 24.0 psi air pressure is applied to switch.

H. Slowly increase air pressure to 26 psi.

I. Slowly decrease pressure, GEARBOX light should come on before 20 psi air pressure is obtained.

J. Decrease air pressure to zero.

K. Retard throttle (if applicable) GEARBOX PRES light will go out.

L. Remove pressure source. Disconnect air hose, valve and gage from pressure switch.

M. Connect pressure line to switch.

N. Inspect area for presents of foreign objects, security of all attachments and leaks.

O. Operate engine and inspect for leakage.

P. Ensure GEARBOX PRES light is out above 12,000 rpm.

Q. Shut down engine, de-energize main and essential dc buses.

R. Install gearbox access panel.

3. Pressure Warning Switch — Functional Test

NOTE: A minute amount of oxide is deposited on the silver contacts during each operation of the pressure warning switches. The oxide is an insulation and after a number of operating cycles, the oxide builds up to a point where the current load is insufficient to burn through the resistance offered by the oxide. Dead-break or no circuit operation will result.

Pressure warning switch may be tested installed or uninstalled on aircraft.

A. Connect unit to a controllable pressure source and connect a 3.0 ampere, 28 volt inductive electrical load in series with both normally open and closed connector pins. The value of inductance should be no less than L/R = 0.015.

B. Slowly apply increasing pressure until switch actuates.

C. Slowly decrease pressure until switch deactuates.

D. Repeat Steps B. and C. above at least ten times.

NOTE: Hydra-Electric P/N 1133-1 (gold contacts) switches are not subject to the type of malfunction discussed above.

4. Pressure Warning Switch — Removal / Installation

A. Removal

(1) Remove accessory gearbox access panel.

(2) Disconnect gearbox oil pressure line from switch and cap line.

(3) Disconnect electrical connector from switch.

(4) Remove switch from mounting bracket.

B. Installation

(1) Install pressure switch on mounting bracket.

(2) Install electrical connector on switch and lockwire.

(3) Install gearbox oil pressure line on switch.

(4) Inspect area for presents of foreign objects, security of all attachments leaks.

(5) Operate engine and inspect for leakage. Ensure GEARBOX PRESS light goes out at speeds above 12,000 rpm.

(6) Shut down engine.

(7) Inspect area for presents of foreign objects, security of all attachments and install access panel.

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM - AIRCRAFT 1 - 86 INCLUDING 114 HAVING ASC 68A AND AIRCRAFT 87 - 120 INCLUDING 322, 323 — DESCRIPTION / OPERATION

1. **Description**

This system is installed as standard equipment in Aircraft 87 through 120 including 322, 323. It is an optional change for earlier aircraft. The purpose of the system is to provide a means of warning the crew when the accessory gearbox
oil pressure drops below 20 psi. The 20 psi oil pressure was selected to provide maximum protection during higher engine rpm ranges. Gearbox oil pressure is proportional to engine rpm, for this reason Dowty Rotol specifications dictate that oil pressure must not exceed 20 psi at 12,000 rpm of engine speed. The system utilizes a pressure switch for each engine-driven gearbox which will complete an electrical circuit to a warning light in the master caution light display. The two warning lights are labeled L GEARBOX PRESS and R. GEARBOX PRESS. They are located just above the OUTSIDE BAT SW warning light at the bottom of the outboard row. The pressure switches are located on the frame just aft of each gearbox in the upper section of each nacelle.

2. **Emergency DC Operation — Effect on System**

The right gearbox pressure warning light will be inoperative during emergency dc operation.

The left gearbox pressure warning light will be operative within the limitations of the master warning system during emergency dc operation.

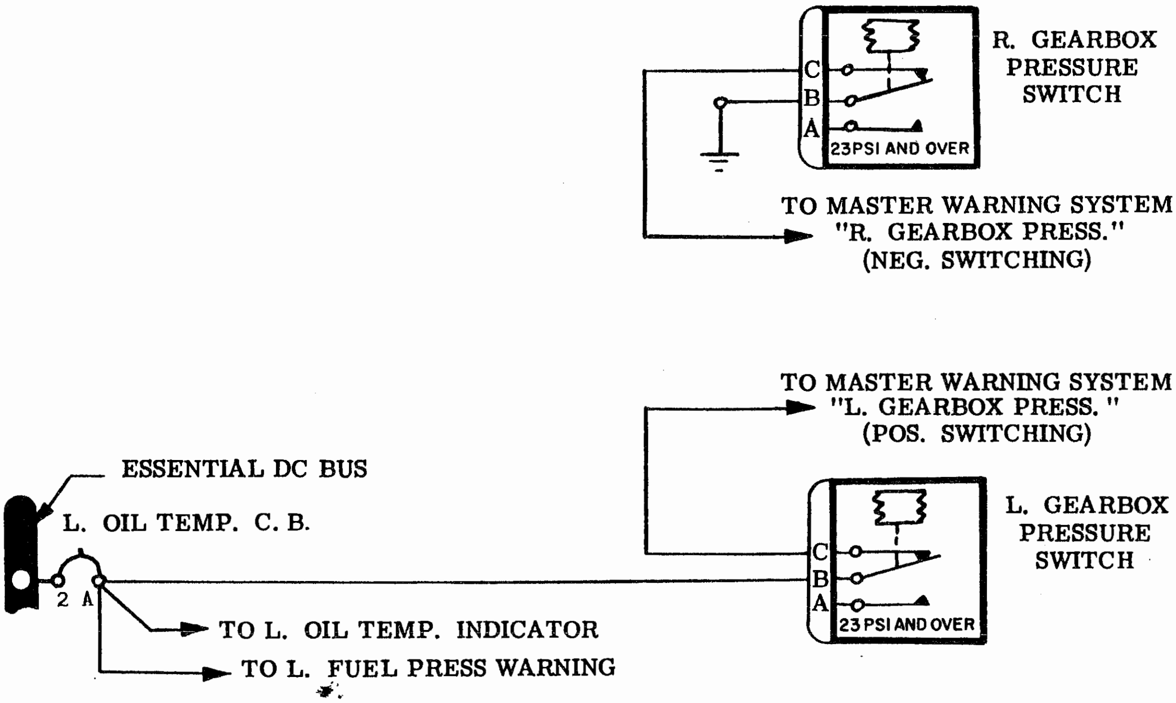
3. **Operation** (See Figure 1)

Since the right and left gearbox pressure systems are wired differently they will be discussed separately.

The right gearbox pressure switch is wired so that it completes a ground to the appropriate circuit of the master warning system. This is necessary since the warning capsule utilized is a negative switching capsule. It has a common positive feed from the master warning circuit breaker, which is fed from the main dc bus. When the pressure drops below 20 psi in the gearbox oil system, the right pressure switch closes. This places a ground on pin C, and completes a circuit to the master warning control box. This box in turn converts the information into a signal for the R GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 ± 1 psi turning off its respective light.

The left gearbox pressure switch is wired so that a circuit from the essential dc bus through the L OIL TEMP circuit breaker exists. This circuit passes through the left gearbox pressure switch to the appropriate circuit of the master warning system. This is necessary since the warning capsule utilized is a positive switching capsule. When the pressure drops below 20 psi in the gearbox oil system, the left pressure switch closes. This places a positive feed on pin C, and completes a circuit to the master warning control box. This box in turn converts the information into a signal for the L GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 ± 1 psi turning off its respective light.

WARNING: ILLUMINATION OF EITHER GEARBOX LIGHT IN FLIGHT REQUIRES IMMEDIATE FEATHERING OF THE AFFECTED ENGINE, UNLESS RPM OF AFFECTED ENGINE IS BELOW 10,400.



Gearbox Low Oil Pressure Warning Circuit — Schematic
Figure 1

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM — MAINTENANCE PRACTICES

1. **Pressure Warning Switch — Function Test**

NOTE: A minute amount of oxide is deposited on the silver contacts during each operation of the pressure warning switches. The oxide is an insulation and after a number of operating cycles, the oxide builds up to a point where the current load is insufficient to burn through the resistance offered by the oxide. Dead-break or no circuit operation will result.

A. Connect unit to a controllable pressure source and connect a 3.0 ampere, 28 volt inductive electrical load in series with both normally open and closed connector pins. The value of inductance should be no less than L/R = 0.015.
B. Slowly apply increasing pressure until switch actuates.
C. Slowly decrease pressure until switch deactuates.
D. Repeat Steps B. and C. above at least ten times.

For procurement of spares, purchase Hydra-Electric P/N 1133-1 (gold contacts) direct from Hydra-Electric, Burbank, California. This part is not subject to the type of malfunction discussed above.

2. **Gearbox Low Oil Pressure Switch — Functional Test**

NOTE: Aircraft 1 - 120 including 322 and 323 having ASC 68B and Aircraft 121 - 200 have gearbox low oil pressure system with power lever cutoff feature. Aircraft 1 - 86 having ASC 68A and Aircraft 87 - 120 have gearbox low oil pressure system without power lever cutoff feature.

A. Remove accessory gearbox access cover to gain access to pressure switch.
B. Disconnect pressure line from elbow on switch.
C. Install protective cap on line.
D. Connect pressure source to switch with air hose valve and pressure gage.

CAUTION: DO NOT APPLY PRESSURE TO SWITCH AT THIS TIME.

E. Energize main and essential dc buses; with power lever advanced, observe that associated GEARBOX light on master warning lights panel is on.
F. Slowly apply increasing pressure to switch, GEARBOX light should go out when 22.0 to 24.0 psi air pressure is applied to switch.
G. Slowly increase air pressure to 26 psi.
H. Slowly decrease pressure, GEARBOX light should come on before 20 psi air pressure is obtained.
I. Decrease air pressure to zero; disconnect air hose, valve, and gage from pressure switch.
J. Connect pressure line to switch.
K. Install accessory gearbox access cover.

3. **Pressure Switch — Removal / Installation**
A. Removal
   (1) Remove accessory gearbox access panel.
   (2) Disconnect gearbox oil pressure line from switch and cap line.
   (3) Disconnect electrical connector from switch.
   (4) Remove switch from mounting bracket.
B. Installation
   (1) Install pressure switch on mounting bracket.
   (2) Install electrical connector on switch and lockwire.
   (3) Install gearbox oil pressure line on switch.
   (4) Run engine and inspect for leakage. Ensure GEARBOX PRESS. light goes out at speeds above 12,000 rpm.
   (5) Shut down engine and install access panel.

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM
AIRCRAFT 121 - 135 — DESCRIPTION / OPERATION

1. **Description**

This system is installed as standard equipment on Aircraft 121 - 135. The purpose of the system is to provide a means of warning the crew when the accessory gearbox oil pressure drops below 20 psi. The 20 psi oil pressure was selected to provide maximum protection during higher engine rpm ranges. Gearbox oil pressure is proportional to engine rpm, for this reason Dowty Rotol specifications dictate that oil pressure must not exceed 20 psi at 12,000 rpm of engine speed. The system utilizes a pressure switch for each engine driven gearbox which will complete an electrical circuit to a warning light in each master caution light display panel.

The two warning lights are labeled L GEARBOX PRESS and R GEARBOX PRESS (located on Master Warning Panel). The pressure switches are located on the frame just aft of each gearbox in the upper section of each nacelle.

2. **Emergency DC Operation — Effect on System**

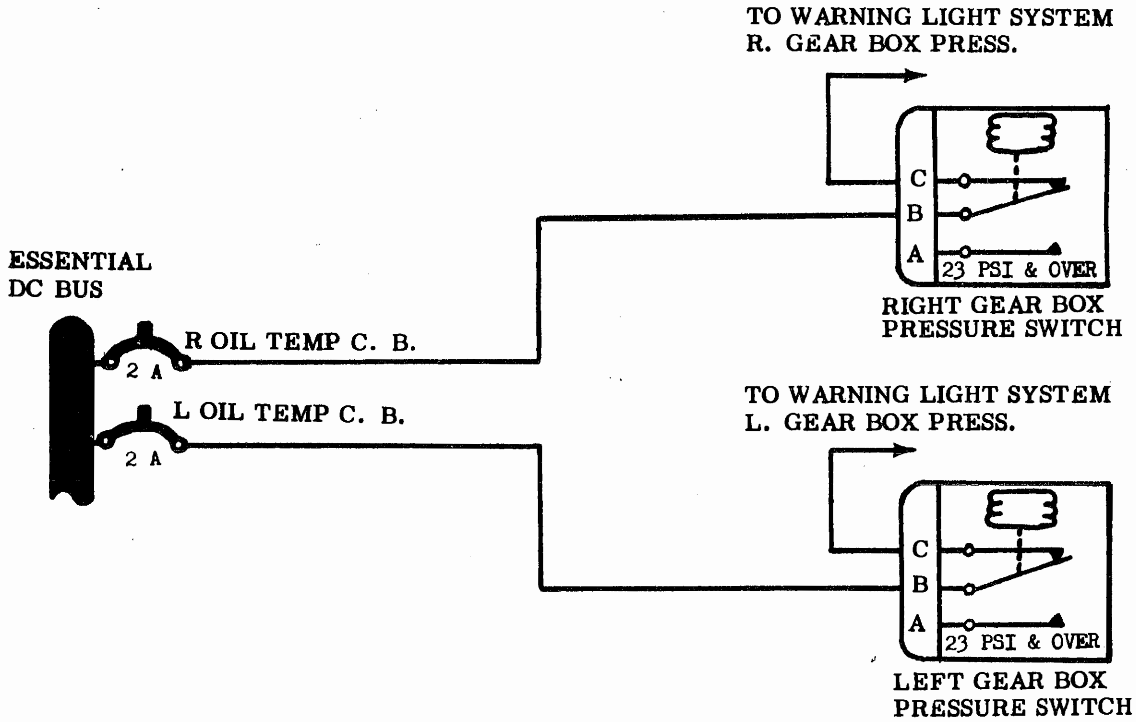
Both gearbox pressure warning lights will be operative within the limitations of the Master Warning System during EMERGENCY DC operation.

3. **Operation** (See Figure 1)

Since the right and left gearbox pressure systems are identical only the left system will be described.

The left gearbox pressure switch is wired so that a circuit from the ESSENTIAL DC BUS through the L OIL TEMP circuit breaker exists. This circuit passes through the left gearbox pressure switch to the appropriate circuit of the Master Warning System. When the pressure drops below 20 psi in the gearbox oil system, the left pressure switch closes. This places a positive feed on pin C, and completes a circuit to the Master Warning light system and turns on the L GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 ± 1 psi turning off its respective light.

WARNING: ILLUMINATION OF EITHER GEARBOX LIGHT IN FLIGHT REQUIRES IMMEDIATE FEATHERING OF THE AFFECTED ENGINE.



Gearbox Low Oil Pressure Warning Circuit — Schematic
Figure 1

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM — MAINTENANCE PRACTICES

For Maintenance Practices, Refer to 83-1-1 page 201

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM - AIRCRAFT 1 - 120 INCLUDING 322, 323 HAVING ASC 68B — DESCRIPTION / OPERATION

1. **Description**

This is an optional service change on Aircraft 1 - 120 including 322, 323.

Service change 68A designed to warn the crew when the gearbox oil pressure dropped below 20 psi. This setting was selected to provide maximum protection in higher engine rpm range such as cruise.

The gearbox oil pressure is proportional to engine rpm. Specifications published by Dowty Rotol dictate that the oil pressure should not be below 20 psi at engine speeds above 12,000 rpm. This was the warning point selected for Aircraft Service Change 68A. Since the system was installed, there have been serveral cases of the warning system operating during descent and landing, when the engine is normally below 12,000 rpm, with the gearbox oil pressure dropping below 20 psi. The warning in this case is inadvertent and annoying. Aircraft Service Change 68B prevents this situation by disarming the system when the power lever is at 12,000 rpm or below. This action prevents the annoyance without endangering the basic purpose of the system.

The system uses a pressure switch for each gearbox, a microswitch in the pedestal for each power lever and a warning light, part of the master warning system, for each gearbox. The pressure switch, mounted on a frame aft of the gearbox is set to open at 23 psi and above and close at 20 psi and below. The power lever switch is set to close at a setting of 12,000 rpm and remain closed above the setting.

2. **Emergency DC Operation — Effect on System**

The right gearbox pressure warning light will be inoperative during emergency dc operation.

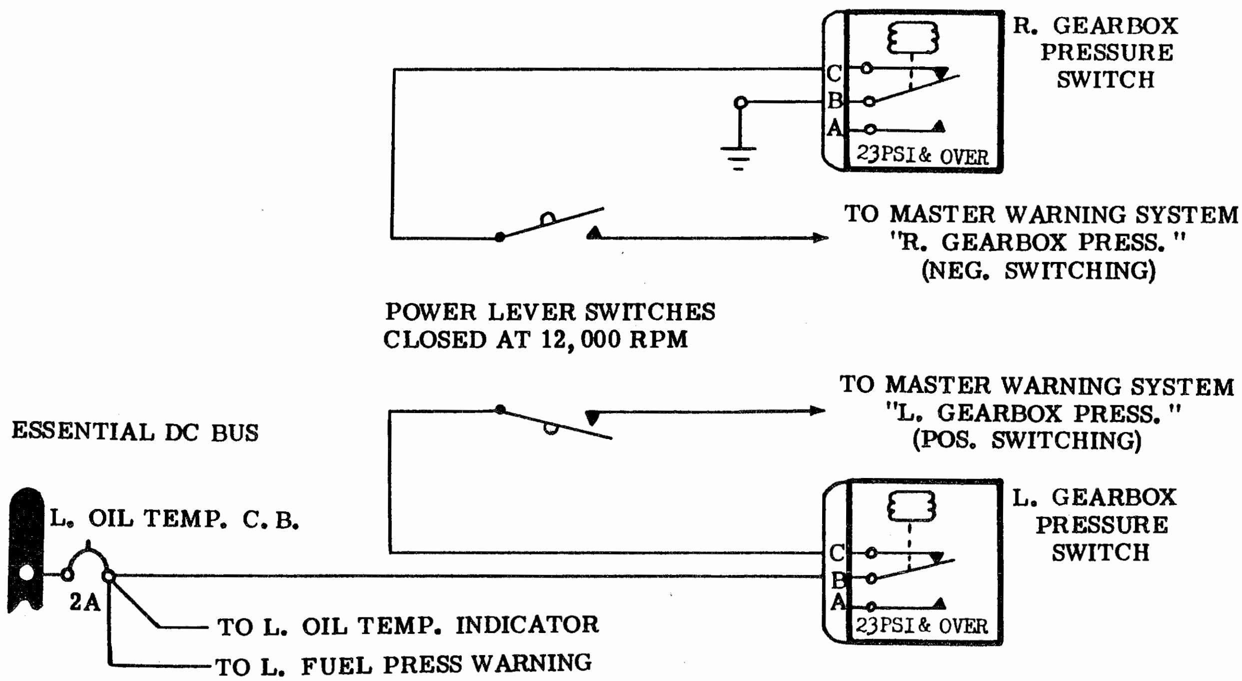
The left gearbox pressure warning light will be operative within the limitations of the master warning system during emergency dc operation.

3. **Operation** (See Figure 1)

Since the right and left gearbox pressure systems are wired differently they will be discussed separately.

The right gearbox pressure switch is wired, through a power lever switch, so that completes a ground to the appropriate circuit of the master warning system. This is necessary since the warning capsule utilized is a negative switching capsule. It has a common positive feed from the master warning circuit breaker, which is fed from the main dc bus. When the pressure drops below 20 psi in the gearbox oil system, the right pressure switch closes. This places a ground on pin C, and if the engine is operating above 12,000 rpm, completes a circuit to the master warning control box. This box in turn converts the information into a signal for the R GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 ± 1 psi turning off its respective light.

The left gearbox pressure switch is wired so that a circuit from the essential dc bus through the L OIL TEMP circuit breaker exists. This circuit passes through the left gearbox pressure switch and the left power lever switch to the appropriate circuit of the master warning system. This is necessary since the warning capsule utilized is a positive switching capsule. When the pressure drops below 20 psi in the gearbox oil system, the left pressure switch closes. This places a positive feed on pin C, and if the engine is operating above 12,000 rpm, completes a circuit to the master warning control box. This box in turn converts the information into a signal for the L GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 ± 1 psi turning off the respective light.



Gearbox Low Oil Pressure Warning Circuit — Schematic
Figure 1

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM — MAINTENANCE PRACTICES

1. **Pressure Warning Switch — Operational Test**

To perform an operation test of the gearbox low oil pressure warning system daily, before first flight proceed as follows:

A. With electrical power on the aircraft place battery switch in normal.
B. Move right power lever (with engine dead) forward to a point where gearbox low pressure light comes on, then return to idle.
C. Repeat procedure for left system (with engine dead).

NOTE: Daily operational test of this system is required.

2. **Pressure Warning Switch — Functional Test**

NOTE: Aircraft 1 - 120 including 322, 323 having ASC 68B and aircraft 121 - 200 having gearbox low oil pressure system with power lever cutoff feature. Aircraft 1 - 86 including 114 having ASC 68A and Aircraft 87 - 120 including 322, 323 having gearbox low oil pressure system without power lever cutoff feature.

A. Remove accessory gearbox access cover to gain access to pressure switch.
B. Disconnect pressure line from elbow on switch.
C. Install protective cap on line.
D. Connect pressure source to switch with air hose, valve and pressure gage.

CAUTION: DO NOT APPLY PRESSURE TO SWITCH AT THIS TIME.

E. Energize main and essential dc buses; with power lever advanced, observe that associated GEARBOX light on master warning lights panel is on.
F. Slowly apply increasing pressure to switch, GEARBOX light should go out when 22.0 to 24.0 psi air pressure is applied to switch.
G. Slowly increase air pressure to 26 psi.
H. Slowly decrease pressure, GEARBOX light should come on before 20 psi air pressure is obtained.
I. Decrease air pressure to zero; disconnect air hose, valve, and gage from pressure switch.
J. Connect pressure line to switch.
K. Install accessory gearbox access cover.

3. **Gearbox Low Oil Pressure Switch — Functional Test**

A. Remove accessory gearbox access cover to gain access to pressure switch.
B. Disconnect pressure line from elbow on switch.
C. Install protective cap on line.
D. Connect pressure source to switch with air hose, valve and pressure gage.

CAUTION: DO NOT APPLY PRESSURE TO SWITCH AT THIS TIME.

E. Energize main and essential dc buses; with power lever advanced, observe that associated GEARBOX light on master warning lights panel is off.
F. Advance throttle until GEARBOX PRESS light comes on.
G. Slowly apply increasing pressure to switch, GEARBOX light should go out when 22.0 to 24.0 psi air pressure is applied to switch.
H. Slowly increase air pressure to 26 psi.
I. Slowly decrease pressure, GEARBOX light should come on before 20 psi air pressure is obtained.
J. Decrease air pressure to zero.
K. Retard throttle, GEARBOX PRES light will go out.
L. Remove pressure source. Disconnect air hose, valve and gage from pressure switch.
M. Connect pressure line to switch.
N. Run engine and inspect for leakage.
O. Ensure GEARBOX PRES light is out above 12,000 rpm.
P. Shut down engine, de-energize main and essential dc buses.
Q. Install gearbox access cover.

4. **Low Oil Pressure Switch — Removal / Installation**

A. Refer to Section 83-1-1 this chapter.

ACCESSORY GEARBOX LOW OIL PRESSURE WARNING SYSTEM-
AIRCRAFT 121 - 135 HAVING ASC 68B AND AIRCRAFT 136 - 200 — DESCRIPTION / OPERATION

1. **Description**

This is installed on Aircraft 136 - 200 and is an optional ASC on Aircraft 121 - 135 including 322, 323.

Aircraft Service Change 68A was designed to warn the crew when the gearbox oil pressure dropped below 20 psi. This setting was selected to provide maximum protection in higher engine rpm ranges such as cruise.

The gearbox oil pressure is proportional to engine rpm. Specifications published Dowty Rotol dictate that oil pressure should not be below 20 psi at engine speeds above 12,000 rpm. This was the warning point selected for Aircraft Service Change 68A. Since the system was installed, there have been several cases of the warning system operating during descent and landing, when the engine is normally below 12,000 rpm, with the gearbox oil pressure dropping below 20 psi. The warning in this case is inadvertent and annoying. Aircraft Service Change 68B prevents this situation by disarming the system when the power lever is at 12,000 rpm or below. This action prevents the annoyance without endangering the basic purpose of the system.

The system uses pressure switch for each gearbox, a microswitch in the pedestal for each power lever, and a warning light, part of the master warning system for each gearbox. The pressure switch, mounted on a frame aft of the gearbox is set to open at 23 psi and above and close at 20 psi and below. The power lever switch is set to close at a setting of 12,000 rpm and remain closed above this setting.

2. **Emergency DC Operation — Effect on System**

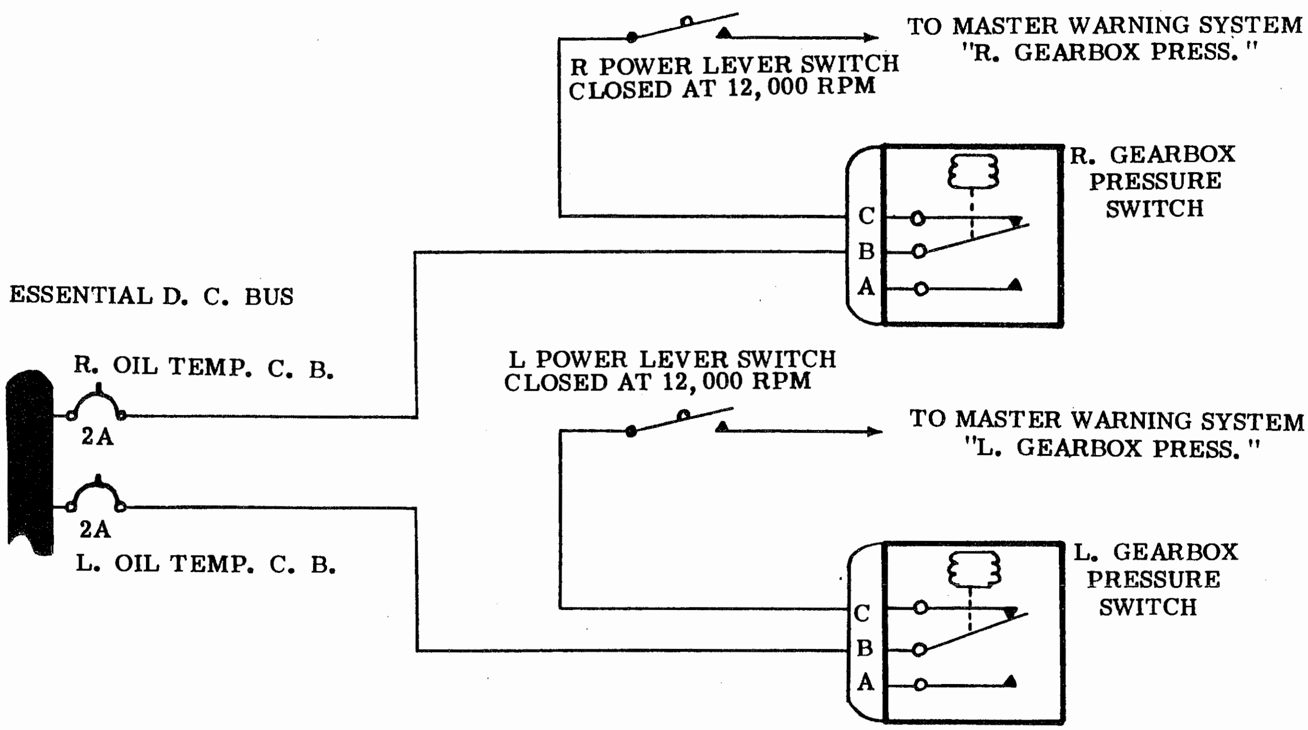
Both gearbox pressure warning lights will be opertive within the limitations of the master warning system during emergency dc operation.

3. **Operation** (See Figure 1)

Since the right and left gearbox pressure systems are identical only the left system will be described.

The left gearbox pressure switch is wired so that a circuit from the essential dc bus through the L OIL TEMP circuit breaker exists. This circuit passes through the left gearbox pressure switch and the left power lever switch to the appropriate circuit of the master warning system. When the pressure drops below 20 psi in the gearbox oil system, the left pressure switch closes. This places a positive feed on pin C, and if the engine is operating above 12,000 rpm, completes a circuit to the master warning light system and turns on the L GEARBOX PRESS light. The pressure switch should open on an ascending pressure of 23 ± 1 psi turns off its respective light.

WARNING: ILLUMINATION OF EITHER GEARBOX LIGHT IN FLIGHT REQUIRES IMMEDIATE FEATHERING OF THE AFFECTED ENGINE, UNLESS ENGINE RPM IS ABOVE 10,400.



Gearbox Low Oil Pressure Warning Circuit — Schematic
Figure 1